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Vol. I

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Marshall Space Flight Center, Alabama

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16. ABSTRACT

This report presents the history and development of the Skylab Airlock Module and the Payload Shroud, NASA Contract No. NAS9-6555, from initial concept through final design, related test programs, mission performance and lessons learned.

Although some problems were encountered, the Airlock Module performed successfully throughout the three manned Skylab missions.

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* NOTE: The Final Technical Report for the Payload Shroud is presented in MDC Report G4679A.

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SECTION 1 INTRODUCTION

1.1 PURPOSE AND SCOPE

The Airlock Module was one element of a very successful Skylab Program. This report documents the technical results of the Airlock Project, i.e., the conception, development, and verification of flight and ground support hardware, and includes the controlling program functions that resulted in the on-schedule delivery of a flightworthy spacecraft. Problems and their solutions are also documented so that experience gained during all phases of this program may be used as building blocks for future spacecraft programs.

To provide a full understanding of the Airlock Module, each system is described in terms of requirements, configuration, verification, and mission performance.

To provide a better understanding of the open-ended test concept, Airlock test philosophy is discussed through its evolution into the final, implemented test plan.

To demonstrate the importance of management control functions to a successful program, the technical disciplines of reliability, safety, and engineering scheduling and control are discussed.

To illustrate the method and extent of activity required to support a long-term, complex space operation system, mission operation support is detailed.

To allow further refinement of the Nation's space efforts, conclusions derived from total program results are discussed and recommendations for future programs are made.

Additionally, direct support of the Manned Space Flight Center (MSFC) and other NASA centers, during both prelaunch and mission operations, is summarized, as is the results of the New Technology Reporting Program.

The Airlock Program Contract (NAS9-6555) covers the Airlock Module, including the ATM Deployment Assembly (DA), the Fixed Airlock Shroud (FAS), the Payload Shroud (PS), and all associated Ground Support Equipment (GSE) and trainers. These elements, with the exception of the Payload Shroud, were designed, fabricated, and verified at the McDonnell Douglas, St. Louis, Missouri Facility and are covered in this report (MDC Report E0899, Airlock Module Final Technical Report).

The Payload Shroud was designed, fabricated, and verified at the McDonnell Douglas, Huntington Beach, California Facility and is discussed in MDC Report G4679A, Payload Shroud Final Technical Report.

These two reports, MDC Reports E0899 and G4679A, together comprise the Skylab Airlock Project Final Technical Report.

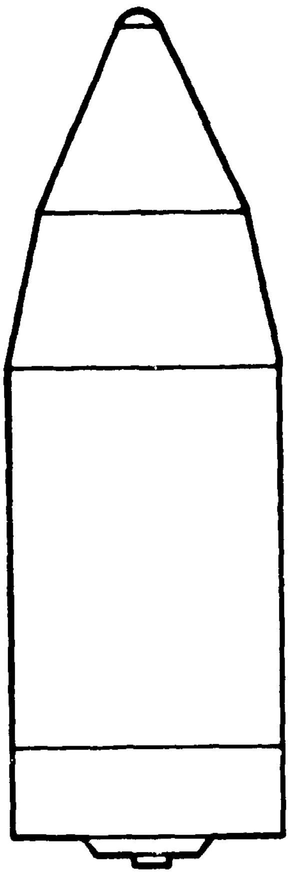
1.2 SUMMARY

The Airlock Module (AM), Fixed Airlock Shroud (FAS), Deployment Assembly (DA), and Payload Shroud (PS), shown in Figures 1-1 and 1-2, and all associated trainers and Ground Support Equipment were designed, fabricated and verified under NASA Contract as basic elements of the Skylab cluster, shown in Figure 1-3. This orbiting laboratory was launched aboard a Saturn V launch vehicle on 14 May 1973 and was subsequently manned by crews launched in modified Apollo Command and Service Modules on Saturn IB launch vehicles (shown in Figure 1-4 and Figure 1-5). The Skylab supported solar, celestial, and earth observations; medical, scientific, engineering, and technology experiments, during three manned missions of 28, 59 and 84 days, respectively, from 25 May 1973 through 8 February 1974. As shown in Figure 1-6, the active operation of the as-flown mission exceeded the planned mission by 31 days and total mission duration exceeded that planned by 35 days.

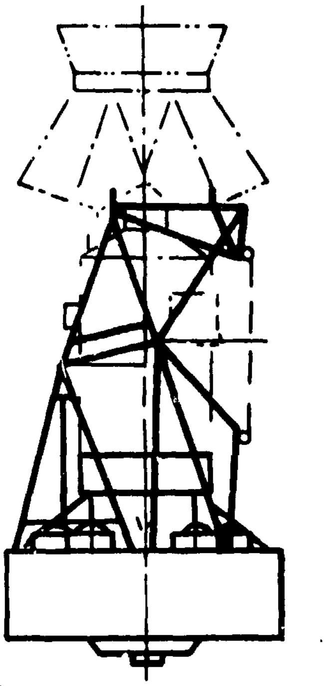
1.2.1 Airlock Features

The AM provided the following features:

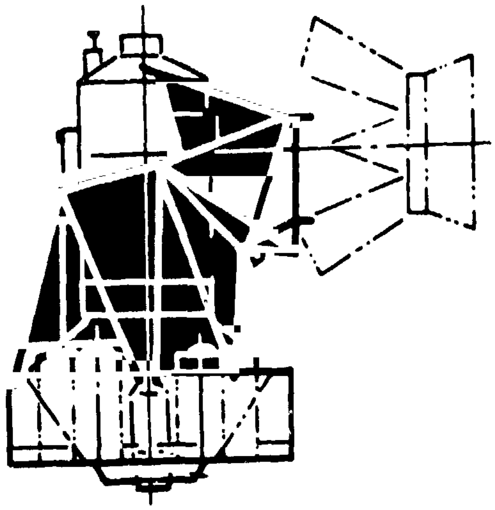
- Interconnecting passage between MDA and OWS.
- Lock, hatch and support system for extravehicular activity (EVA).
- Purification of the Skylab atmosphere.
- Thermal control of the Skylab atmosphere (cooling only for MDA and OWS).
- Atmospheric supply and control.
- Apollo Telescope Mount (ATM) launch support and orbital deployment.
- Payload protection during launch (Payload Shroud).



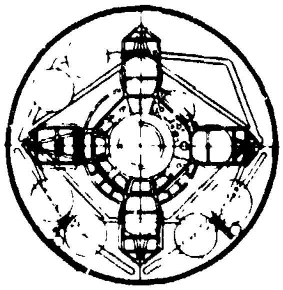
Launch Configuration



Payload Shroud Jettisoned



Deployment Assembly Deployed
(FAS Cutaway)



View Looking Forward

Orbital Configuration

FIGURE 1-1 AIRLOCK MODULE GENERAL ARRANGEMENT

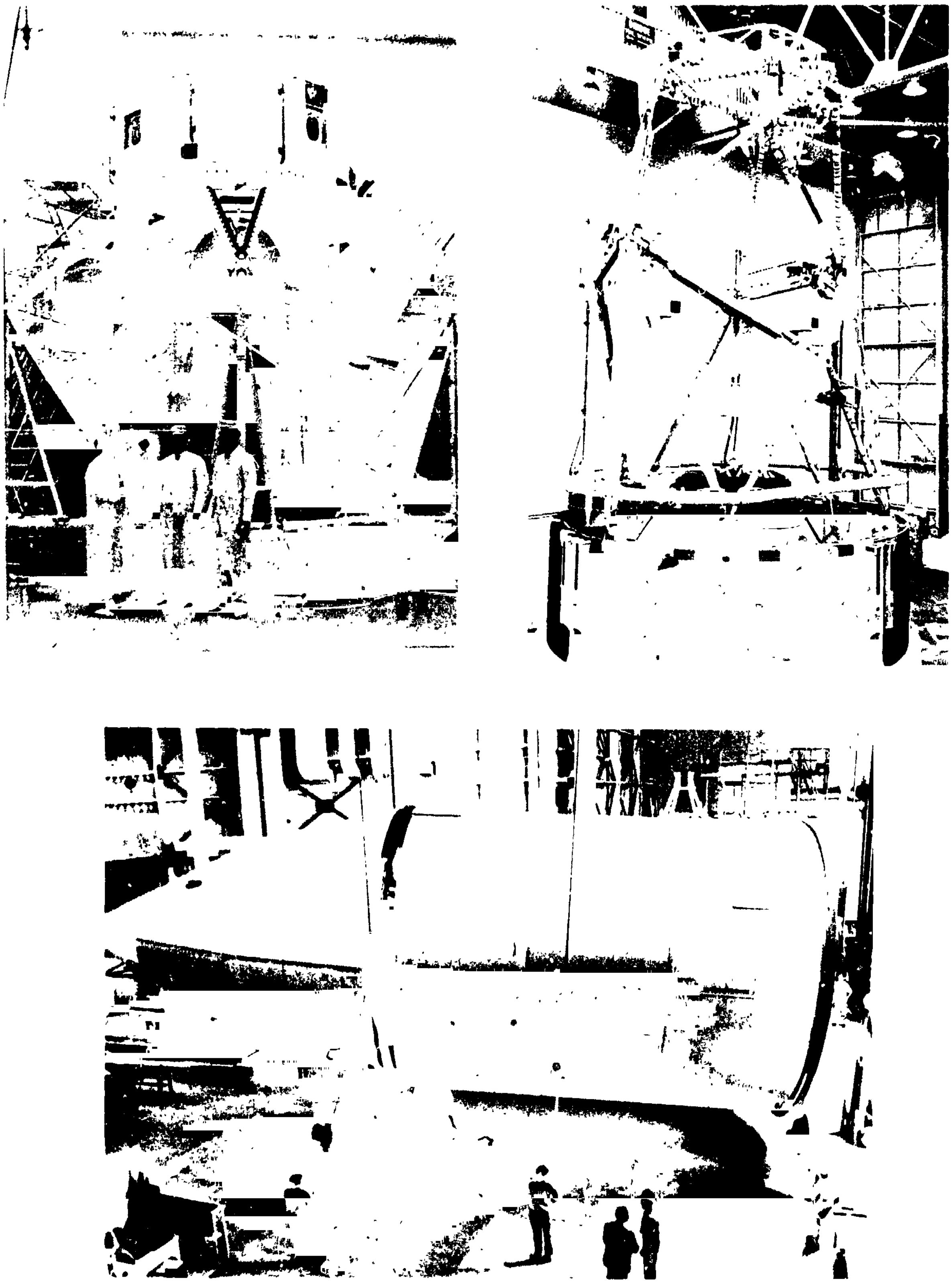


FIGURE 1–2 AIRLOCK COMPONENTS

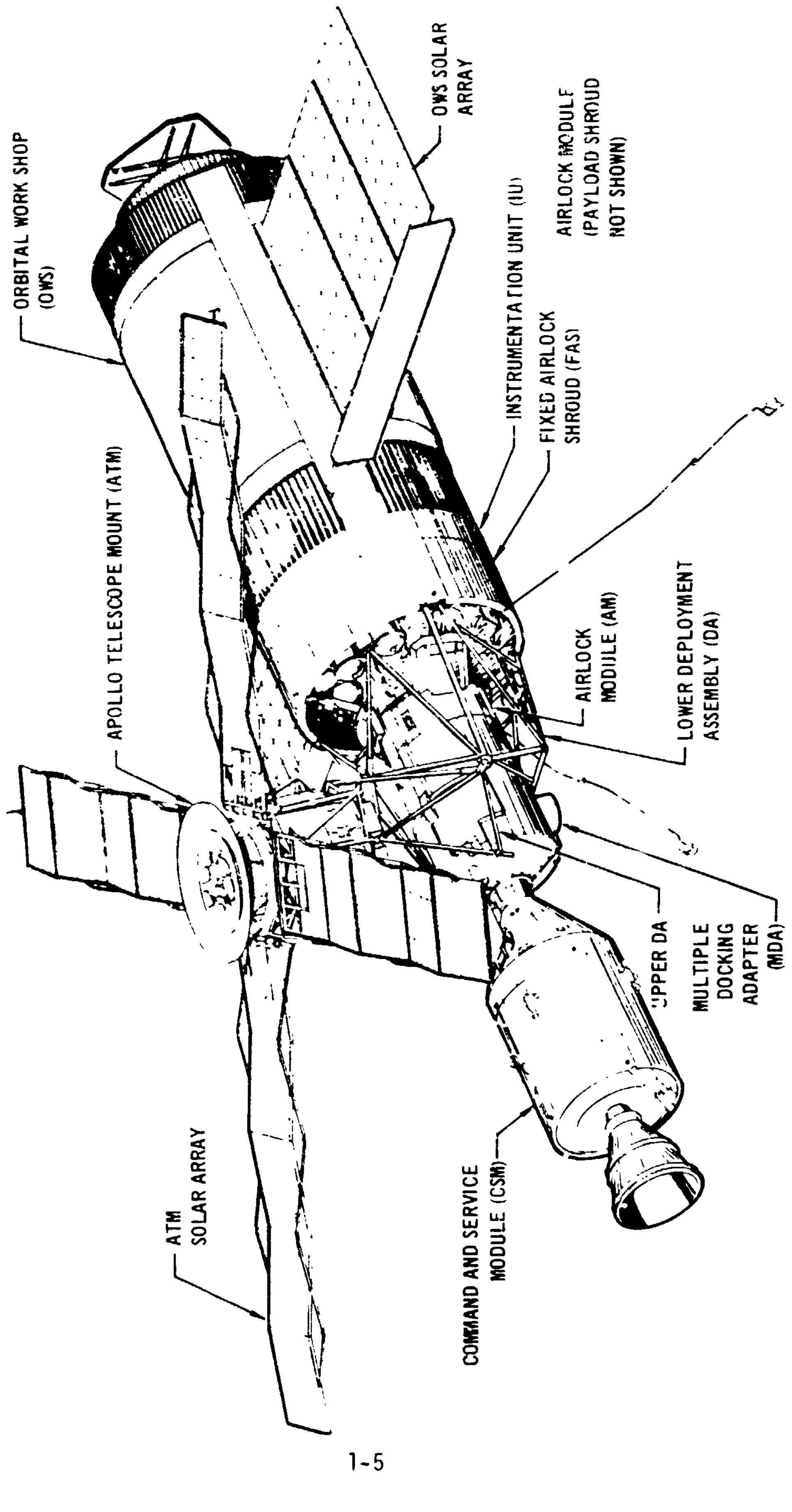


FIGURE 1-3 SKYLAB CLUSTER CONFIGURATION – MANNED MISSION

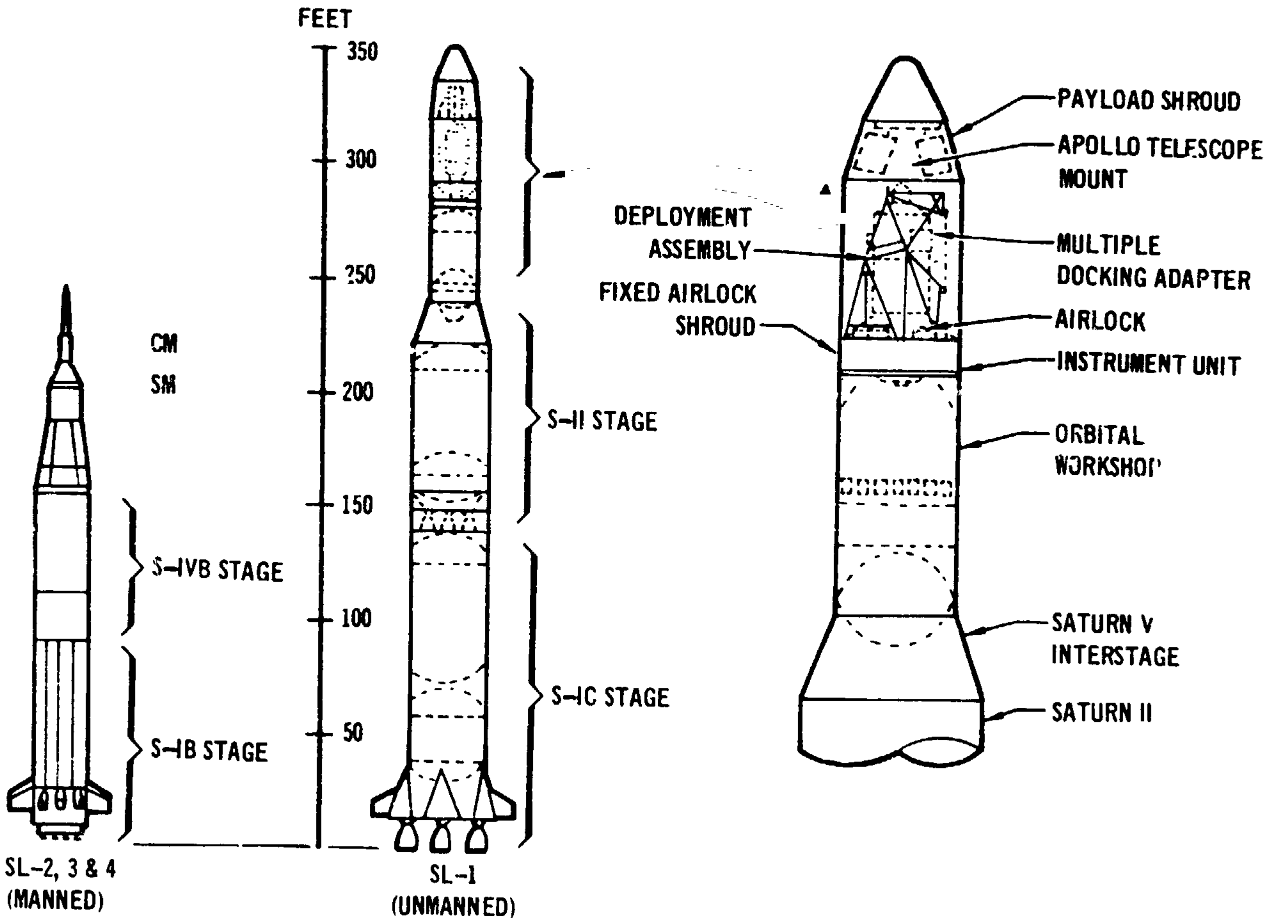


FIGURE 1-4 SKYLAB LAUNCH CONFIGURATIONS

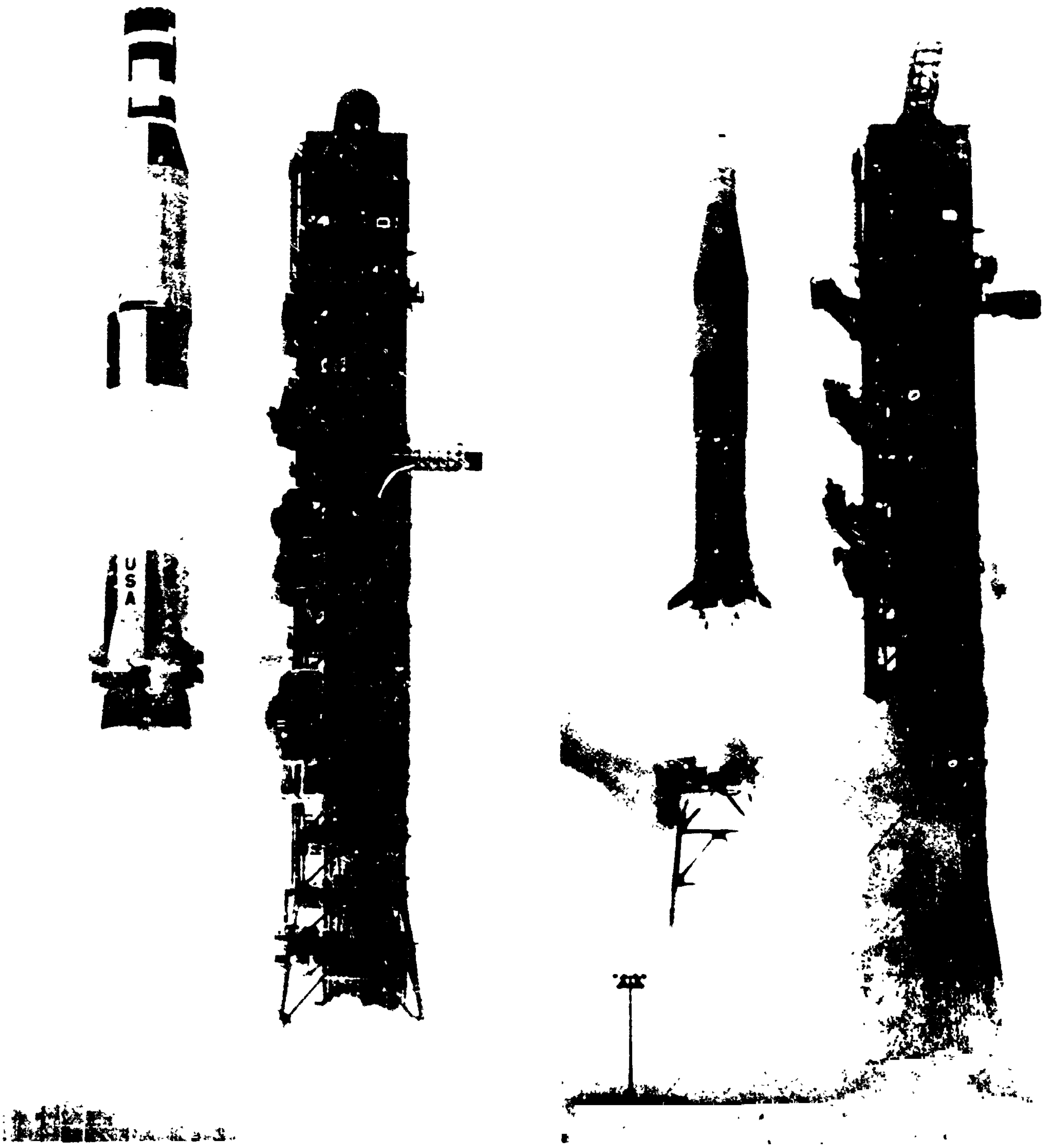


FIGURE 1-5 SKYLAB SL-1 AND SL-2 LAUNCHES

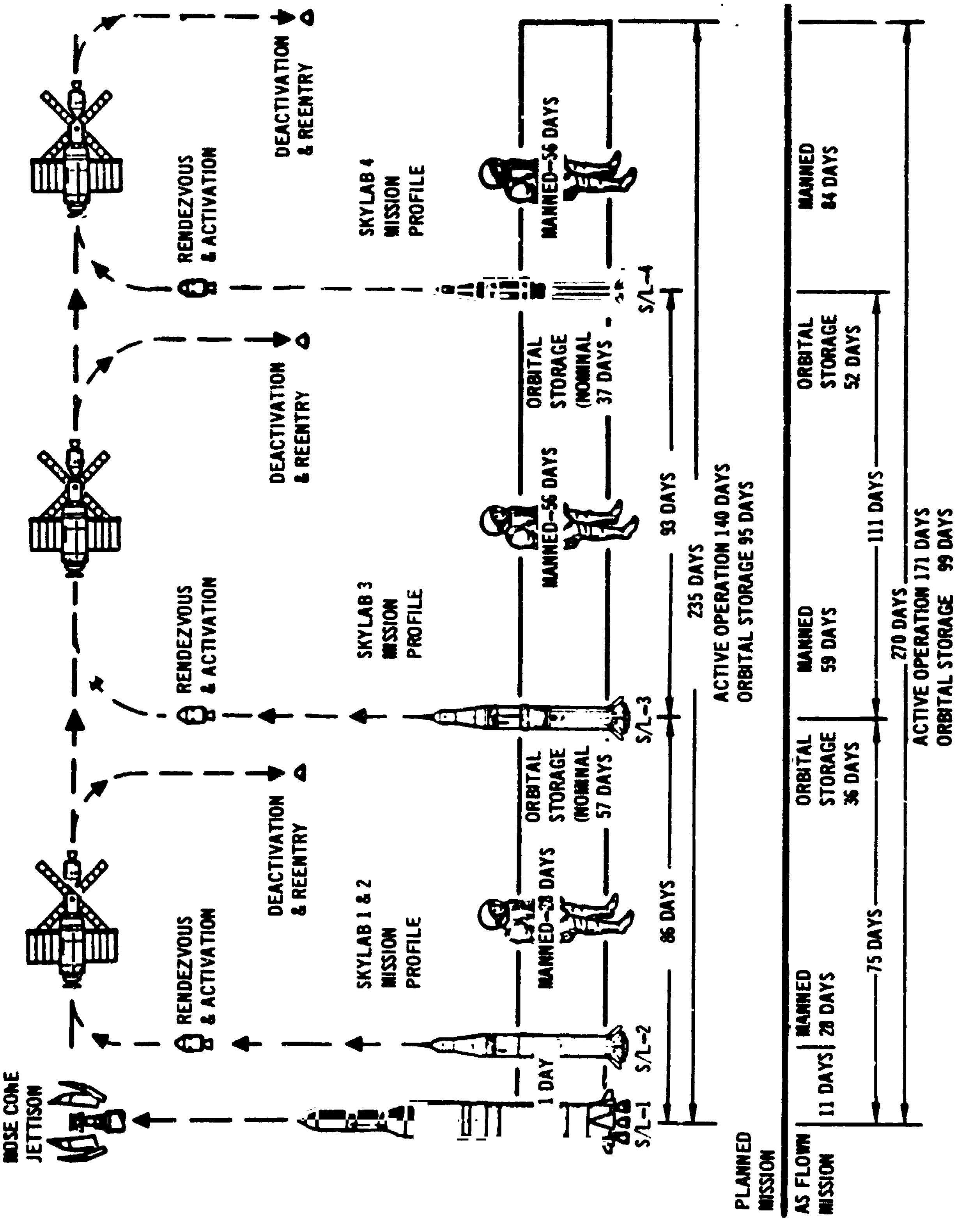


FIGURE 1–6 SKYLAB MISSION PROFILES

- Electrical power conditioning control, and distribution.
- Real and delayed time data.
- Cluster intercommunication.
- Cluster failure warning.
- Command system link with ground network.
- VHF ranging link for CSM rendezvous.
- Controls and displays.
- Teleprinter.
- Experiment installation of D024 sample panels.
- Experiment antennas (EREP and radio noise burst monitor).
- ATM C&D Panel cooling.

1.2.2 Airlock Module Weight and Dimensions

• Gross AM Weight	15,166 lb.
• AM Working Volume	610 cuft.
• AM Overall Length	211.54 in.
Tunnel Assembly	
Length	153 in.
Diameter	65 in.
Volume	322 cuft.
Structure Transition Section (STS)	
Length	47 in.
Diameter	120 in.
Volume	288 cuft.
Pressurized AM to OWS Passageway	
Length	11.54 in.
Diameter	42.5 in.

1.2.3 FAS Weight and Dimensions

• Gross Weight	22,749 lb.
• Length	80 in.
• Diameter	260 in.

The FAS provided the capability of structurally supporting the Apollo Telescope Mount (ATM), AM, MDA, and Payload Shroud (PS) during the launch phase of the mission. The structural shell consisted of thick skin and ring construction with local intercostals for structural support of the ATM Deployment Assembly (DA).

1.2.4 DA Weight and Dimensions

• Gross Weight	3,744 lb.
• Length (Upper DA)	122 in.
• Length (Lower DA)	194 in.

The DA consisted of two tubular truss assemblies connected by a pair of trunnion joints which allowed the upper truss assembly to rotate through 90° to deploy the ATM. The DA rotation system consisted of two redundant springs that retarded rotation and redundant deployment reels, cables, gear train and motors to pull the ATM into the deployed position. A redundant pyrotechnically-operated latch actuator allowed mechanical disengagement of the stabilization struts, and a spring-loaded latch mechanism retained the ATM-DA in the deployed position as shown in Figure 1-3. The DA included two major carrier wire assemblies to interconnect the cluster electrical power systems and to connect the ATM with the ATM C&D Panels in the MDA. Detailed information on Airlock structures/mechanical systems and on mass properties may be found in Paragraphs 2.2 and 2.3, respectively.

1.2.5 Payload Shroud (PS)

The PS consisted of a cylindrical section and a biconical nose section; both sections were thick skinned, ring reinforced, monocoque structures. The PS supported the ATM during the prelaunch and launch phases and provided aerodynamic protection during launch and contamination protection for the AM, MDA, and ATM through S-II on-orbit retrofire. After achieving orbit, the PS was jettisoned as part of the unmanned cluster activation sequence; it was separated radially into four quadrants via a discrete latching system and a longitudinal thrusting joint system. Both of these separation systems were powered by redundantly fired linear explosive devices.

Configuration characteristics were:

• Gross PS Weight	25,473 lbs.
• PS Overall Length	674 in.
Cylinder Length	350 in.
Biconical Nose Length	324 in.
• PS Diameter	260 in.

The PS design was verified by separation element and panel tests, discrete latching system tests, and three full-scale separation tests conducted by the NASA in the Plum Brook Space Power Facility vacuum chamber. In addition, the full-scale PS was installed during the vibro-acoustic testing at JSC.

Payload Shroud S/N 03 was launched on Skylab 1 and was subsequently jettisoned in-orbit without problem; all functions performed as planned at the correct attitude and the designed separation velocity was imparted.

Complete details of the requirements, design configuration, verification, and mission performance of the PS is given in MDC Report G4679A, Payload Shroud Final Technical Report.

1.2.6 Environmental/Thermal Control Systems (ECS/TCS)

The AM ECS/TCS consisted of the following subsystems:

- The gas system permitted prelaunch purge, stored high pressure O_2 and N_2 regulated pressure and distribution for cabin atmosphere, and other uses.
- The atmospheric control system provided moisture removal, carbon dioxide and odor removal, ventilation and cabin gas cooling. Moisture was removed from the cluster atmosphere by condensing heat exchangers and molecular sieves. Carbon dioxide and odor were also removed by the molecular sieve system. Ventilation was provided by fans and condensing heat exchanger compressors. Gas cooling was provided by the condensing and cabin heat exchangers.
- The condensate system provided the capability of removing atmospheric condensate from the condensing heat exchangers, storing it, and disposing of it. In addition the condensate system provided the capability of removing gas from the liquid gas separator and disposing of it as well as providing a vacuum source for servicing/deservicing activities.
- The suit cooling system provided astronaut cooling during EVA and IVA by circulating temperature controlled water through the astronauts suit umbilical, Liquid Cooled Garment (LCG), and Pressure Control Unit (PCU).
- The active cooling system consisted of two separate, redundant loops for active cooling of the suit cooling module, atmospheric control modules, selected experiment modules and coldplate mounted electrical/electronic equipment.
- The ATM C&D Panel and EREP cooling system provided cooling to the ATM C&D Panel and to EREP components by circulating water to this equipment.

- The passive thermal system utilized thermal coatings, thermal curtains and insulations to control the gain and loss of heat both internally and externally.

Details of these systems are given in Paragraphs 2.4, 2.5, and 2.6.

1.2.7 Electrical Power System

The AM housed eight nickel-cadmium (Ni-Cad) batteries and their chargers and regulators to power the many electrical devices aboard the Skylab. These eight Power Conditioning Groups conditioned power from the Orbital Workshop Solar Array every orbit.

Power Conditioning Group (PCG) outputs were applied to the various AM EPS buses by appropriate control switching provided on the STS instrument panel or by ground control via the AM Digital Command System (DCS). Each PCG provided conditioned power to using equipment and recharged the batteries during the daylight period. A comprehensive description of the EPS is given in Paragraph 2.7.

1.2.8 Sequential System

The Sequential System of the Airlock controlled mission events to establish the initial orbital configuration of Skylab. The following events were planned to follow launch:

- Payload Shroud jettison.
- Discone antenna deployment.
- Deployment Assembly activation to position the ATM.
- OWS and ATM solar wing deployment.
- Venting operations.
- OWS radiator shield jettison.
- Attitude control transfer.

Although the Airlock sequential system functioned as required, an OWS meteoroid shield malfunction prevented automatic deployment of the OWS solar wings. Sequential System details are in Paragraph 2.8.

1.2.9 Instrumentation System

The Airlock Instrumentation System sensed, conditioned, multiplexed, and encoded vehicle, experiment, and biomedical data for transmission to ground stations in either real-time or recorded delayed time. In addition, it provided data for on-board displays, and through hardlines, enabled readout during ground checkout. The system included the following subsystems:

- Sensors - Used to convert physical quantities being measured (such as temperature or pressure) into proportional electrical signals and Signal Conditioners - consisting of interface circuits used to condition incompatible signals.
- Regulated Power Converters - Devices used to provide stable excitation voltages to the instrumentation hardware.
- PCM Multiplexer/Encoder - System used to provide time sequenced and coded data for transmission to the Spaceflight Tracking and Data Network (STDN).
- Tape Recorder/Reproducer - Devices used to acquire and store between station data for subsequent playback to STDN in delayed time.

A description of these subsystems is provided in Section 2.9.

1.2.10 Communications System

The Communications System transmitted and received voice, instrumentation data, the television data between: crew members in the Skylab and on EVA; crew members and ground tracking stations; Skylab systems and ground tracking stations; and Skylab and the rendezvousing Command/Service Module. The Communications System consisted of the following subsystems:

- Audio System - Used in conjunction with the Apollo Voice Communications Systems to provide communications among the three crewmen and between Skylab and the Spaceflight Tracking and Data Network (STDN).
- Digital Command System (DCS) - A sophisticated, automatic command system which provided the STDN with real-time command capabilities for the AM, OWS, and MDA. The Digital Command System permitted control of experiments, antennas, and cluster system functions.
- Teleprinter - In conjunction with the AM receiver/decoders, the teleprinter provided on-board paper copies of data transmitted by the STDN.
- Time Reference System (TRS) - Provided time correlation to the Pulse Code Modulation (PCM) Data System, automatic reset of certain DCS commands, automatic control of the redundant DCS receiver/decoders, and timing data to the Earth Resources Experiment Package (EREP) and on-board displays in the AM and OWS.

- Telemetry Transmission System - Used in conjunction with the Airlock Antenna System, the Telemetry System provided RF transmission capability to the STDN during prelaunch, launch, and orbit for real-time data. delayed time data, delayed time voice, and emergency voice (during rescue transmission), in both stabilized and unstabilized vehicle attitudes. This system included four telemetry transmitters, three of which could be operated simultaneously during orbital phases.
- Antenna System - Consisted of a modified Gemini Quadriplexer, two modified Gemini UHF Stub Antennas, four RF Coaxial Switches, two Antenna Booms, two Discone Antennas and a helical VHF Ranging Antenna.
- Rendezvous Systems - Consisting of a VHF Ranging System and four tracking lights, these systems facilitated rendezvous of Command Modules (SL-2, -3, and -4) with the Saturn Workshop (SWS). The Airlock equipment comprised a VHF Transceiver Assembly, a Ranging Tone Transfer Assembly (RTTA), and a VHF Ranging Antenna.

Detailed information on the Communications System may be found in Paragraph 2.10.

1.2.11 Caution and Warning System (C&W)

The Caution and Warning System monitored critical Skylab parameters and provided the crew with audio/visual alerts to imminent hazards and out-of-spec conditions which could lead to hazards. Emergency situations resulted in activation of a Klaxon horn which could be heard throughout the Skylab vehicle. Caution or warning conditions were brought to the crew's attention through crew earphones and speaker/intercom panels. Emergency parameters were defined as:

- MDA/STS fire.
- AM aft compartment fire.
- OWS forward/experiment/crew compartment fire.
- Rapid change in vehicle pressure.

Warning parameters included:

- Low oxygen partial pressure.
- Primary and secondary coolant flow failure.
- AM and ATM regulated power bus out-of-spec.
- Cluster attitude control failure.
- EVA suit cooling out-of-spec.
- AM and CSM crew alerts.

Caution parameters consisted of:

- Mole sieve overtemperature, high carbon dioxide content, flow failure, and sequencing.
- OWS ventilation out-of-spec.
- Rapid condensate tank pressure change.
- Primary and secondary coolant temperature out-of-spec.
- C&W system bus voltage out-of-spec.
- EPS voltages out-of-spec.
- ATM attitude control system malfunctions.
- ATM coolant system malfunctions.

System details may be found in Paragraph 2.11.

1.2.12 Crew Systems

The Airlock functioned as a nerve center for monitoring and operating many complex vehicle systems, either automatically or by the crew.

A. STS - Primary crew controls for AM systems:
   - Electrical Power System.
   - Environmental Control System (ECS)
     Molecular Sieve
     Atmospheric Fans
     Coolant Control
     Condensate System
   - Intravehicular Activity (IVA) Control Panel.
   - Flight Logbook and Records.
   - Cluster Caution and Warning Monitor System.
   - O_2/N_2 Gas Distribution System.

B. Lock Compartment - EVA/IVA Operations
   - EVA/IVA Control Panels (2).
   - Internal and EVA Lighting Controls.
   - Compartment Pressure Displays.
   - Vacuum Source.

C. Aft Compartment
   - OWS Entry Lighting.
   - Thermal Fan and Valve Control.
   - M509 Recharge Station.

Other AM Crew Systems included the following:

- Mobility Aids.
- Internal Lighting.
- Communications - Placement of internal voice communications.
- Stowage.

Additional information concerning Crew Systems may be found in Paragraph 2.12.

1.2.13 Trainers

MDAC-E designed, built, maintained, and updated the NASA Trainer (NT), the Neutral Buoyancy Trainer (NBT), the Zero-G Trainer and the zero-g aft compartment (part task) trainer. In addition, MDAC-E assisted MSFC in converting the Airlock Static Test Article (STA-3), after completion of full-scale vibro-acoustic testing, into the Skylab Systems Integration Equipment (SSIE) unit. Of lower fidelity than the NASA Trainer at JSC, the SSIE was used at MSFC for mission support of crew EVA.

The NBT was used in the MSFC Neutral Buoyance Facility both premission and during the mission to support EVA task training. It was used extensively during the early days of SL-1 mission to develop the techniques and procedures used by the SL-2 crew to release and deploy SAS Wing #1 and to erect a solar shield. The NBT was used throughout the mission for this type of real time mission support.

1.2.14 Experiments

The experiments and experiment support equipment which were mounted on the Airlock are as follows:

- D024 Thermal Control Coatings - Evaluated selected thermal control coatings exposed to near-earth space environment.
- S193 Microwave Radiometer Scatterometer/Altimeter - Determined land/sea characteristics from active/passive microwave measurements.
- S230 Magnetospheric Particle Collection - Measured fluxes and composition of precipitating magnetospheric ions and trapped particles.
- Radio Noise Burst Monitor - Permitted prompt detection of solar flare activity.
- M509 Gaseous Nitrogen (GN_2) Bottle Recharge Station - Supporting hardware for recharging three OWS-stowed N_2 bottles.

Paragraph 2.14 provides detailed information on AM experiment hardware.

1.2.15 Ground Support Equipment (GSE)

Airlock GSE is nonflight hardware and software used in support of the flight article to satisfy a specific support function or to accomplish a defined test. It is used to inspect, test, calibrate, assemble/disassemble, transport, protect, service, checkout, etc., or to otherwise perform a designated function in support of the flight article during development testing, manufacturing assembly, acceptance testing, systems testing, delivery, prelaunch checkout, and launch

GSE used in support of the AM, FAS, DA and PS is categorized as follows:

- Handling, Transportation and Mechanical GSE.
- Electrical/Electronic GSE.
- Servicing and Fluids GSE.

Comprehensive information concerning AM GSE is given in Paragraph 2.15.

1.2.16 Reliability and Safety

The basic approach for achieving Airlock reliability goals of 0.85 for mission success and 0.995 for crew safety was to design reliability into all Airlock systems and maintain that reliability throughout the fabrication, test, and end use phases of the program. Major activities for achieving the necessary Airlock reliability included the following:

- FMEA - A Failure Mode and Effect Analysis identified critical modes of equipment failure and facilitated corrective design changes.
- CIL - A Critical Item List, which included Single Failure Points (SFP's) derived from the FMEA, critical redundant/backup components, and launch critical components, identified primary components requiring test emphasis, contingency procedures, and management control.
- Reliability Model - Contained a quantitative assessment of mission reliability and crew safety for purposes of recommending design improvements to meet AM reliability goals.
- Trade and special studies.
- Design Reviews.
- Potential suppliers evaluation.
- Reporting system for analysis and nonconformance correction.
- NASA Alert investigation and origination of MDAC-E Alerts.

An MDC Report G671, "Airlock Systems Safety Plan," established the requirements for performing all Airlock functions from design through altitude chamber testing and shipment to KSC without injury to personnel or damage to equipment. An Airlock Safety Officer verified compliance with the Safety Plan and provided additional guidance in areas involving potentially hazardous operations not specifically covered by the plan.

Sections 3 and 4 provide comprehensive coverage of the Airlock Reliability and Safety Programs.

1.2.17 Testing

MDAC-E accomplished all structural, dynamic, functional and system tests necessary for the development, qualification, acceptance and verification of the Airlock Module prelaunch checkout capability. Refer to Section 5.

A test plan was implemented for verification tests to define the test documentation used to verify the integrity of the Airlock hardware and to provide historical test data.

Development tests were performed to establish a design concept or to prove the feasibility of an established design concept. Development tests supplemented the design process with performance data on equipment and systems, strength characteristics of structural elements, and the effects of long-term exposure of materials and components to a hard vacuum, as well as to space radiation and corrosive environments.

The Airlock Qualification Test Program was designed to verify the capability of the component hardware to function as specified within the design and performance requirements. This program was based on the Apollo Applications Test Requirements (AATR) Document (NHB 8080.3) which required that equipment qualification testing be varied depending upon the criticality relationship to the crew safety and achievement of the mission objectives.

Qualification of the Airlock systems was accomplished with component level testing and equipment endurance tests. In some instances, components such as the Environmental Control System/Thermal Control System (ECS/TCS) were combined into a module test to verify the system. In this way, endurance testing at the highest practical level verified the equipment performance in mission-simulated environments and mission-simulated duty cycles. Qualification tests verified that the hardware met the performance/design requirements to assure operational suitability of the anticipated environments.

MDAC-E delivered to MSFC a Structural Test Article (STA-1) which was subsequently refurbished into a Dynamic Test Article (STA-3). The structural testing was performed at the MSFC facilities at Huntsville, Alabama and at the JSC facilities at Houston, Texas by a joint NASA/MDAC/MMC test team. The Airlock Dynamic Test Article provided a structurally and dynamically representative vehicle of the Airlock Module. It consisted of a Structural Transition Section, Tunnel and Trusses. The test configuration included the Fixed Airlock Shroud, the ATM Deployment Assembly, the Payload Shroud, and the test article ballas which simulated the equipment and experiments in mass, center of gravity and attachment points. The dynamic configuration was representative of the flight article overall weight, center of gravity and mass moments of inertia.

The objective of the dynamic test was to subject the dynamic test article to the predicted flight level acoustic and vibration environments to esperimentally determine the frequency mode shapes and damping values of the Skylab assembly, equipment and subsystems in both launch and orbit configurations.

Section 5.0 provides detailed information on the Airlock Test Philosophy.

1.2.18 Mission Operations Support

A Skylab Communications Center was installed at MDAC-E to support MSFC prior to and during Skylab launch anf flight operations and to evaluate the Skylab mission performance. In addition, Orbital Assembly flight operations support was provided by MDAC-E via the MSFC Huntsville Operations Support Center (HOSC). MDAC-E support included analyzing off-nominal Skylab conditions, providing additional engineering data, and providing systems simulations for systems performance. For additional information, see Section 7.

1.2.19 New Technology

The initial Airlock concept of a state-of-the-art vehicle had a limited new technology requirement. However evolution, particularily the wet to dry launch configuration change, required advanced state-of-the-art designs, i.e., emergency warning system, a two-gas spacecraft environment, increased electrical power, active cooling for ATM usage, etc.

Of the 451 New Technology Disclosures submitted, 15 were published as NASA Tech Briefs and it is anticipated that additional Tech Briefs will be published subsequent to submittal of this report. Three of the submittals resulted in the preparation of patent applications by the NASA, and one was filed in the U.S. Patent Office. Additional information on New Technology is given in Section 8.

1.2.20 Conclusions

The successful Airlock system performance during the Skylab Program indicates the effectiveness of the MDAC-E design, fabrication, and test activities that preceded the flight mission. It also indicates the effectiveness of the mission support activity in responding to discrepant conditions and providing real-time work around plans.

The major conclusion that can be drawn from a program point of view is that the Airlock program philosophy of maximum use of existing, qualified space hardware with extensive use of system engineering analysis and previous test results to identify the minimum supplemental test program required to complete system verification was proven as a valid, economical approach to a successful mission.

The most important lesson learned, from its impact on future space system planning, is the demonstrated capability of the crewman to function as a major link in the system operation. He demonstrated the capability to function effectively in zero-g for long periods of time and to perform, with proper constraints, tools, and procedures. Additionally, the ability of the crew to perform contingency EVA's and to accomplish difficult repair/maintenance activities will be a significant input to all future manned space programs.

Each section of this report discusses conclusions and recommendations for the system or engineering activity being covered.

Section 9.0 enumerates what MDAC-E considers the most significant "Lessons Learned" from the Airlock Program and their applicability to future programs.

SECTION 2 SYSTEM DESIGN AND PERFORMANCE

2.1 GENERAL

2.1.1 Program Inception

The inception of the MDAC-E Airlock Program dates back to 23 December 1965 when NASA directed MDAC-E to appraise the applicability of Gemini hardware for inclusion in an Airlock Module to support use of a spent S-IVB Stage as a manned shelter and workshop. Subsequently, on 5 April 1966, MDAC-E received a Request for Proposal from the NASA to design, develop, manufacture, and check out a Spent Stage Experiment Support Module (SSESM) for manned launch aboard a Saturn I-B vehicle. This module was to provide an interconnecting tunnel and airlock between the Apollo Command Module and the S-IVB stage, which would subsequently be converted into a manned orbital workshop after its propellant content was expended and it had been purged.

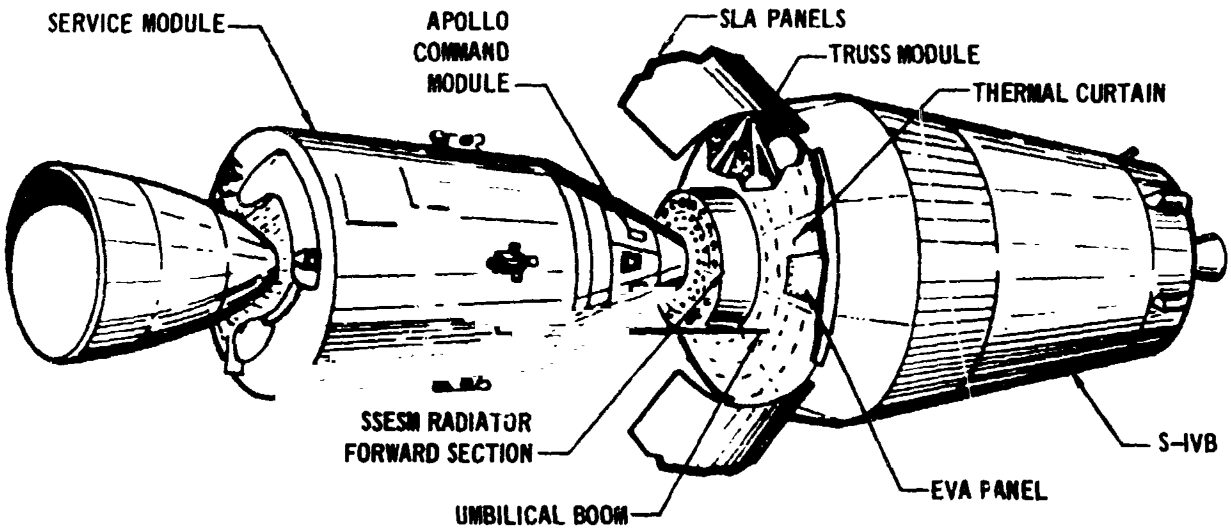
The SSESM proposal was submitted on 17 June 1966 and verbal go-ahead was received on 19 August 1966.

2.1.2 SSESM

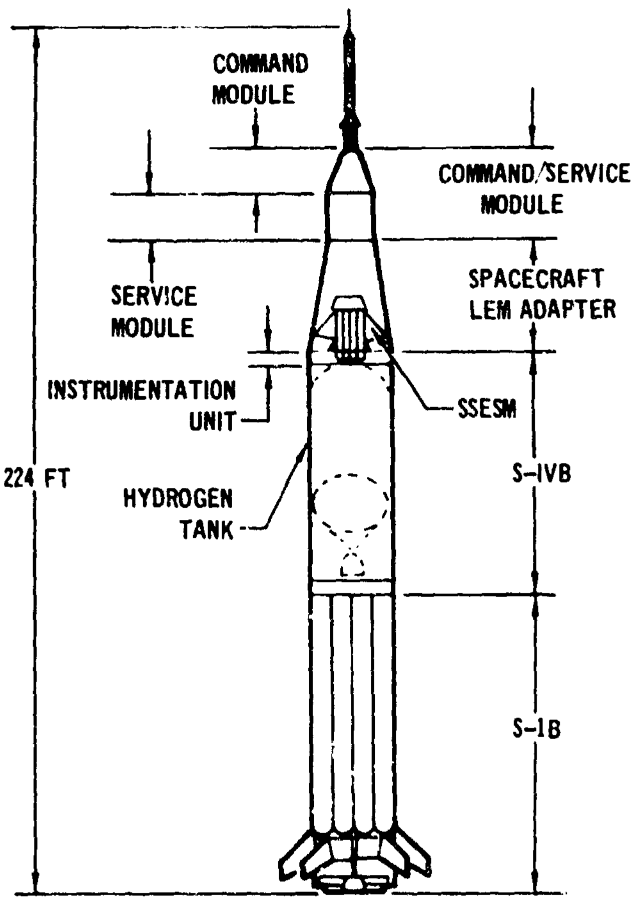
The objective of the SSESM was to demonstrate the economical utilization of an S-IVB spent stage hydrogen tank as a workshop for a manned mission. As shown in Figure 2.1-1 the SSESM was to be launched on a Saturn I-B with an Apollo CSM; it was to be installed in the Spacecraft Lunar Adapter (SLA) on the Lunar Exploration Module (LEM) attach points.

In orbit, the CSM was to separate from the remaining vehicle, rotate 180°, and dock using the SSESM docking adapter.

The SSESM consisted of a tunnel/airlock that provided a habitable pressure vessel between the spent S-IVB stage and the docked CSM and that supported EVA. It included a section of a Gemini adapter/radiator and four mounting trusses that supported cryogenic O_2 and H_2 bottles.



Orbital Configuration



Launch Configuration Saturn IB

FIGURE 2.1-1 SPENT STAGE EXPERIMENT SUPPORT MODULE (SSESM)

During activation, a crew member would perform an EVA to remove and stow the S-IVB dome manhole cover, connect a flexible tunnel extension to complete the pressurized passageway, and connect the O_2 and H_2 boom umbilicals to the Service Module. After purging and furbishing, the S-IVG spent stage would have served as a manned laboratory. SSESM mission philosophy was that of an open-ended flight operation subsequent to the first 14 days with 30-day goal.

Over 98% of the SSESM components were Gemini flight qualified hardware and no additional qualification testing was to have been performed as long as operational requirements were similar to Gemini.

2.1.3 Wet Workshop Evolution

As the program matured and requirements were firmed up, it underwent considerable evolution of mission definition and systems requirements.

Initially, to support additional radiator area and to provide increased pressurized volume for expendables and experiment launch stowage, the Gemini adapter was replaced with a short cylindrical pressure vessel with an axial docking port and external radiators (Refer to Figure 2.1-2). This version was to be launched on a Saturn I-B with a CSM for a 30-day mission; it was designated the Airlock Module.

Subsequently, in December 1966, the pressurized cylindrical compartment was lengthened and four radial docking ports were added (the single axial docking port was retained). Additionally, a solar array system was evaluated for Airlock installation and gaseous O_2 and N_2 tanks were designed for installation on the Airlock trusses (the cryogenic tanks were retained for CSM fuel cell usage). A molecular sieve experiment was added.

This configuration (refer to Figure 2.1-2) was to be launched unmanned on a Saturn I-B with the crew following in a CSM on a second Saturn I-B; crew revisit and station resupply was planned. Additional Saturn I-B launches were required to orbit and rendezvous either a Lunar Mapping and Survey Station Module (SM&SS) or a Lunar Module/Apollo Telescope Mount (LM/ATM) which was to be docked into one of the radial docking ports by remote control. The orbital configuration is shown in Figure 2.1-3.

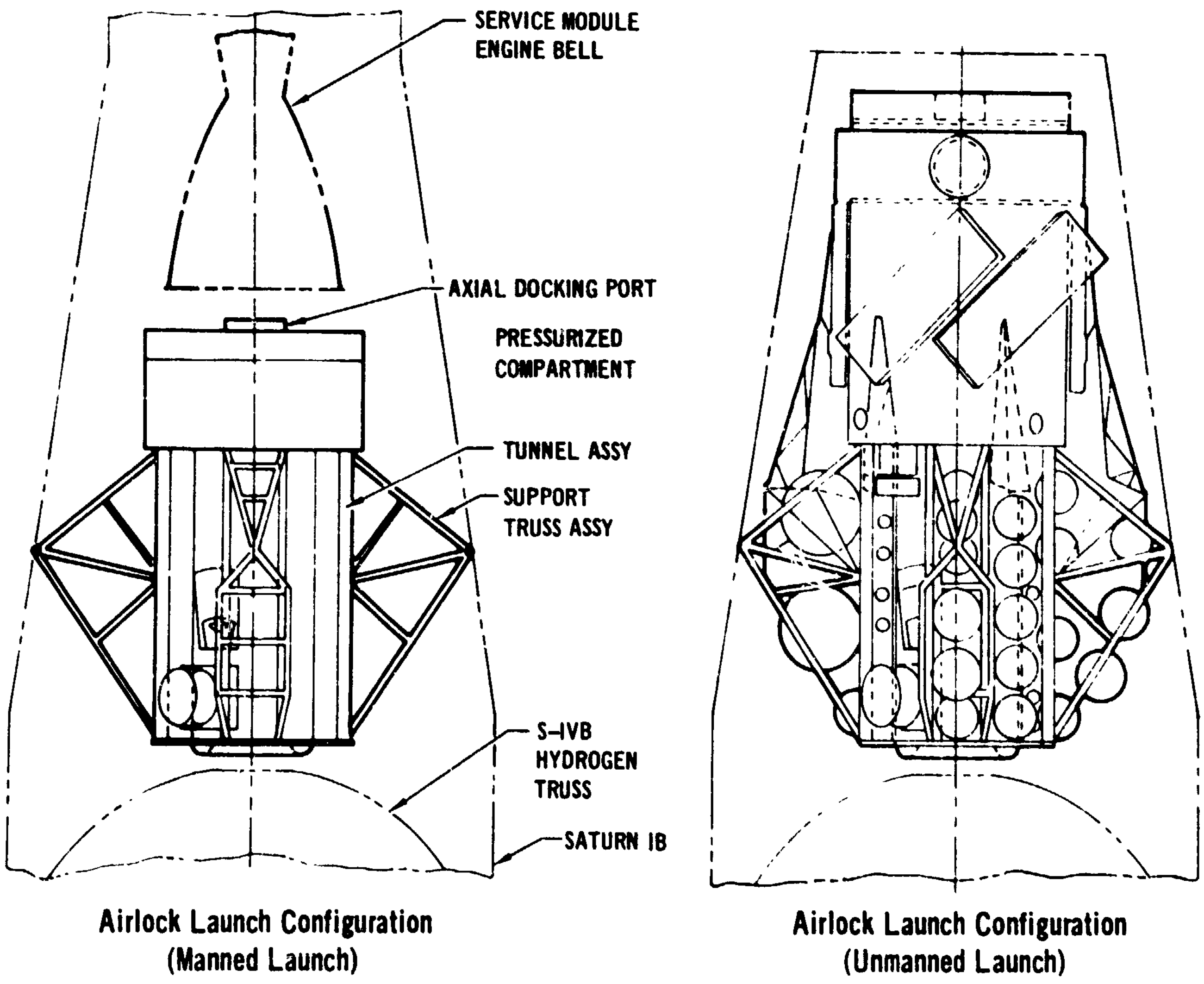


FIGURE 2.1-2 WET WORKSHOP CONFIGURATION EVOLUTION FROM SPENT STAGE EXPERIMENT SUPPORT MODULE

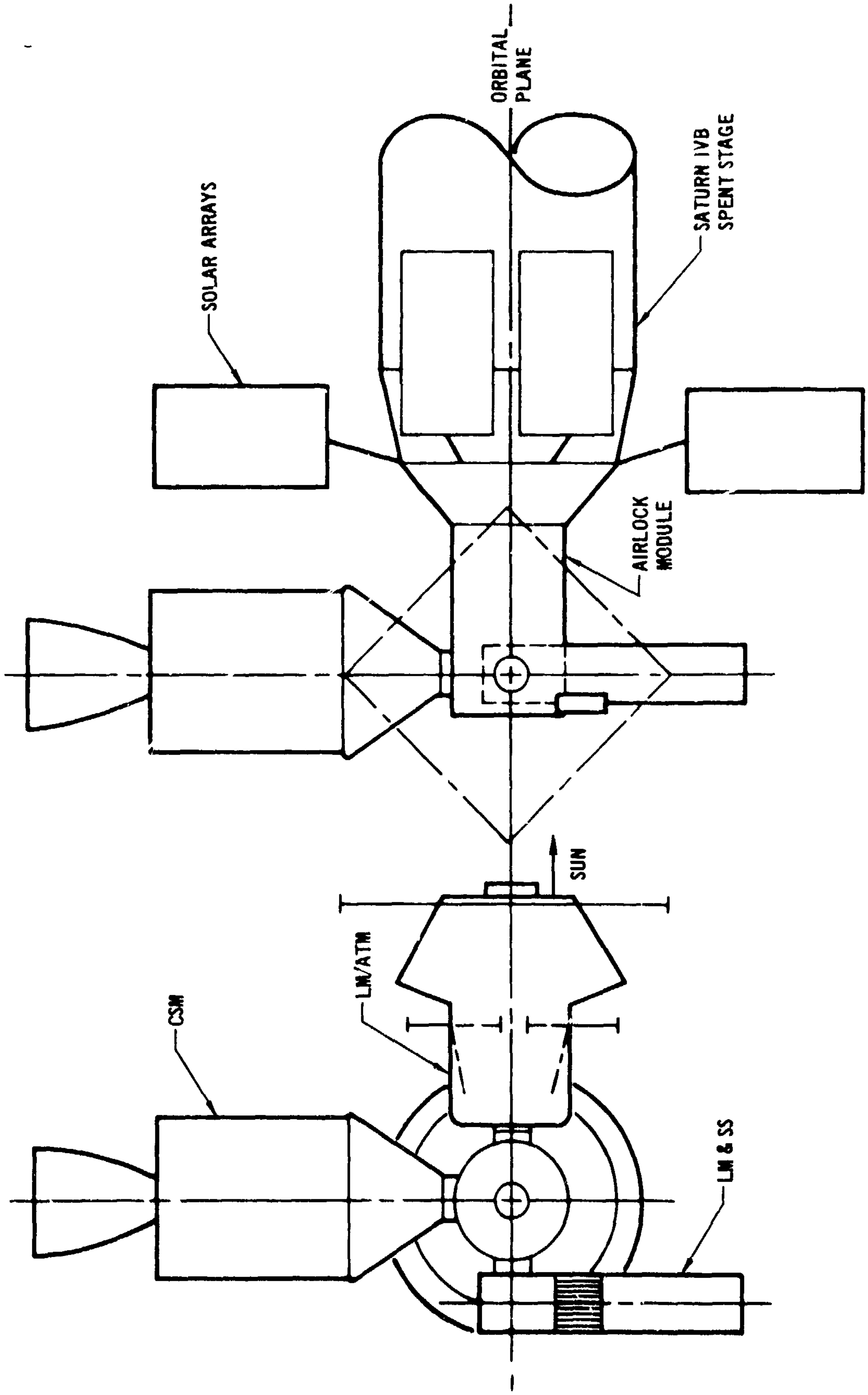


FIGURE 2.1-3 ORBITAL WET WORKSHOP CONFIGURATION (UNMANNED LAUNCH)

2.1.4 Wet Workshop Configuration

By Mid-1967 a firm workshop configuration had evolved and major changes had been made to the Airlock Module and its systems.

The forward end of the pressurized cylinder, including the docking ports, was removed as part of the AM and a new module, the Multiple Docking Adapter (MDA) created. The MDA was to be Government Furnished, but the radiators covering the exterior of the MDA remained part of the AM task. The solar arrays were removed from the AM and added onto the OWS. Both the cryogenic O_2/H_2 and gaseous O_2/N_2 supplies were removed from the AM; gases were to be supplied from the CSM through an umbilical. Battery modules were added onto the AM trusses and a scientific airlock was added to the AM. This configuration as shown in Figures 2.1-4 and 2.1-5 was the Apollo Application Program (AAP) "wet" workshop configuration.

The AAP "wet" workshop mission profile also underwent considerable change. In Mid-1967 the mission consisted of two CSM launches, an unmanned orbital workshop launch and an unmanned LM/ATM launch. All launches were on Saturn I-B vehicles with total mission duration of up to 9 months, as shown in Figure 2.1-6. A possible CSM revisit was considered within 6 to 12 months after AAP-3 splashdown.

By Mid-1968 the AAP mission had evolved into a 28-day mission and two 56-day missions with 90-day orbital storage periods in between; all five launches were by Saturn I-B:

- Three manned CSM launches.
- One unmanned workshop launch.
- One unmanned LM/ATM launch.

2.1.5 Dry Workshop Configuration

On 28 August 1969 the wet workshop configuration was superseded by a dry workshop configuration -- the Skylab. The basic change was to launch the workshop, including all experiments and expendables, in a single unmanned Saturn V launch.

- The S-IVB stage was to be launched dry after having been configured on the ground for manned laboratory use.
- Separate launch of LM/ATM was eliminated, and the ATM was included in the unmanned workshop launch payload.

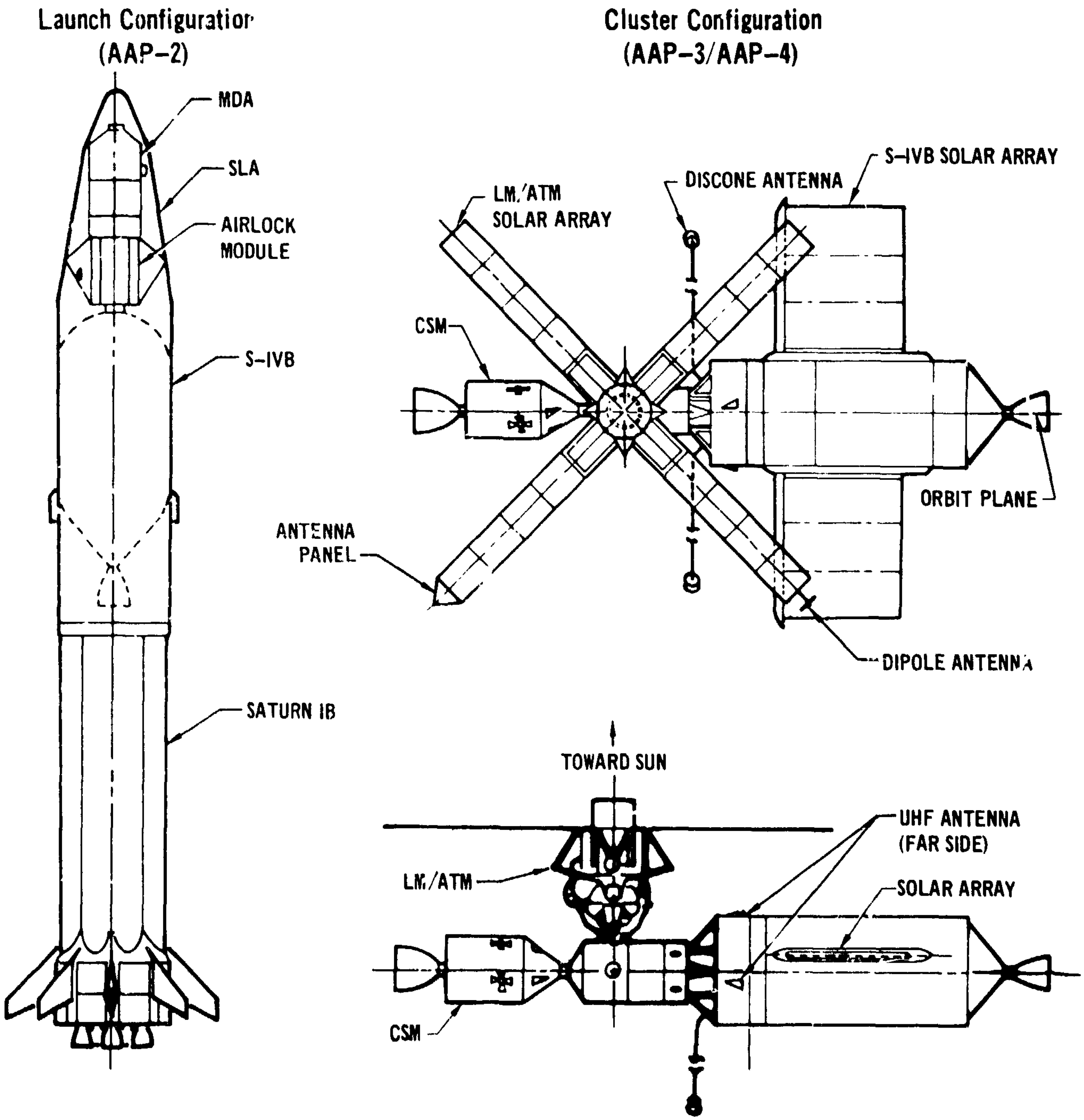


FIGURE 2.1-4 APOLLO APPLICATIONS PROGRAM – WET WORKSHOP CONFIGURATION

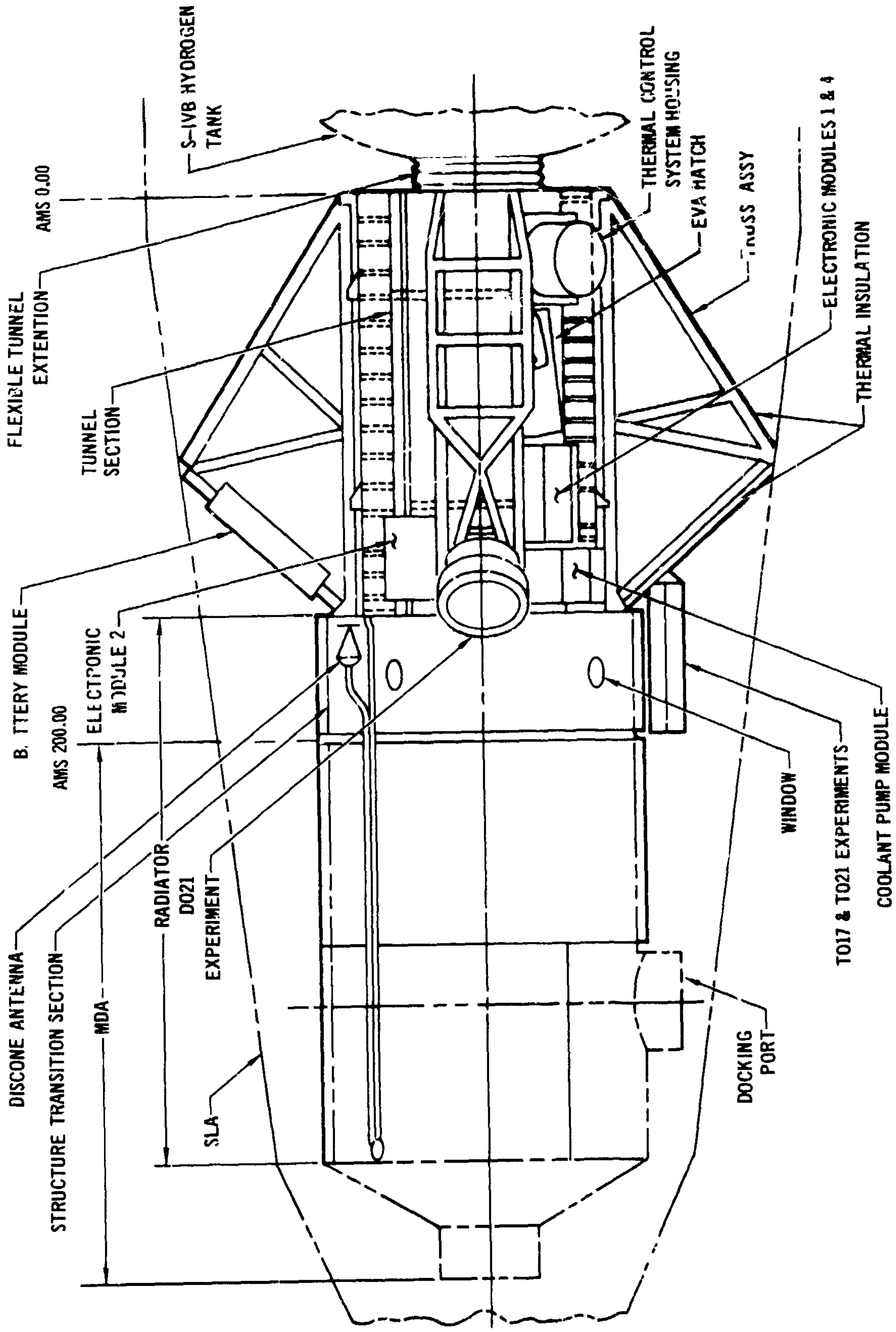


FIGURE 2.1-5 AIRLOCK MODULE ARRANGEMENT (AAP-2)

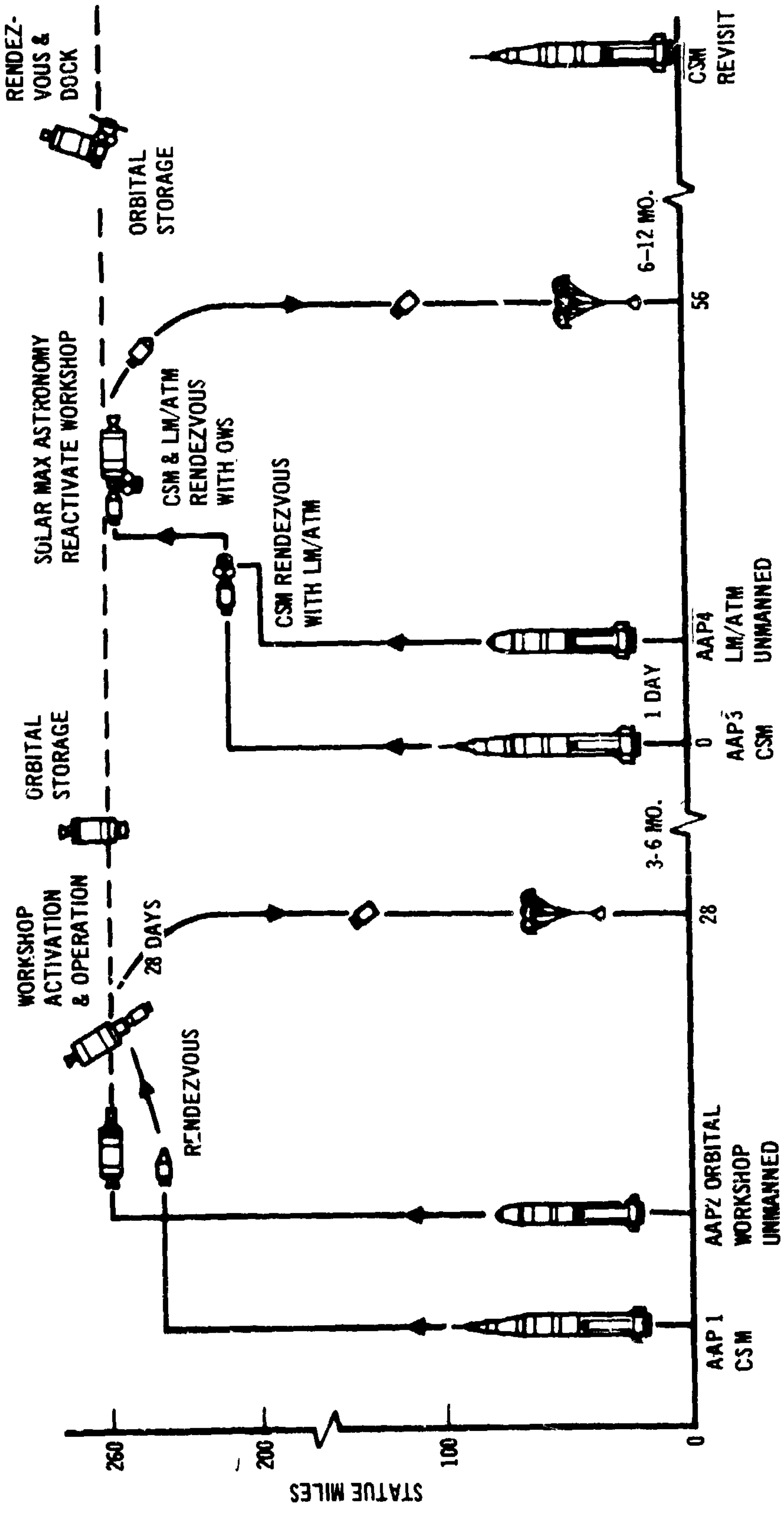


FIGURE 2.1–6 WORKSHOP MISSION PROFILE (AAP)

The planned mission profile of the Skylab, shown in Figure 1-6, included an unmanned workshop launch on a Saturn V vehicle and three successive CSM launches on Saturn I-B vehicles. Manned operation periods of 28, 56, and 56 days were planned with nominal unmanned storage periods of 57 and 37 days. Launch configurations of both the unmanned workshop and the manned CSM are shown in Figure 1-4.

The change to the dry workshop involved major changes to the Airlock Module.

- Addition of Deployment Assembly (DA), to deploy the ATM 90° from launch to orbital operating position.
- Addition of a new design, jettisonable Payload Shroud (PS) to support the ATM during launch and to provide aerodynamic and contamination protection until jettisoned in orbit -- the PS replaced the SLA.
- Addition of a Fixed Airlock Shroud (FAS) to provide launch support for the AM/MDA/PS/DA/ATM.
- Addition of tankage to supply gaseous O_2 and N_2 for the cluster atmospheric gas system.
- Addition of two-gas control system.
- Deletion of the scientific airlock (moved to OWS).
- Change in MDA docking port configuration (from five to two) and a matching change in AM radiator panels installed on the MDA.
- Addition of an active cooling system for the ATM control and display panel.
- Thermal blanket relocation and redesign.
- Revised AM electrical power system to provide for cluster power load sharing with the ATM electrical power system, and deletion of the CSM as a cluster electrical power source.
- Provision of a cluster caution and warning system.

The as-flown Airlock Module configuration is shown in Figure 1-3 with the other modules of the Skylab cluster configuration. Figure 2.1-7, the Airlock Module weight history from SSESM to SL-1 launch, indicates, on a weight basis, the magnitude of the Airlock system changes through its design phase -- from 7985 lbs to 75978 lbs all-up launch weight with the major change associated with the conversion to a dry launch workshop configuration.

Concurrent with the major mission and vehicle changes were many AM system requirement changes and hardware redesigns and modifications. Where pertinent to

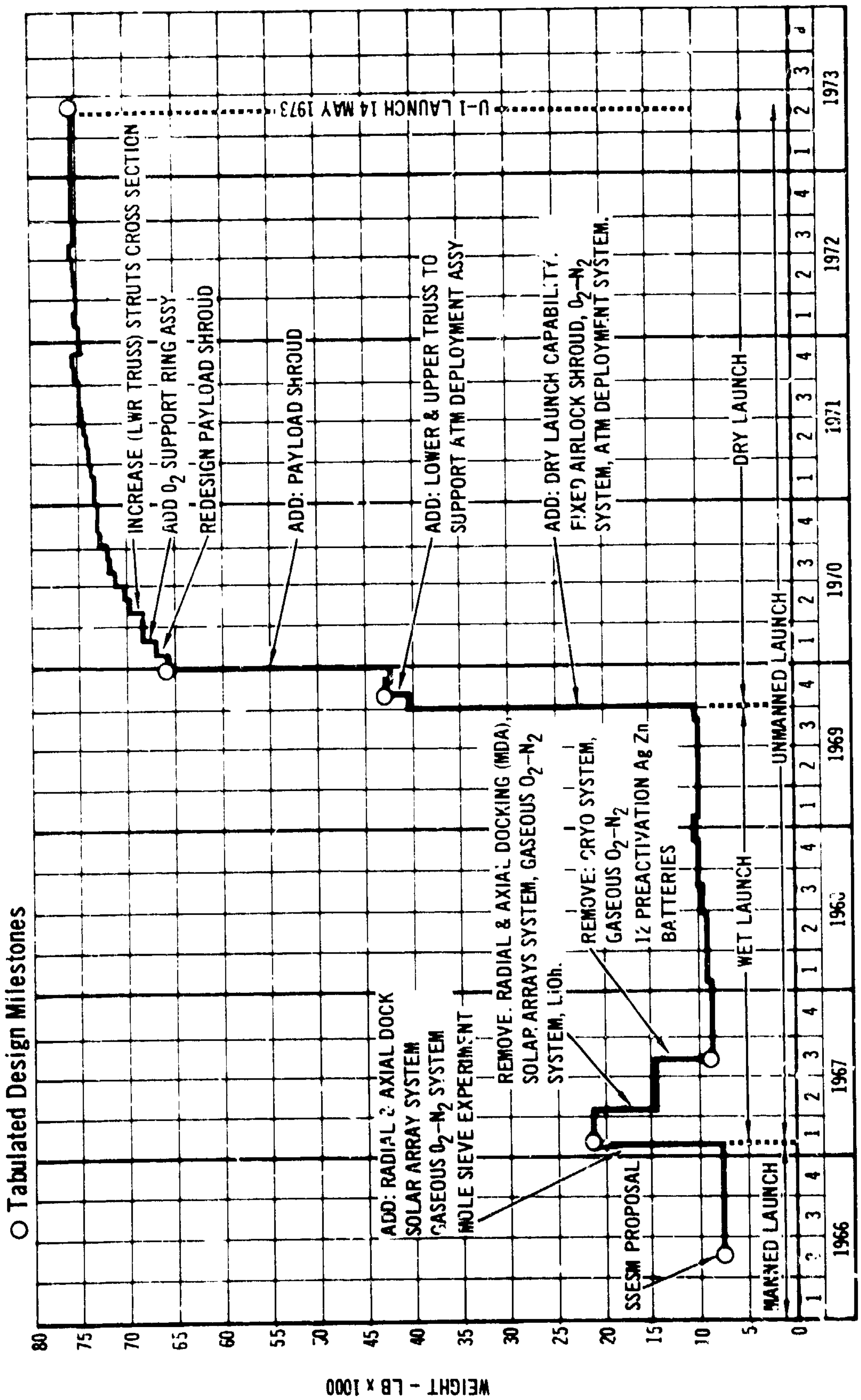


FIGURE 2.1-7 AIRLOCK WEIGHT GROWTH SUMMARY

understanding, system design evolution is discussed in the individual system report sections.

Although the Airlock Module evolved from the simple SSESM to a highly complex space vehicle over the life of the program, the primary design requirement of making maximum use of existing flight qualified hardware remained. Additionally, the verification process continued to stress extensive use of system engineering analysis and previous test results in identifying the supplemental tests necessary to assure confidence in achieving primary mission objectives and preserving crew safety.

2.2 STRUCTURES AND MECHANICAL SYSTEMS

2.2.1 Design Requirements

The structures and mechanical systems were the basic framework on which all Airlock systems depended. The requirements grew through the evolution period, resulting in four elements; the Airlock Module, the Deployment Assembly, Fixed Airlock Shroud and the Payload Shroud. (The Payload Shroud is discussed in MDC Report G4679A.)

2.2.1.1 Airlock Module (AM)

The AM was required to provide a pressurized vessel to house cluster controls, allow passage between the CSM and the OWS, to permit EVA, and to be a structural support to other cluster elements.

The AM was configured, as shown in Figure 2.2-1, with four major elements.

A. Structural Transition Section (STS) and Radiators - The STS was the structural transition from the 120-inch diameter MDA to the 65-inch diameter AM tunnel section. The STS contained four windows for external viewing, with movable window covers for thermal/meteoroid protection. Radiators were mounted around the periphery of the STS and portions of the MDA to provide thermal/meteoroid protection as well as perform their basic function as space radiators. The internal volume of the STS housed equipment and controls for the electrical, communication, instrumentation, thermal, environmental, and EVA/IVA systems.

B. Tunnel Assembly - The tunnel assembly was a pressure vessel providing a system of hatches that functioned as an Airlock to permit EVA. The size of the lock compartment with all hatches closed was required to accommodate two pressure suited astronauts with their EVA equipment. All hatch operations were to be designed such that they would be easily operated by a pressure suited astronaut. The internal volume of the tunnel assembly was sized to house and support equipment and controls for the electrical, communications, instrumentation, environmental and crew systems.

C. Flexible Tunnel Extension - The configuration of the Airlock Module and the OWS dictated the need for a pressure-tight passageway between these two modules that would accommodate relative deflections with minimum load transfer. A redundantly sealed, flexible tunnel was designed to provide this passageway.

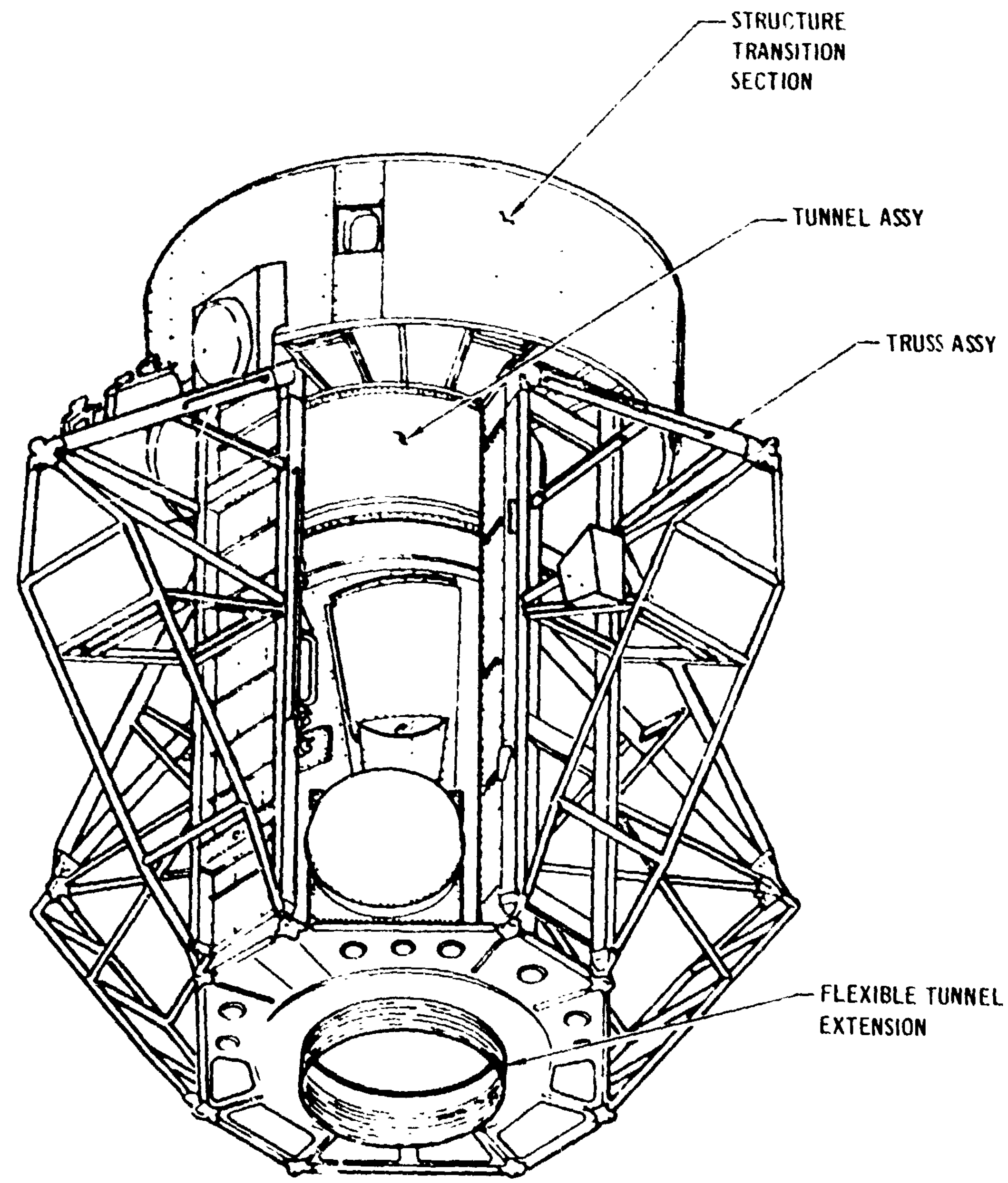


FIGURE 2.2-1 AIRLOCK MODULE

D. Support Truss Assembly - The AM and MDA were supported by four truss assemblies that attached to the tunnel assembly and mated with four attach points on the FAS. The trusses were also used to support N_2 tanks, battery modules, experiments and miscellaneous equipment.

2.2.1.2 Deployment Assembly (DA)

A deployment assembly was required for rotation of the ATM from a launch stowed position to the mission operating position. The ATM was supported during ground operations and launch by the PS. Upon PS separation the ATM was mechanically rigidized to the DA which was then rotated 90° into its in-orbit position, with a pointing accuracy of $\pm 1°$. Rotation of the ATM was to be accomplished in less than 10 minutes. The natural frequency of the deployed ATM/DA was to be greater than 0.6 Hertz.

2.2.1.3 Fixed Airlock Shroud (FAS)

A structural assembly was to interface with the IU, provide continuity of external surface configuration and provide attachments for the DA, AM, PS, and O_2 tanks. Concentrated loads generated at these attachments were to be distributed by the FAS to the IU interface. Access and ground umbilical doors were required in the FAS. The FAS was also used to support antennas and miscellaneous EVA equipment.

2.2.2 Systems Description

2.2.2.1 Airlock Module (AM)

The final configuration of the airlock module extended from the MDA interface at Station 200 to the four FAS attach points at Station 100 and the OWS dome at Station -11.45.

A. Structure Transition Section (STS) and Radiators - The STS structure, shown in Figure 2.2-2 provided the structural transition from the Multiple Docking Adapter (MDA) to the airlock tunnel. The enclosed volume of the STS was 288 cu. ft.

The STS structure was an aluminum welded pressurized cylinder, 47 inches long and 120 inches in diameter, of stressed skin, semi-monocoque construction. At the forward end a machined ring interfaced with the MDA. Stringers and longerons were resistance welded externally to the skin to carry bending and axial loads. Intermediate internal rings added support. Eight internal intercostals along with the truss attachment fittings transferred STS shell loads to the support trusses. The STS bulkhead provided the transition from 120-inch diameter to 65-inch diameter to mate with the tunnel assembly. Machined rings were utilized to make a typical flanged, bolted interface. The STS bulkhead along with the tunnel shear webs and the aft octagon ring provided shear continuity of the AM and redistributed loads to the AM support trusses. Sixteen radial sheet metal channels and eight machined titanium radial fittings, which included lugs for attaching the STS to the trusses, stiffened the STS bulkhead that interfaced with the AM tunnel. Four double pane glass viewing ports allowed visibility. Each window was protected when not in use by an external movable cover assembly, actuated from inside the STS by a manual crank. The cover served a dual purpose: to minimize meteoroid impacts on the glass, and to minimize heat loss from the cabin area.

The Airlock Module Radiator panels served as a meteoroid shield for part of the pressure vessel skin in addition to their basic function as space radiators. The radiators were mounted on the STS and MDA. To minimize

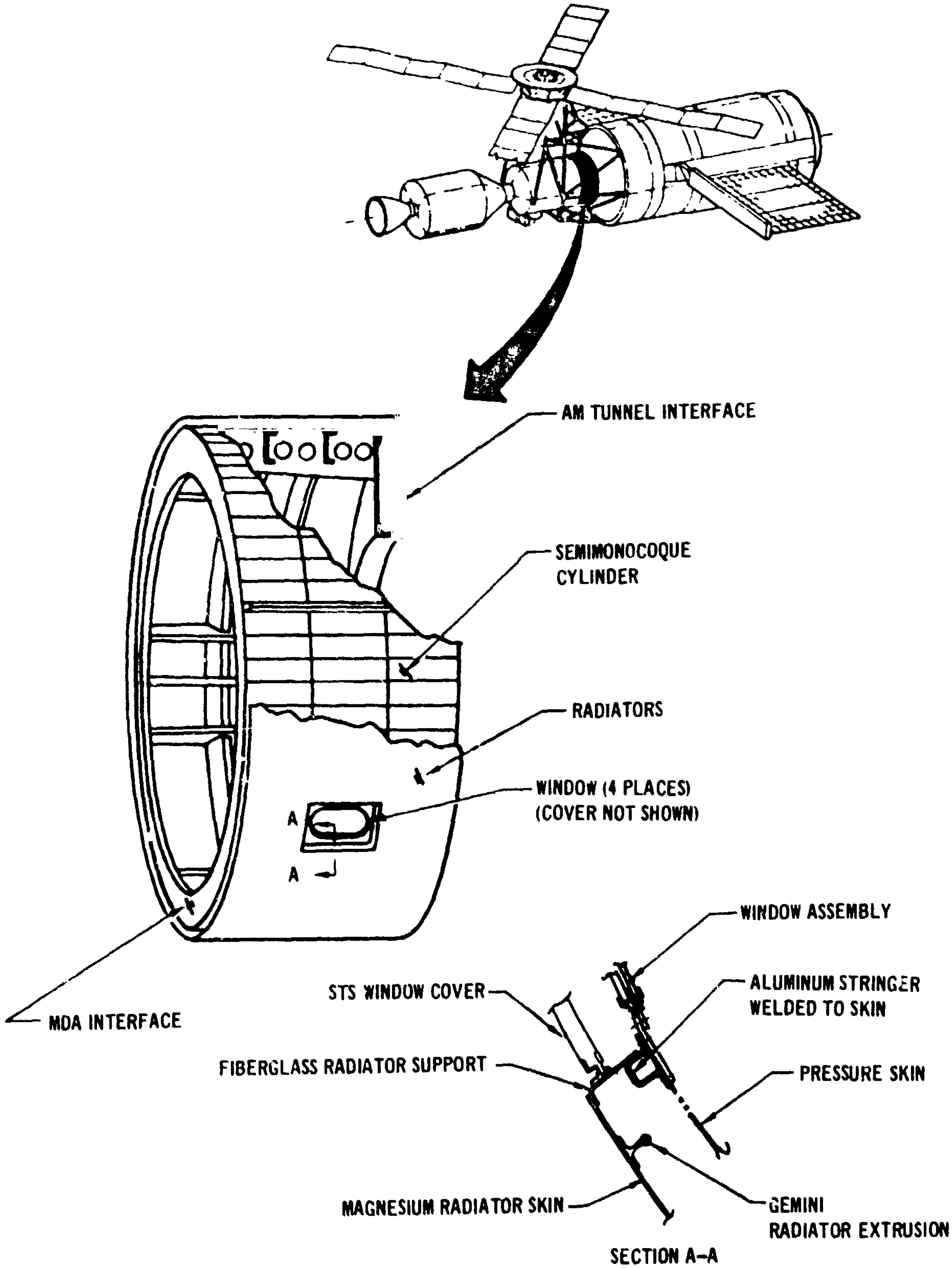


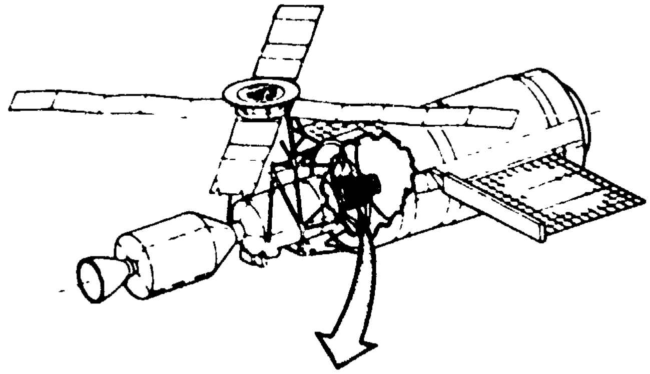
FIGURE 2.2–2 STS AND RADIATORS

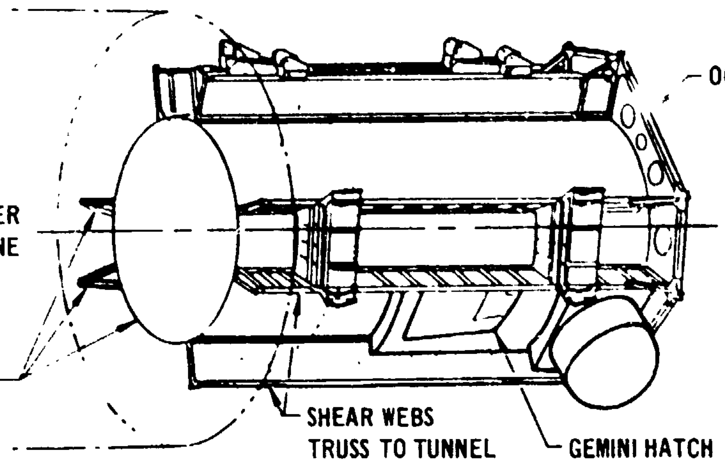
development and thermal testing, the panels were designed of the same materials and detail construction used on the Gemini Spacecraft radiator. Existing Gemini bulb-tee shaped magnesium alloy extrusions which provided a flow path for the coolant fluid were seam welded to a magnesium alloy skin. Each radiator panel was supported three inches outside the pressure vessel skin by fiberglass laminate angles. The fiberglass laminate angles minimized the heat conduction from the cabin area. Radiator locations for the STS are shown in Figure 2.2-2. Welded joints connecting most of the radiator coolant tubes minimized possibility of leakage. Mechanical connectors, utilizing Voi-Shan washers for seals, connected the radiator to the coolant loop and joined the radiator panel assemblies together.

B. Tunnel Assembly - The tunnel assembly was a pressurized semimonocoque aluminum cylinder 65 inches in diameter, 153 inches long and was configured as shown in Figure 2.2-3. External shear webs, an octagonal bulkhead and the STS bulkhead provided attachment and shear continuity between the tunnel assembly and the four truss assemblies. Two internal circular bulkheads with mating hatches divided the tunnel assembly into three compartments. Hatch seals and latching mechanisms were provided in these bulkheads.

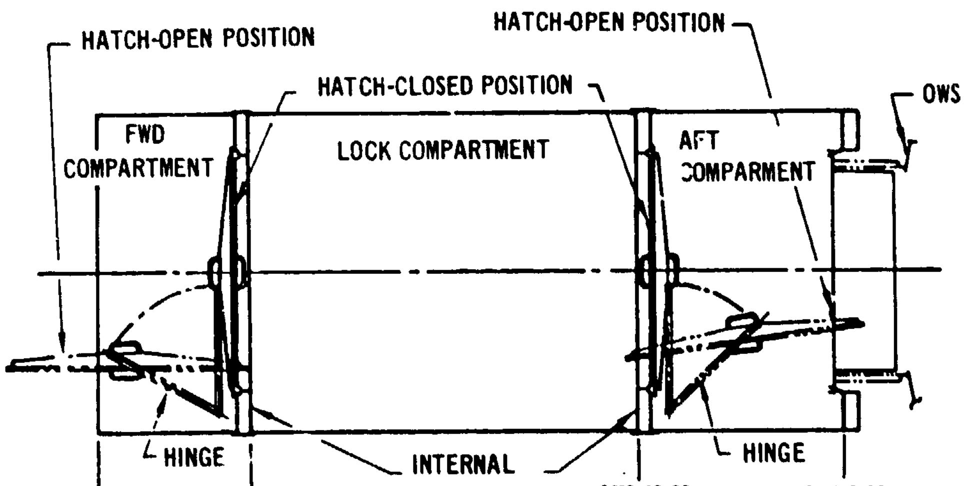
- The forward compartment was 31 inches long and interfaced with the STS section. It provided support for stowage containers, tape recorders, and miscellaneous equipment.
- The center (lock) compartment (volume 170 cu ft) was 80 inches long and included a modified Gemini crew hatch for ingress/egress during EVA.
- The aft compartment was 42 inches long and provided a housing to support the OWS environmental control system.

(1) Internal Hatches - The forward and aft internal hatches illustrated in Figures 2.2-3 and 2.2-4 were located at AMS 122 and AMS 42, respectively. Their original function was to seal off the lock compartment from the rest of the Skylab during EVA, however, the OWS hatch was used in conjunction with the AM forward hatch to perform this function during the mission. Both AM hatches were machinings 49.5 inches in diameter, with stiffeners attached radially. An





External Configuration
View Rotated 180° About Cluster Centerline to Show EVA Hatch



Internal Configuration

FIGURE 2.2-3 TUNNEL ASSEMBLY

8.5 inch diameter dual pane window in each hatch enabled viewing of the lock from both forward and aft compartments. Each hatch was hinged to fold along the tunnel wall and ensure correct closing orientation. A molded elastomer hatch seal was installed on each bulkhead.

Each latching system used a cable which was routed around the compartment bulkhead near the periphery of each hatch, driving nine (Gemini) hatch latch assemblies. Each hatch was latched when the handle was rotated through approximately 145 degrees, with a 25 lb. maximum load applied on the handle. A positive lock was included in the handle mechanism on the aft hatch.

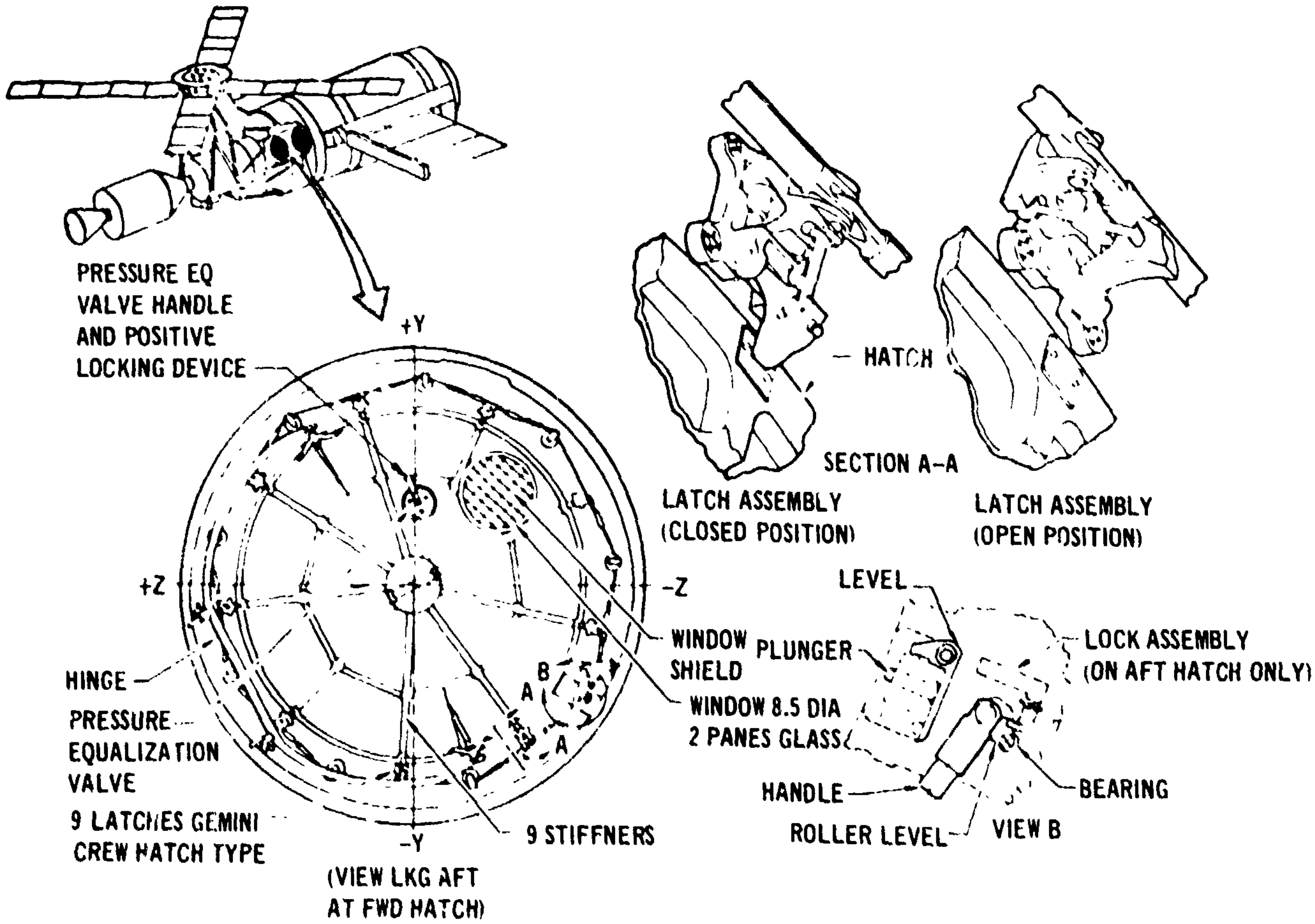


FIGURE 2.2-4 INTERNAL HATCH

(2) Extra Vehicular Activity (EVA) Hatch - The EVA hatch (Figure 2.2-5) was a modified Gemini design titanium structure shaped like a conical section, hinged to the AM torque box by means of four lugs. A molded elastomer hatch seal was installed on the sill assembly. A single stroke handle motion through approximately 153 degrees actuated the latching system consisting of a series of gear, links and twelve latches. This differed from the Gemini hatch in that the Gemini configuration used a multistroke ratchet-type handle motion. A double pane window in the hatch enabled viewing of the aft portion of the EVA quadrant. A tie-down harness was attached to the EVA hatch window frame to restrain a government-furnished removable machined aluminum protective window cover during EVA.

C. Flexible Tunnel Extension Assembly - A metallic convolute flexible bellows 42.5 inches inside diameter by 13.0 inches long formed the pressurized passageway between the AM and OWS, as shown in Figure 2.2-6. The attachment to the AM and OWS was made with 60 indexed .50 inch diameter holes and .25 inch diameter bolts, centered on a 43.863 inch diameter. The over size holes allowed for alignment tolerances. The mating flanges at the aft AM bulkhead and OWS forward dome interfaces were sealed by a molded elastomer material. All attaching hardware was selected to maintain clamp-up during periods of AM/OWS thermal expansion and contraction. A fluorocarbon coating applied to the internal surface of the bellows provided a redundant pressure seal.

D. Support Truss Assemblies - The basic truss assembly shown in Figure 2.2-7 is typical for all four truss assemblies. Minor modifications were required on each truss assembly to support miscellaneous equipment. The trusses were fusion welded aluminum tubes. Weight saving was accomplished by selective chemical milling. Machined fittings, fusion welded to the truss tubes, provided attachment to adjoining structure. The N_2 tanks were mounted on gimbals to isolate them from truss deflections and resulting loads.

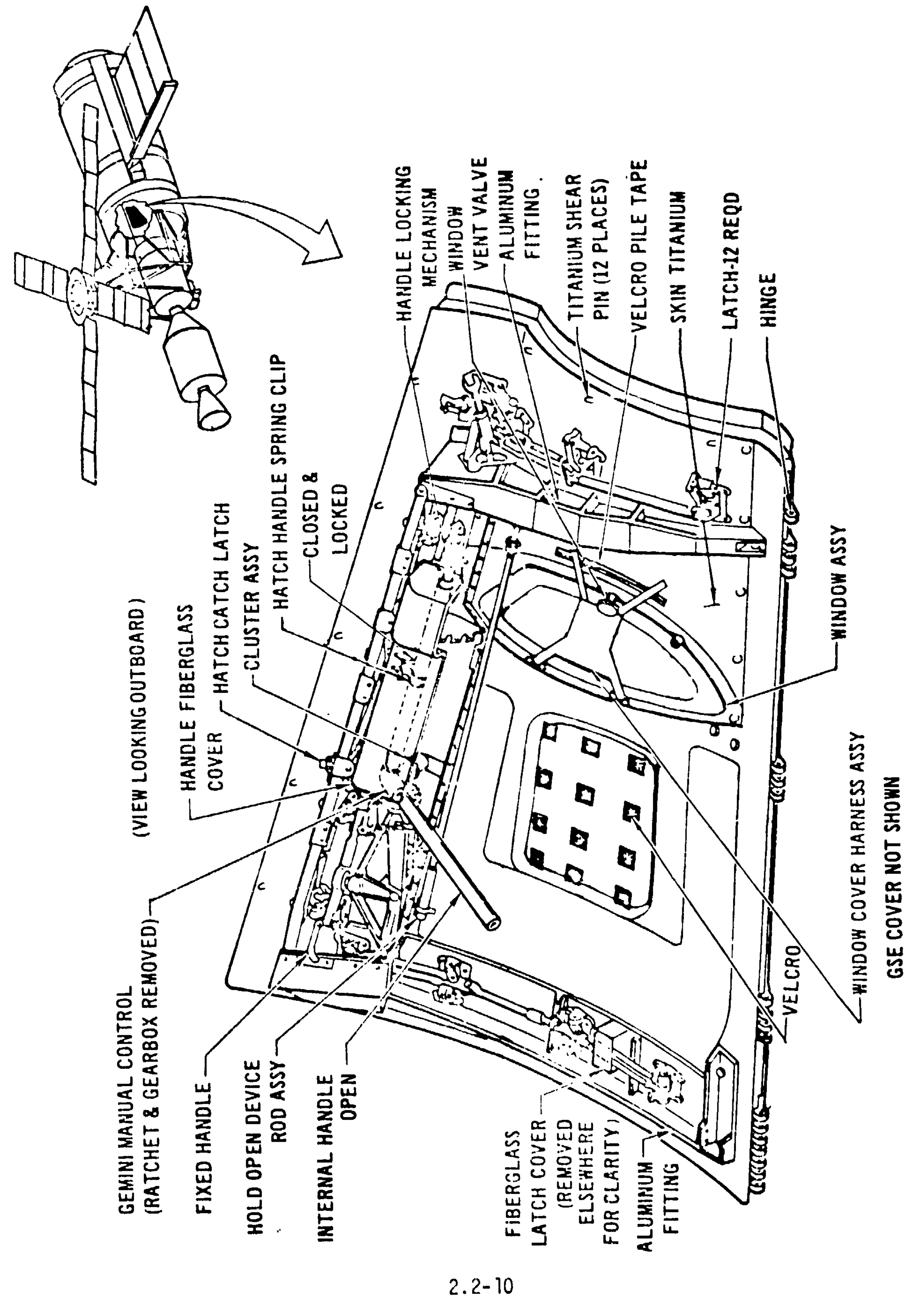


FIGURE 2.2-5 EVA HATCH

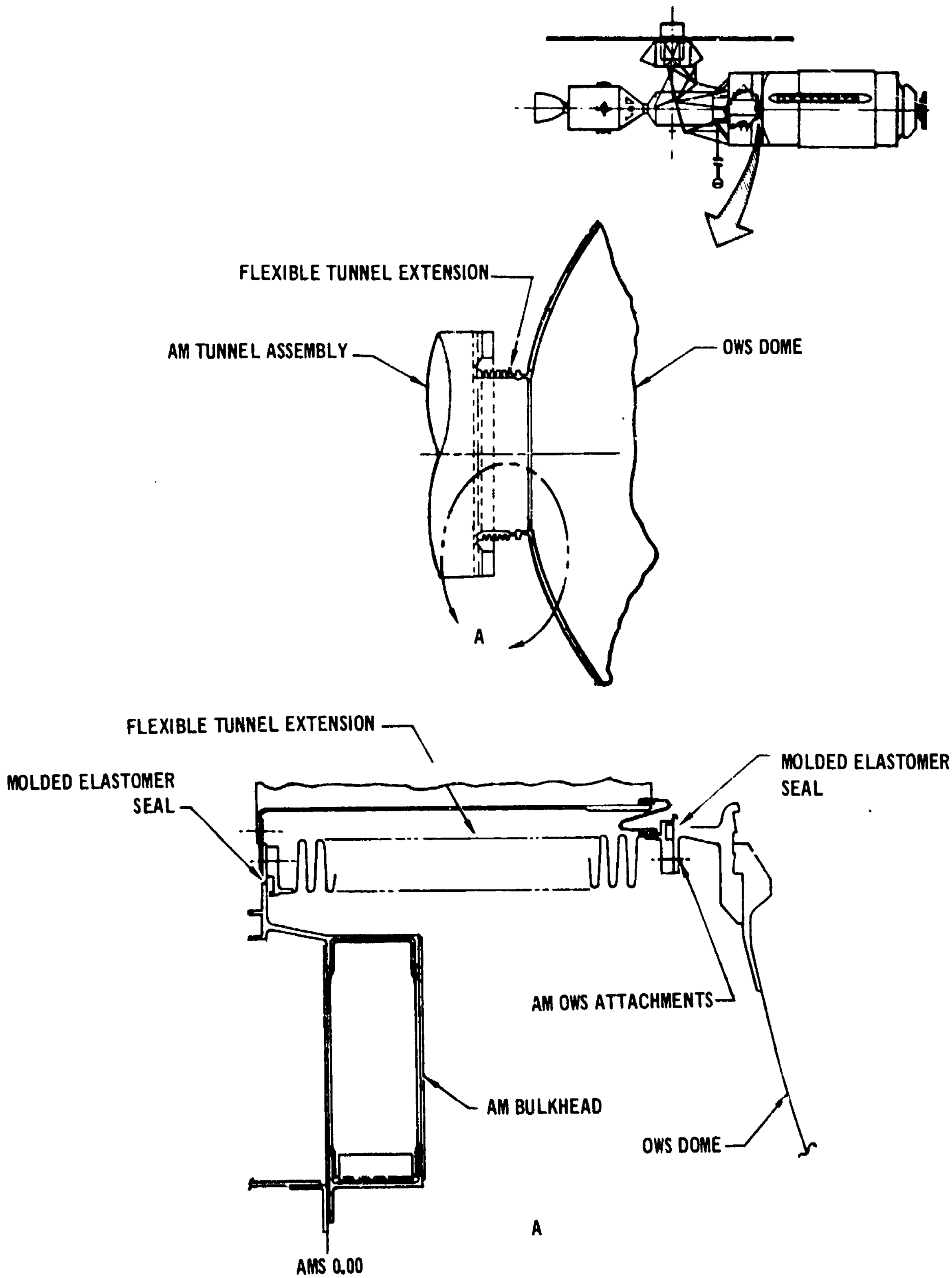


FIGURE 2.2-6 FLEXIBLE TUNNEL EXTENSION

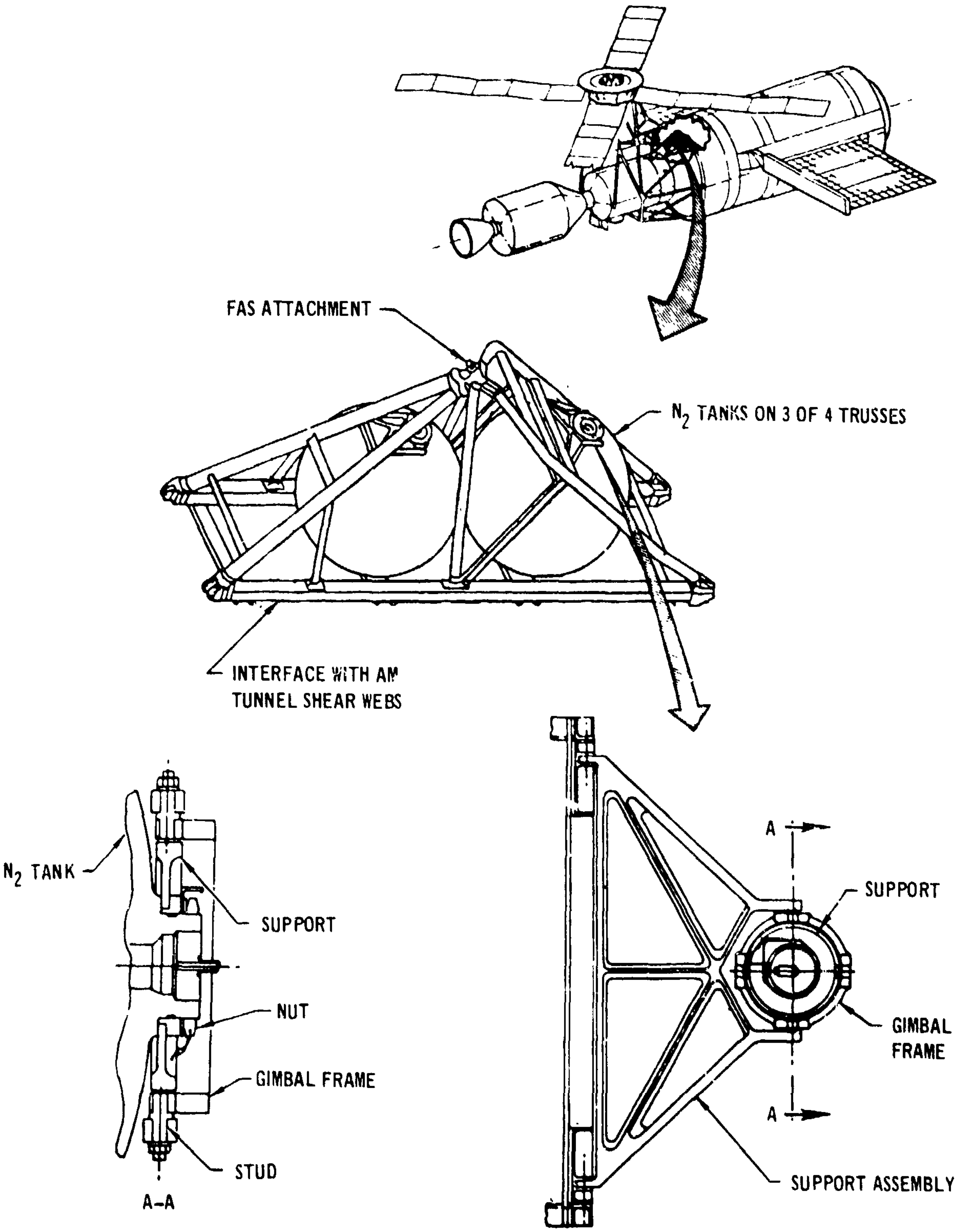


FIGURE 2.2-7 SUPPORT TRUSS ASSEMBLY

2.2.2.2 Deployment Assembly (DA)

The DA, shown in Figure 2.2-8, consisted of two aluminum tube truss assemblies connected by a pair of trunnion joints, which allowed the upper truss assembly to rotate 90° to deploy the ATM. The DA also supported wire bundles, experiments, antennas, and miscellaneous equipment. The lower truss assembly was made up of bipods, with the base of the bipods attached to the top ring of the FAS. A framework atop the upper truss assembly provided mounts for the four

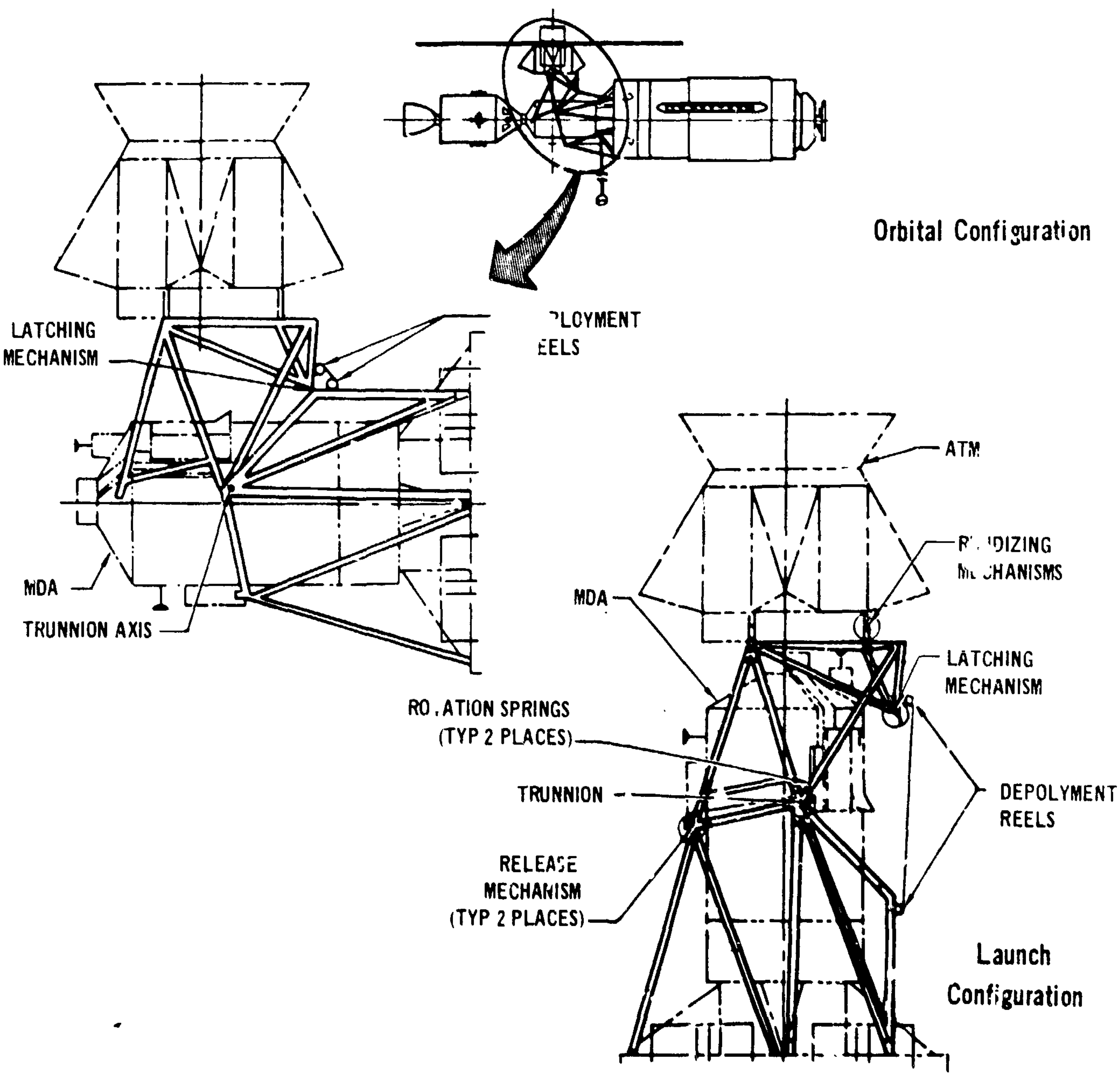


FIGURE 2.2-8 DEPLOYMENT ASSEMBLY

ATM attach points (rigidizing mechanisms). These rigidizing mechanisms, shown in Figure 2.2-9, attached to the ATM through four adapter fittings. During ground operations and launch, the ATM was supported by the PS but loosely attached to the DA by the rigidizing mechanism in the floating position. Following PS separation, the springs in each rigidizing mechanism retracted and rigidly attached the ATM to the DA. On the ground, alignment of the ATM was provided by the DA attachments at the rigidizing mechanisms. The DA rotation system provided a means of rotating the ATM from its launch position to its in-orbit configuration, as shown in Figure 2.2-10. The rotation system consisted of the following major components.

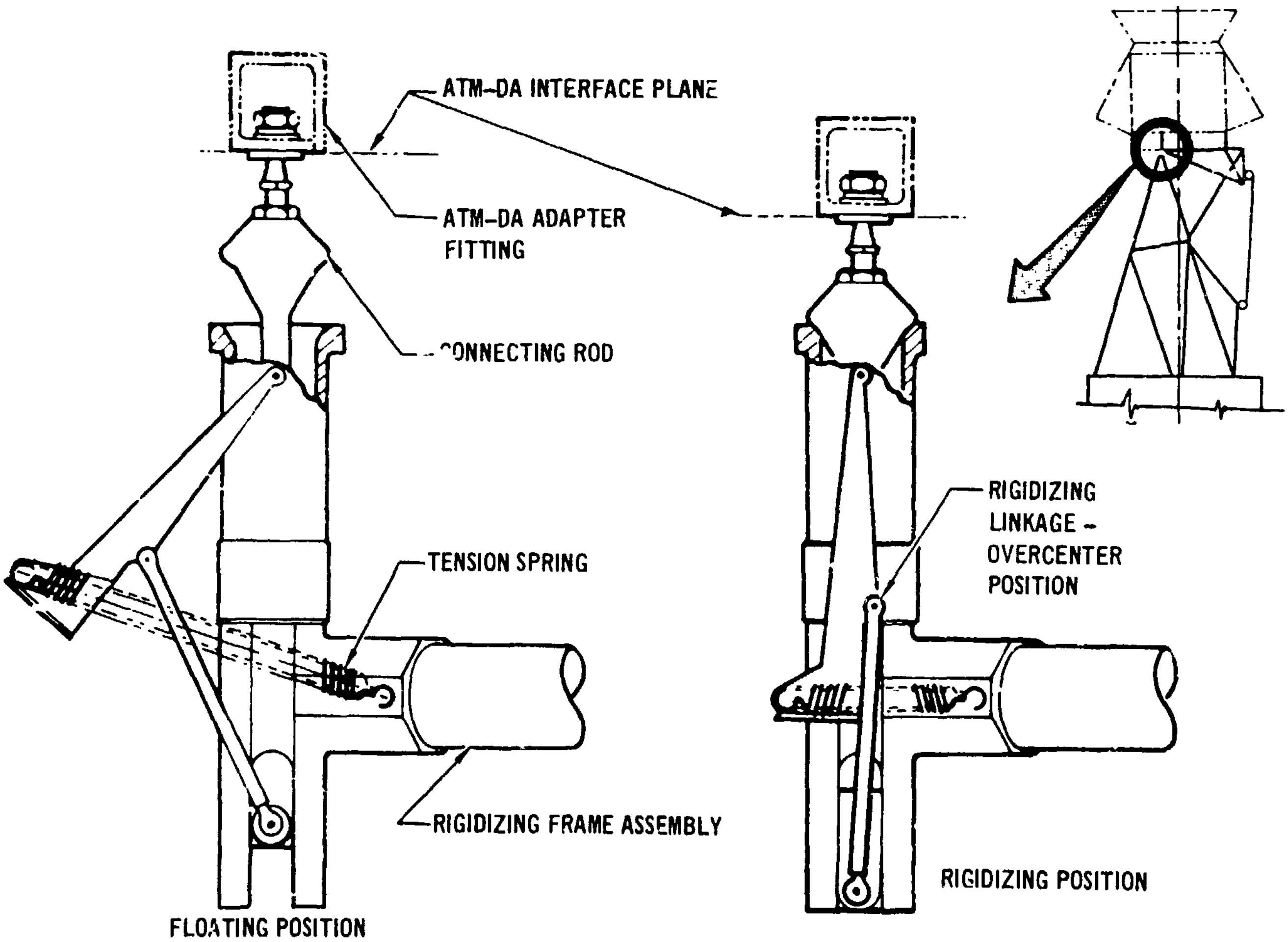


FIGURE 2.2-9 ATM RIGIDIZING MECHANISM

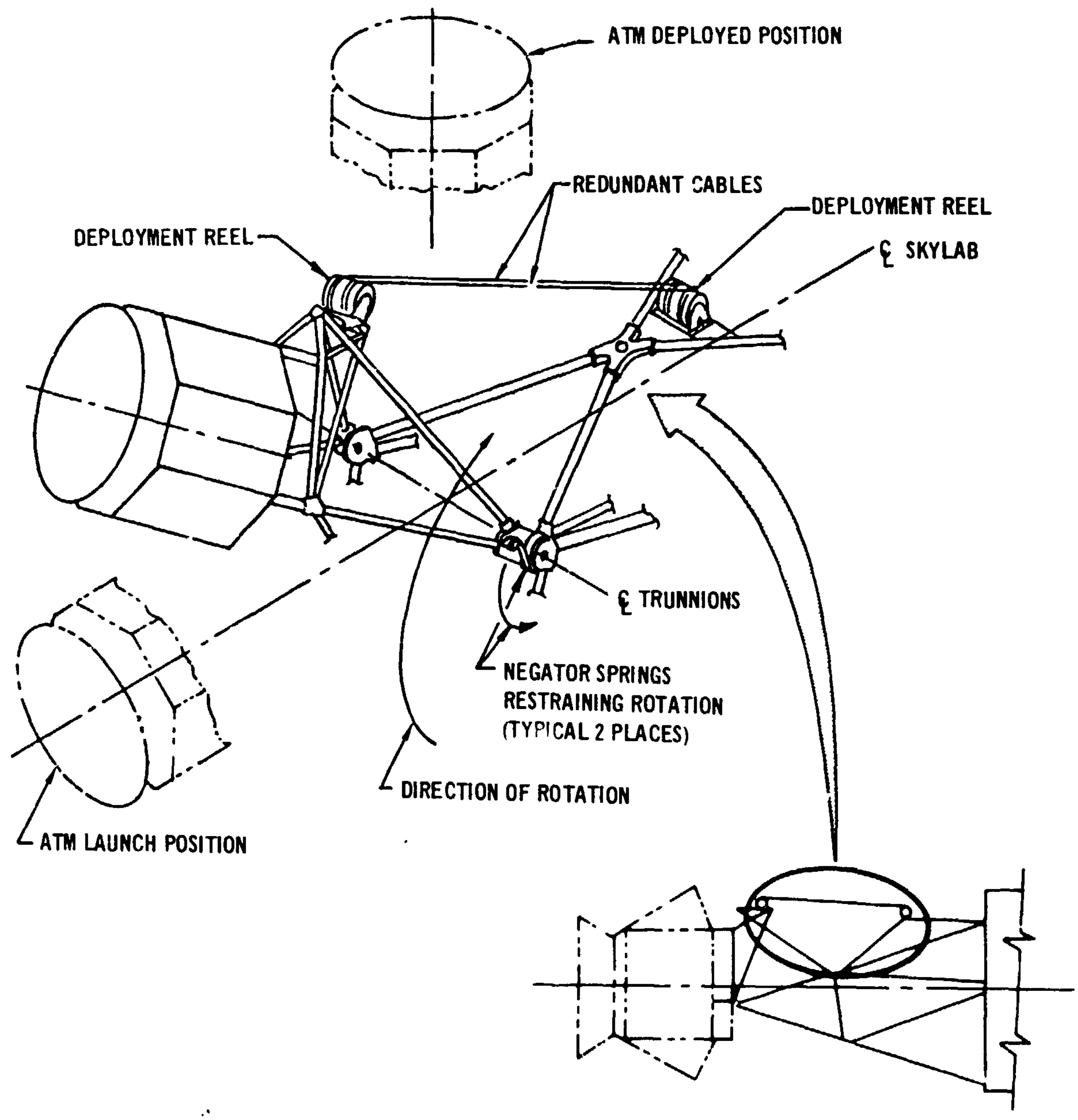


FIGURE 2.2-10 DEPLOYMENT ASSEMBLY ROTATION MECHANISM

A. Two release mechanisms each redundantly released the upper truss to allow rotation as shown in Figure 2.2-11. Release was accomplished by pyrotechnic pin retractors that were initiated by redundantly interconnected Confined Detonating Fuses (CDF) and manifolds as shown in Figure 2.2-12.

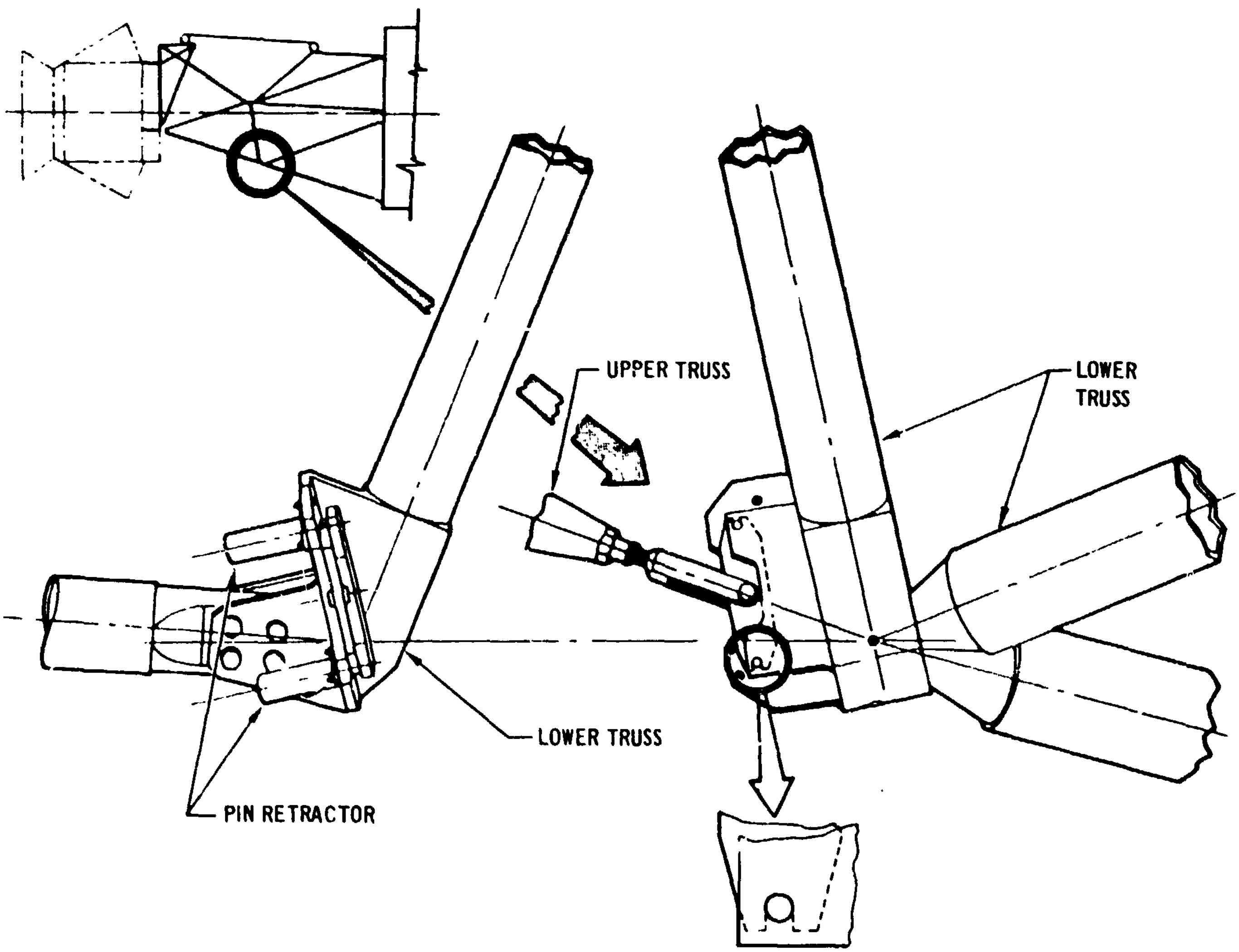


FIGURE 2.2-11 DEPLOYMENT SYSTEM RELEASE MECHANISM

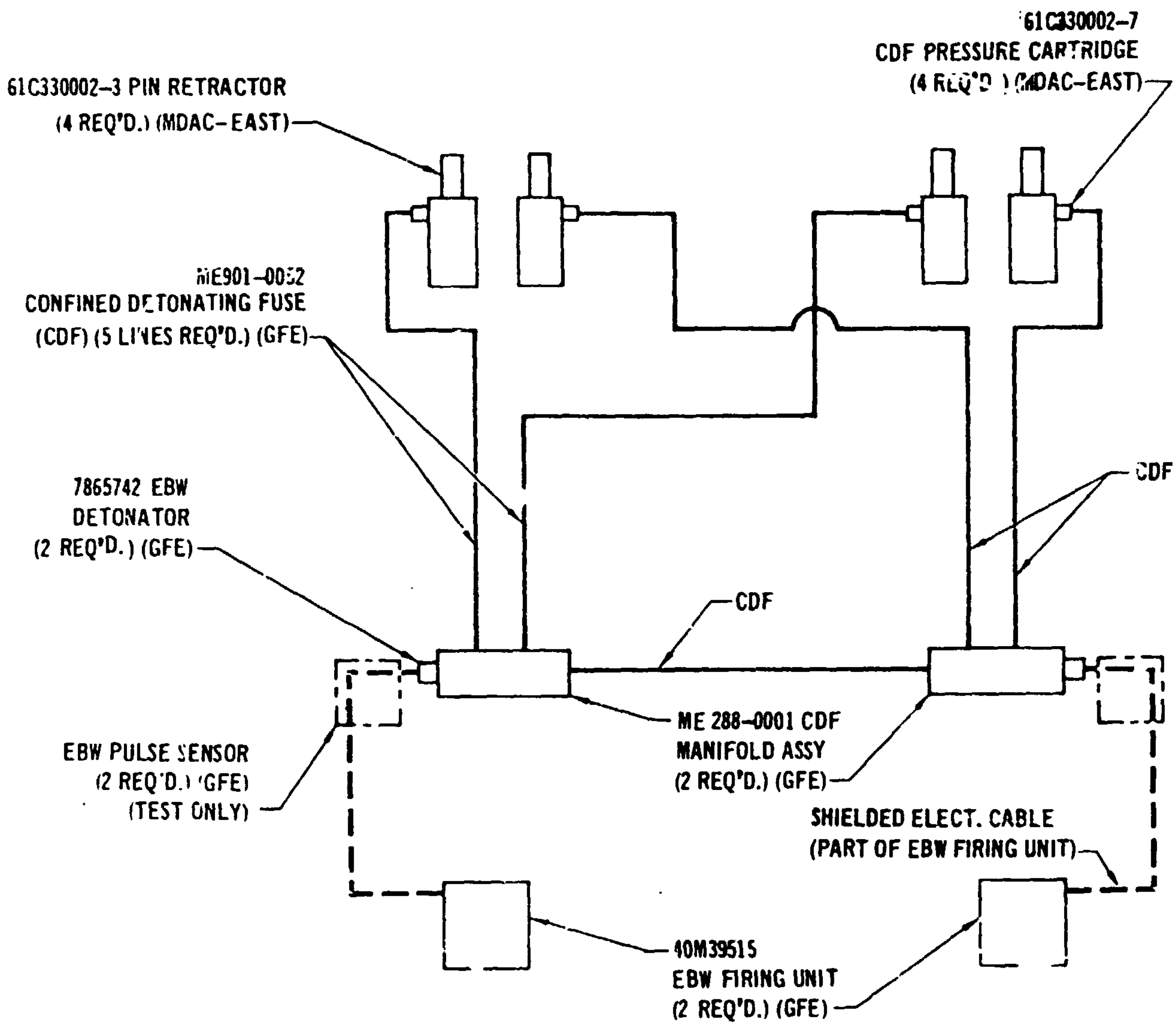


FIGURE 2.2-12 DEPLOYMENT SYSTEM PYRO SYSTEM SCHEMATIC

B. Two trunnions provided the pivots to rotate the upper truss. Each trunnion, as shown in Figure 2.2-13, contained a spherical monoball bearing and a negator spring that retarded rotation to maintain control of ATM during deployment. Single point failure of bearings was eliminated by making the outer race of the bearing a light press fit. This would allow rotation between outer bearing race and fitting should the bearing fail.

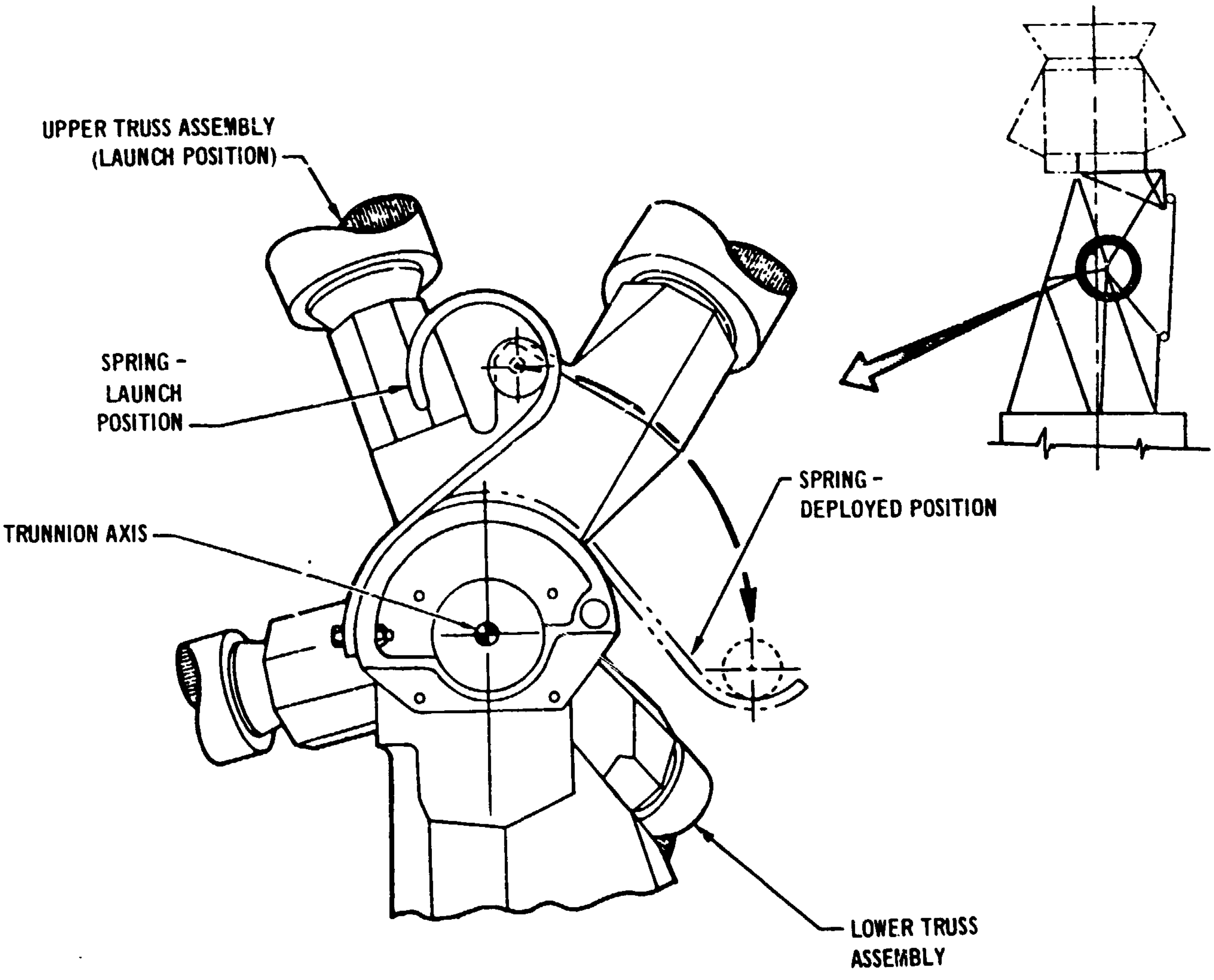


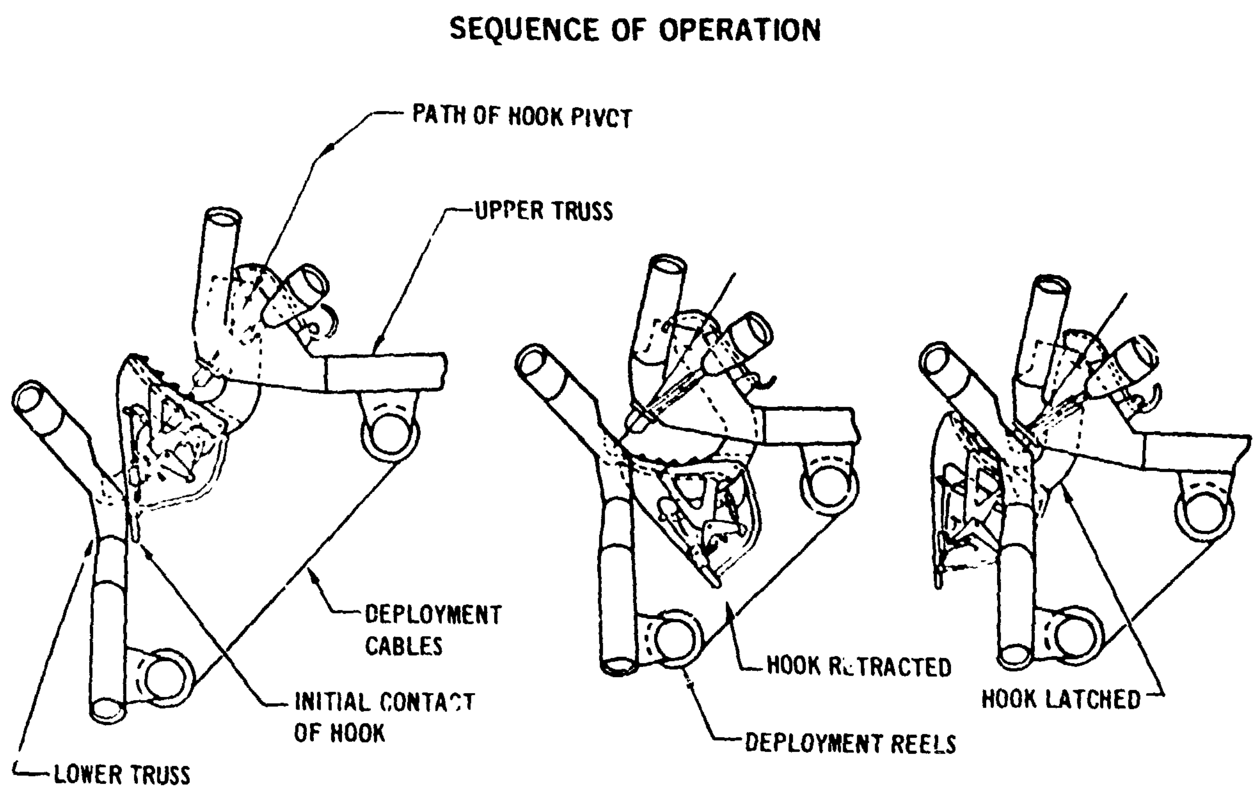
FIGURE 2.2-13 DEPLOYMENT SYSTEM TRUNNION MECHANISM

C. Two deployment reels provided the redundant means to pull (rotate) the ATM into the deployed position. The reels were mechanically designed to redundantly lock against cable paying out and were sized to be capable of reeling in all the cable required for total deployment with one reel inoperative. Each reel was capable of total deployment regardless of the point of failure of the other reel. When the ATM DA reached the deployed position, switches on the latch mechanism were cycled, initiating a time delay relay which cut off power to the reels after they had achieved their full stall load and DA deploy latching was complete.

D. The latch mechanism, shown in Figure 2.2-14, was used to retain the ATM/DA in the deployed position. Cam action retracted the spring loaded latch as the ATM/DA approached the deployed position. At the deployed position the spring force latched the hook, eliminating all assembly movement due to thruster attitude control system firings. A ratchet mechanism made the latch irreversible, locking the ATM/DA in the deployed position. Upon latching, redundant switches were cycled initiating turn off of the deployment reels. This triggered the TM signal that deployment was completed and removed the inhibit from the ATM SAS deployment system.

2.2.2.3 Fixed Airlock Shroud

The FAS was a ring-stiffened thick-skinned cylinder approximately 80 inches in height, 260 inches in diameter and configured as shown in Figure 2.2-15. Intercostals distributed concentrated loads introduced by the DA, AM and O_2 tank support points. Two doors were provided in the FAS; one for access to the FAS interior and the AM EVA hatch during ground operations and the other for access to ground umbilical connectors. Four antennas; two deployable discones, and two UHF antennas were mounted on the FAS. The FAS structure also contained EVA support equipment as follows: egress handrails, work platform, film cassette tree supports, film transfer boom also called TEE, a TEE hook stowage box and lights.



(Latch Spring Cover Omitted for Clarity)

FIGURE 2.2-14 DEPLOYMENT SYSTEM LATCHING MECHANISM

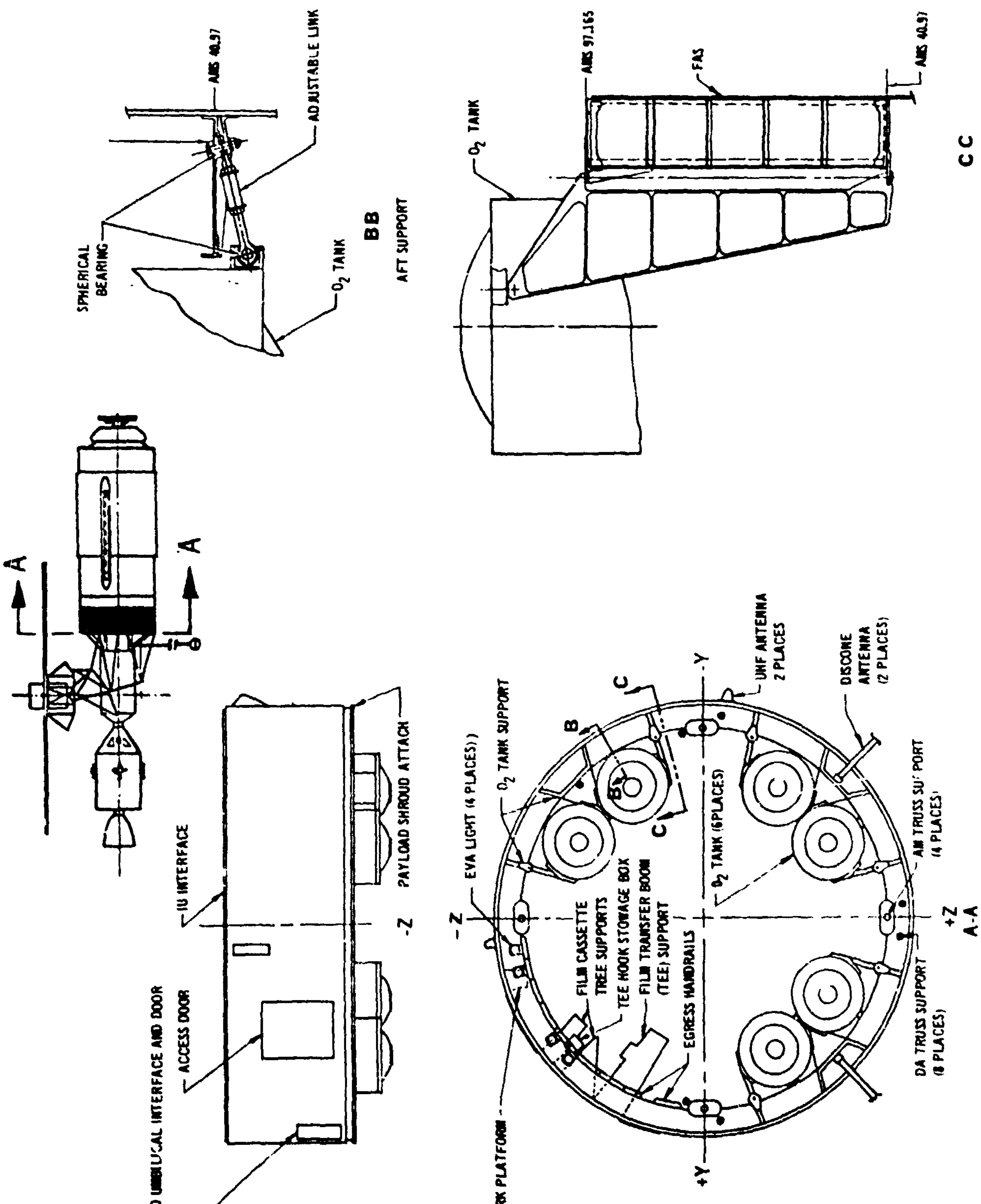


FIGURE 2.2-15 FIXED AIRLOCK SHROUD

2.2.3 System Verification

2.2.3.1 Qualification Testing

A. Structural integrity of the Airlock Tunnel and STS was demonstrated with Static Test Article No. 1 vehicle mated to the static test MDA. This structure was subjected to 12.4 psid and to ultimate loads simulating "wet" workshop launch and ascent loads. The launch and ascent loads for the "dry" workshop configuration were later verified by analysis as reported in "Verification of U-1 Launch and Ascent Structural Capabilities Based on Evaluation of STA-1 Static Test Results," MAC Report E0517. Structural capability for subsequent weight increases was verified by analysis and reported in "Effect of AM/MDA Mass Properties Change on AM Structural Capabilities," MAC Report E0654.

B. The EVA hatch and internal hatch seals were initially fabricated to the same ultra low durometer silicone rubber compound requirements as those successfully used on the Gemini flights. Although these seals did not leak it became apparent during AM checkout and seal specimen testing that the Gemini seal compound was unacceptable for the Skylab long-term space environment. This was evidenced by specimen testing which reflected a low and inconsistent state of cure resulting in excessive outgassing, inconsistent hardness, undesirable surface adhesion, poor bond integrity and unacceptable permanent set. Therefore, a new seal compound was developed which was basically the same as the Gemini seal except fabricated with up-to-date rubber industry technology, methods and techniques. A consistent low durometer silicone rubber compound was developed that would fully cure with low outgassing properties and was resistant to compression set and reversion. A cleaning procedure and the application of a surface release agent was developed for maximum reduction of surface adhesion. Verification testing was conducted by subjecting a flight article seal segment to temperature altitude testing. The results verified that the above objectives had been met.

C. The internal hatch and its mechanism system was qualified in a test fixture which simulated a 3-foot section of the AM tunnel structure including a production type forward hatch bulkhead and sill assembly. All functional, pressure, leakage, handle loads, life cycles and environmental requirements were successfully demonstrated.

D. To establish confidence in the design concept for the flexible tunnel extension bellows, a readily available bellows, similar to the eventual production configuration, was subjected to vibration, stiffness and fatigue testing. Results of these tests proved the concept acceptable.

E. Prior to availability of the first production DA, a steel truss with spring rates equivalent to the flight article was used for development testing of the DA design concept. The steel truss, with production fittings and components at the critical areas, was supported horizontally with the rotation axis normal to the ground. A system of pulleys, counterbalances and pivots were employed to simulate the effect of zero gravity and ATM inertia upon deployment. Overall system performance was evaluated with induced failure modes. Cycle life of the deployment reels and the effect of flexing of the wire bundles at the trunnions were prime objectives. The only significant change that resulted was the addition of a spring mechanism to the deployment reels to insure control of the deployment reel cables during deployment. All objectives of the development testing were successfully met.

F. Qualification testing of DA components was conducted at the component level as recorded in the Airlock Equipment Acceptability Review-Structural and Mechanical Systems, MDC Report G499, Volume 4. In addition, the deployment assembly was subjected to a total system qualification test. The first production DA (STA-3) was mounted horizontally to a simulated FAS with the trunnion axis vertical. Counterbalances were attached at optimum points to simulate zero-g and the ATM mass was simulated. All mechanisms and systems were successfully operated during the qualification cycle.

G. The FAS and DA were designed to a factor of safety of 3.0 for manned loading conditions and 2.0 for unmanned. Strength analysis, utilizing finite element computer programming when justified, was performed to show structural capability. Based on these large factors of safety and the detailed strength analysis, static testing was not required to verify the structural adequacy. All major structural components of the AM including the FAS and DA were subjected to vibro-acoustic testing and successfully passed these environments.

2.2.3.2 Systems Testing

The sequence of system testing is shown in Figure 2.2-16. Stacking and alignment operations are shown in Figure 2.2-17.

A. Mechanical Systems Component Testing - Vendor procurred mechanical devices were acceptance tested at the vendors for performance to critical design requirements. After receiving the devices at MDAC-E they were subjected to a preinstallation acceptance test to insure the units met the critical design requirements at time of installation.

B. Structure - The structural integrity of the AM/STS modules was verified by proof pressure testing the mated sections at 8.7 psig. Immediately following proof pressure testing, and three times thereafter the vehicle was leak tested in various configurations. Only minor leaks were encountered and these were repaired as they were detected.

C. EVA/Internal Hatch Mechanism Verification - Concurrent with the Leak Test activity, rigging verifications of the EVA and Internal Hatch Latch Mechanisms were performed. The initial rigging of the hatches was accomplished using high durometer seals which were fabricated to Gemini seal compound requirements and later replaced with the newly developed lower durometer flight article seals. Subsequent verification of the rigging with the flight article seals did not disclose any significant deviations.

D. STS Window Cover Mechanism Verification - Each of the four STS View Port Windows has a thermal/meteoroid shield activated manually by the crew. Design criteria for break-away and free running torque was not met until the rack (rack and pinion gears) had been reworked to the minimum allowed thickness and tapered shims installed between the rack and the structure to compensate for warpage due to machining. After rework, all design requirements were met or exceeded.

E. Discone Antenna Boom Deployment - St. Louis testing of the Discone Antenna Booms was performed to verify proper function of that critical mechanism prior to delivery. A set of completed booms were fitted to the mated DA/FAS in the launch position per approved test procedures. The booms were rigged, all mechanical launch parameters were verified and a trial release was performed. No anomalies were encountered during

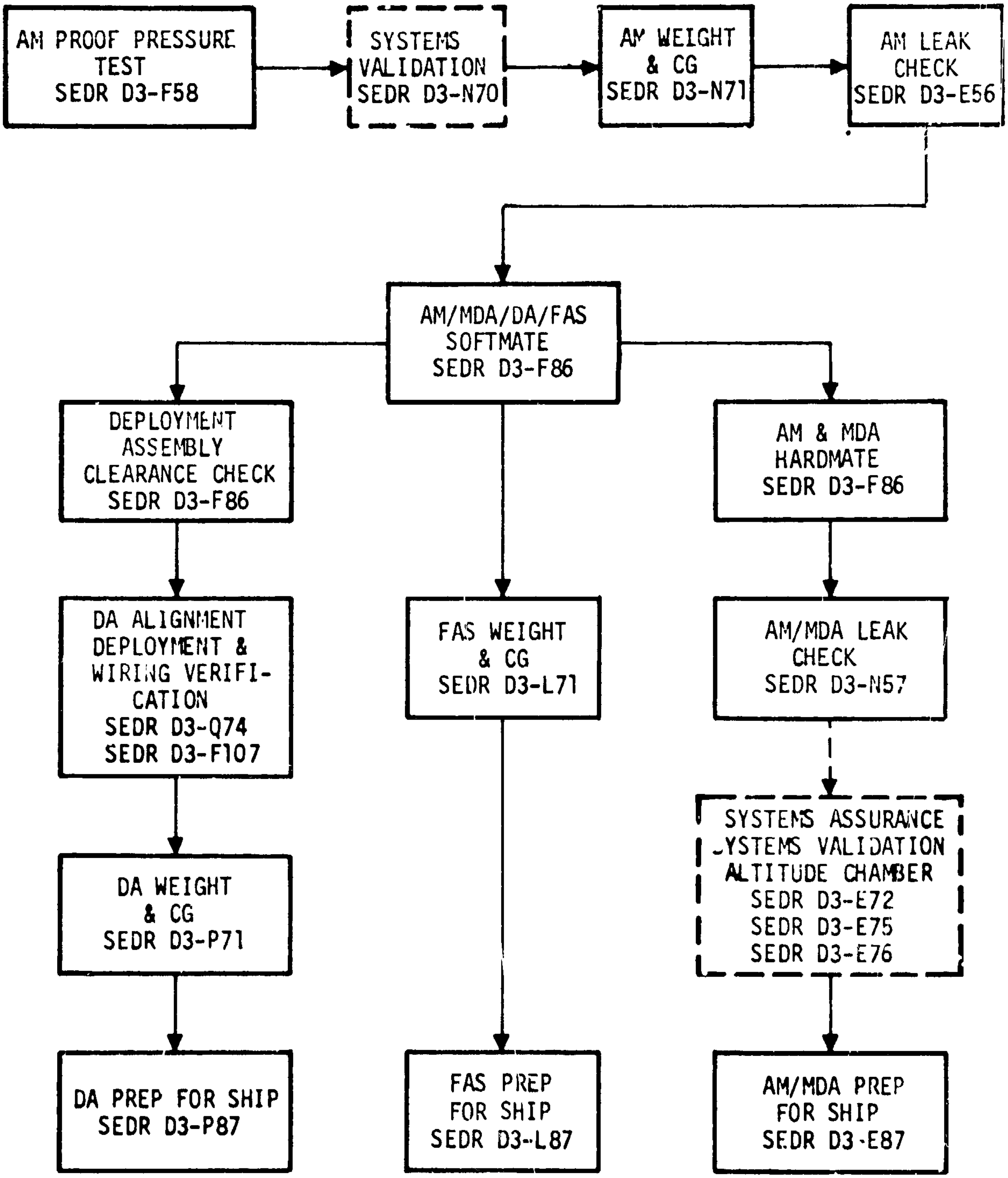


FIGURE 2.2-16 AM/MDA/DA MECHANICAL SYSTEMS TEST FLOW

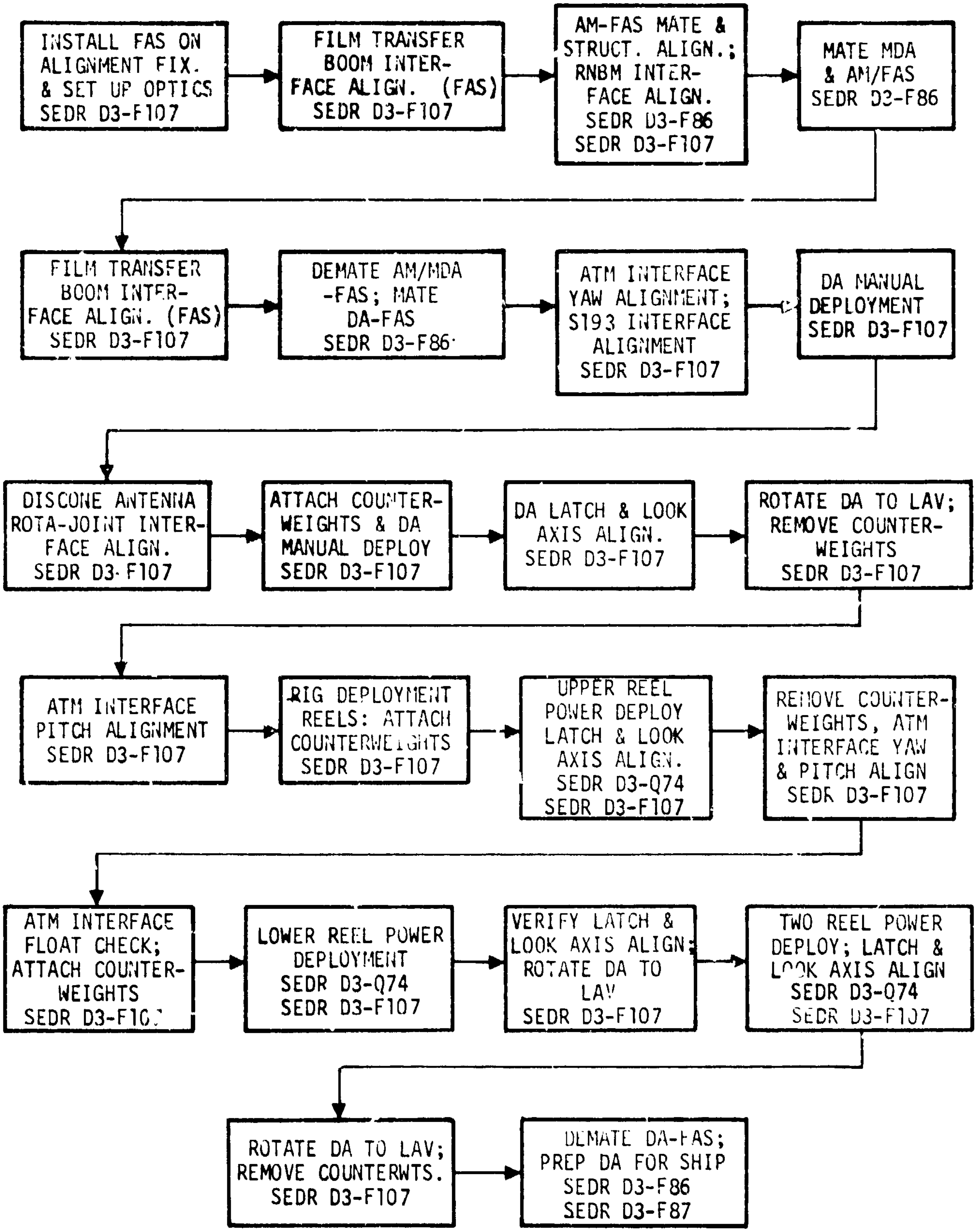


FIGURE 2.2-17 AM, AM/MDA, AND DA STACKING AND ALIGNMENT

the test. The booms originally designated for the U1 vehicle and fitted to the FAS/DA were subsequently found by X-ray tests to contain defective coaxial connectors and were replaced by backup units which had validated connectors. All booms were tested prior to installation on a special developed fixture and level track with air bearing supports that allowed full deployment under simulated zero gravity environment. When the booms were tested in the fixture they had high deployment times which required replacing the outboard rotation joints. The units then successfully passed testing. The discone antenna booms were shipped to KSC and mounted to the FAS/DA. All mechanical parameters were adjusted and/or referified.

F. Weight and CG - Actual weight and center of gravity determinations were performed at MDAC-E on the Airlock Module, Fixed Airlock Shroud, and Deployment Assembly individually. No problems developed during the testing and the results were satisfactory. The measured data was coordinated with the Project Weights Group for updating the overages and shortages for issuance of actual flight weight report. Continued monitoring of the vehicle after delivery to KSC provided an accurate launch weight and center of gravity determination.

G. Alignment - Alignment and alignment verification were integrated with the deployment assembly test This approach resulted from close coordination with design engineering, manufacturing and the test conductors to eliminate duplication of manufacturing operations, eliminate duplicate testing effort, and obtain more precise alignment. Review of the vehicular alignment parameters indicated that the requirements were within the capability of "off-the-shelf" optical instrument and standard equipment. However, many of the requirements were considered to be outside the normal "built-in-by-manufacturing" capability. Therefore, it was necessary to perform alignment during the assembly and stacking operations using optical instruments. In general, the blueprint alignment tolerances were more stringent than those specified on the interface control documents. Working tolerances for the tests were usually less than the blueprint tolerances. This philosophy allowed a portion of blueprint tolerance and some interface cushion for alignment degradation due to disassembly at St. Louis, shipment and reassembly at KSC.

problems developed in meeting the working tolerances. The alignment approach described above was used on the following:

Interface Alignments

STS/MDA
Tunnel/Bellows
Radio Noise Burst Monitor
EREP (S193)
VHF Ranging Antenna
ATM/DA

Hardware Alignments

Deployment Assembly
Discone Antennas
Axes Identification

2.2.3.3 Integrated Testing

A. STS and EVA Hatch Window Inspection - Structural requirements for the STS viewing ports and EVA hatch window specified that each completed assembly be tested at 14.6 ± 1.0 psid in the volume between the two panes. In addition, all surface defects such as scratches, sleeks, and coating nonconformities were to be recorded on a full scale drawing immediately prior to the 14.6 ± 1.0 psid test and a comparison made again during the inspection sequence at KSC. The 14.6 ±1.0 psid test was completed in conjunction with the AM/MDA altitude chamber test by evacuating the 30-foot space chamber to 1 psia and maintaining ambient pressure in the volume between the window panes with a net result of 13.6 psid as a proof pressure test load. Only one new defect was detected during the KSC mapping activity. This new defect was viewed with a 10 power magnifier and determined to be of insufficient magnitude to materially affect the integrity of the glass. None of the defects recorded affected the optical requirements of the windows.

B. AM/MDA Leakage Testing at MDAC-E - Two leakage tests were performed prior to the Altitude Chamber Test. In the first test the AM/MDA was pressurized to 5 psig using GN_2. The recorded leak rate was 2350 SCCM versus an allowable of 5930 SCCM. The second and final leakage test was performed in the altitude chamber with the chamber evacuated to simulate

150,000 ft. and with the AM/MDA pressurized to 5 psia using GN_2. The recorded leak rate was 345 SCCM versus an allowable of 1450 SCCM.

C. EVA/Internal Hatch Flight Seal Installation and Mechanism Reverification - To preclude any possibility of the vehicle being launched with damaged seals, the seals used during all test activity at MDAC-E were replaced at KSC prior to flight. Reverification of the critical rigging requirements was performed after installation of the flight seals.

D. AM/MDA Leakage Test at KSC - The final prelaunch leakage rate at KSC was 2049 SCCM (N_2) at 5 psig versus an allowable of 3930 SCCM.

E. Clearance and Fit Checks - Mechanical and electrical fit checks were performed prior to shipment from St. Louis to assure compatibility with the major modules of various component packages including the S193 experiment and the VHF antenna. Clearance checks were performed in St. Louis to verify that the vehicle could be assembled in the planned sequence upon arrival at the launch site. The DA was hoisted past the mated AM/MDA to assure no clearance problems would later hinder assembly. The upper DA was deployed to its orbital position to assure clearance between the DA and the MDA and associated protrusions.

F. GSE Testing - Mechanical GSE was tested in accordance with the intended use of the equipment. These items were designed to perform hoisting, handling, transportation and mechanical testing functions. Handling and hoisting equipment was proof loaded prior to first usage and at regular intervals thereafter. Mechanical test and transportation GSE that involved functional systems was tested in accordance with acceptance test procedures which reflected design requirements prior to first usage on flight equipment. Where required to assure reliability, readiness tests were performed on all GSE which mated with the flight vehicle prior to use in St. Louis or shipment of the GSE to KSC. Where required to assure reliability, readiness tests were performed prior to each usage of GSE. Trial fit checks were made of the GSE to the static test article prior to usage on U1 flight equipment to preclude possible damage to flight equipment. Periodic inspections were performed on all GSE while in St. Louis to assure any time oriented degradation was detected and corrected.

2.2.4 Mission Results

The structural integrity of the vehicle was maintained throughout the mission and mechanical systems functioned without failure. No special tests were required or conducted to support the mission.

2.2.4.1 Airlock Module

The launch loads on all the AM structural components were well within the design tolerances. Gas leakage for the entire cluster, with leakage rates well within their requirements, indicated that all the pressure vessels and joints were in excellent condition. Adequate structural support of the O_2 and N_2 bottles was also demonstrated.

All crews reported the AM tunnel size as "almost ideal," being large enough to work in, yet small enough to allow the crew to use the walls to push against. The SL-2 crew also commented that the STS size and equipment arrangement were "generally good" and that the bellows area was adequate for equipment transfer.

During both SL-2 EVA's, fogging was noted in the lock compartment at around 3.5 to 3.0 psia while venting the gas pressure. This fogging was visible as it streamed through the vent valve. Moisture began to collect and freeze on the vent, tending to impede the lock depressurization. A screen cap cover for the vent was designed and launched on SL-3. The screen cap was to trap the ice formation so the screen could be removed to free the vent orifice and improve gas flow; it functioned without problems during SL-3 and SL-4.

2.2.4.2 DA

ATM DA deployment performed as planned with no need for DCS backup commands. The rigidizing mechanism rigidized the ATM to the DA upon payload shroud separation.

The TM data, M-0013-530 at 16 min. 36.69 seconds after lift-off and M-0014-530 at 16 min. 36.79 seconds after lift-off, verified capacitor charge and discharge indicating proper initiation of the DA release mechanisms which allowed normal deployment of the ATM/DA.

Voltage was applied to both DA motors at 16 minutes, 52 seconds after lift-off. Three minutes, 11 seconds later the position switch telemetry indicated deployment and latching completed. A comparison of time to deploy with ground test time verified that both deployment motors performed properly and that there was no abnormal operation of the system.

The redundant switches on the latch mechanism both functioned to initiate turn-off of the DA motors, triggered a TM signal that deployment was completed, and removed the inhibit from the ATM solar array deployment system.

Telemetry indicated latching at 20 minutes, 3 seconds after lift-off. Latching was further verified a few minutes later by normal deployment of the ATM SAS.

Data obtained during docking and orbital maneuvering from accelerometers in the ATM rate gyros indicated the ATM had rigidized and the ATM/DA had a natural frequency greater than 0.6 Hertz.

2.2.4.3 FAS

Telemetered accelerometer and pressure data show that FAS design loads were not exceeded and the SL-2 crew reported the FAS structure and FAS mounted equipment to be in excellent condition. Fixed skin temperatures were monitored at four locations near the vehicle axes throughout the mission. Figure 2.2-18 shows the maximum temperature at each location for day of year 154 thru 178 (1973).

2.2.5 Conclusions and Recommendations

As a result of the highly successful prelaunch, launch, ascent, and in-orbit performance of the structure and mechanical systems, no changes or improvements are recommended.

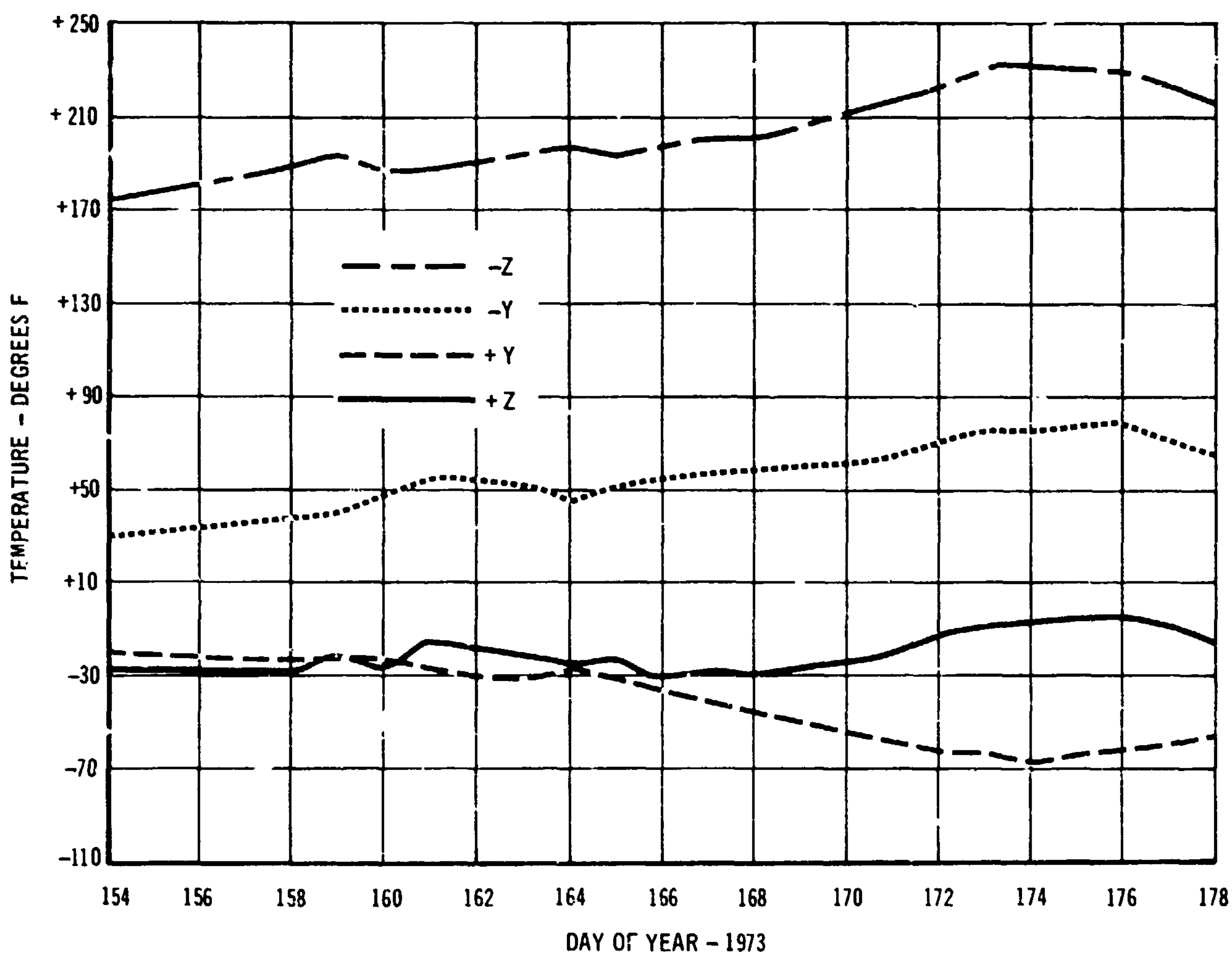


FIGURE 2.2-18 FIXED AIRLOCK SHROUD MAXIMUM DAILY TEMPERATURE

2.3 MASS PROPERTIES

Airlock mass properties which included the basic AM, DA, FAS, experiments, MDAC-E designed MDA components, O_2, N_2, stowed and fixed GFE, and PS, were computed and maintained on a current basis. These data were reported to the NASA monthly in accordance with the mass properties requirements specified in the Statement of Work. The Airlock maximum specification weight was established by the Airlock Performance/Configuration Specification, MDC Report E946 and Payload Shroud Detail CEI Specification, MDC Report E0047.

2.3.1 Airlock Weight Monitoring Plan

Although the Saturn V launch vehicle had ample boost capability for the Skylab mission a weight monitoring and control plan (See Figure 2.3-1) was used to assure continuous Airlock mass properties status. Bogey weights were established for all components and incorporated inhouse and on vendor specifications. Mass properties were calculated for all drawings and overweight conditions reported for corrective action. As actual weights replaced calculated weights, reports were updated to reflect current conditions. The MIL-STD-176A functional weight code identification was utilized to categorize and report weights. The result was published and updated in Airlock Project Mass Properties Status Report, 233-M-501-XX.

The monitoring plan continued at the launch site when parts/materials installed and removed from the vehicle were weighed and recorded in accordance with KSC POP 4-003, Weight and Balance Control Procedure. Daily logs were maintained by MDAC-E personnel through launch. These data were also reported in the Mass Properties Report mentioned above.

2.3.2 Actual Weight Program

An actual weight program was pursued at MDAC-E where as many detail manufactured parts as possible were weighed and, in many cases, assemblies were weighed. In all cases these data were used to verify or adjust calculated weights as soon in the program as possible to verify the predicted launch weights and to give confidence in the total mass properties program. Figure 2.3-2 illustrates the estimated/calculated/actual weight history of the Airlock Module as it evolved in the last two years prior to launch. It should be noted that at the time of delivery, (less gas and fluid expendables

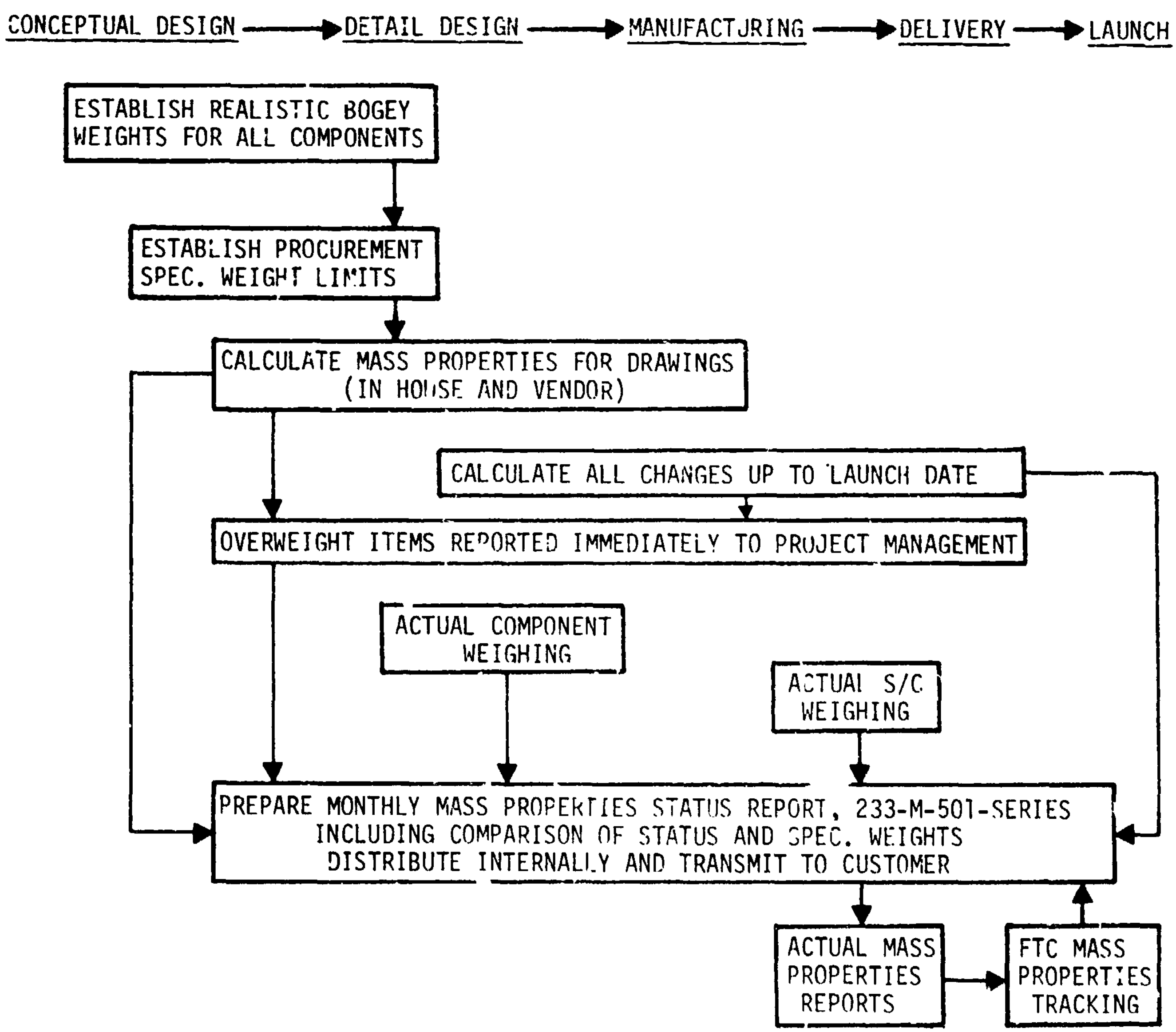


FIGURE 2.3-1 WEIGHT MONITORING PLAN (INCLUDING ALL MASS PROPERTIES)

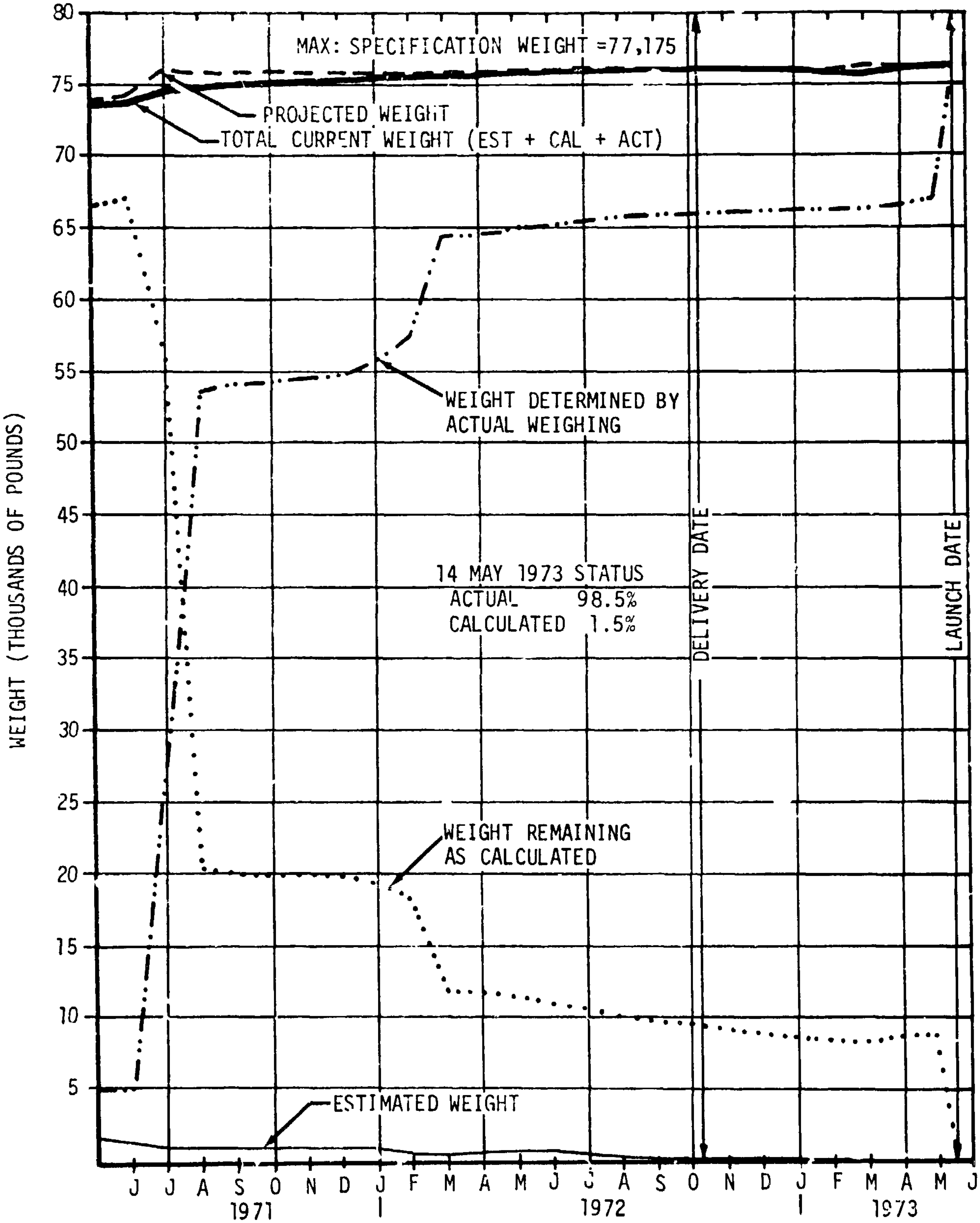


FIGURE 2.3-2 AIRLOCK WEIGHT HISTORY

not yet serviced) the percent actual weight status of the Airlock Modules was 97.2%.

At the appropriate time in the manufacturing/system test cycle each of the module assemblies (i.e. the basic AM, DA, FAS, and PS) were weighed and balanced. Figure 2.3-3 gives the flow sequence of the weighing and C.G. determination activity for the AM, FAS and DA. These data were reported to the NASA in separate data packages as part of the acceptance data package for each module. In addition these data were incorporated into the Airlock Project U-1 Mass Properties Status Report. Results of the module weighings indicated the maturity of the Airlock Mass Properties Program. (See Figure 2.3-4). As shown the total difference between measured and calculated weight was 14 lbs., i.e., a .02% difference.

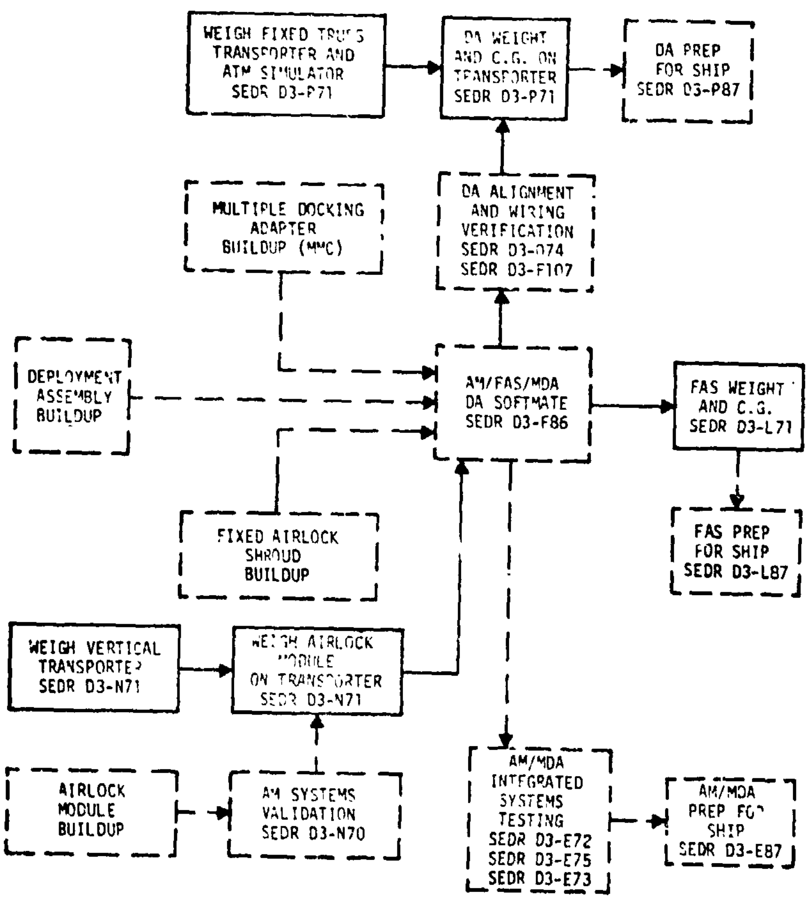


FIGURE 2.3-3 WEIGHING AND CENTER OF GRAVITY DETERMINATION FLOW

	WEIGHT (POUNDS)			
	CALCULATED	MEASURED	Δ	% DIFFERENCE
AM	16888	16937	+ 49	.29
DA	4069	4044	- 25	.62
FAS	28312	28408	+ 96	.34
TOTAL	49269	49389	+120	.24
PS	25617	25483	-134	.53
TOTAL	74886	74872	- 14	.02

	CENTER OF GRAVITY (INCHES)		
	CALCULATED	MEASURED	Δ
AM Y	2.8	2.8	0
AM Z	4.2	4.2	0
DA Y	0.1	-0.3	.4
DA Z	2.1	2.3	.2
FAS Y	-16.3	-15.9	.4
FAS Z	16.3	16.0	.3
TOTAL Y	-8.4	-8.2	.2
TOTAL Z	11.0	10.8	.2
PS X	377.7	377.7	0
PS Y	.4	.5	.1
PS Z	0	.1	.1
TOTAL Y	-5.4	-5.2	.2
TOTAL Z	7.1	7.1	0

FIGURE 2.3-4 AIRLOCK MODULE ACTUAL WEIGHT AND BALANCE RESULTS VERSUS CALCULATED

2.3.3 Launch Weights

The U-1 Airlock Launch Weights versus the Maximum Specification Weights are tabulated on Figure 2.3-5. The total U-1 Airlock Module and Payload Shroud weight, including Government Furnished Equipment was 75,978 pounds. That was 1,197 pounds under Maximum Specification Weight. The complete weight details are in the final report, Airlock Project U-1 Mass Properties Status Report, 233-M-501-69, dated 1 June 1973.

ITEM	LAUNCH WEIGHT *	MAXIMUM SPECIFICATION WEIGHT *	WEIGHT MARGIN
AIRLOCK MODULE (BASIC)	15166	15416	250
DEPLOYMENT ASSEMBLY	3744	3880	136
FIXED AIRLOCK SHROUD	22749	22922	173
MDA (MDAC-EAST COMPONENTS)	501	500	-1
TOTAL CFE (INCL. PROV. FOR GFE)	42160	42718	558
EXPERIMENTS	311	341	30
OXYGEN	6085	6100	15
NITROGEN	1624	1630	6
OTHER GFE PER SOW, EX. B	325	362	37
TOTAL GFE	8345	8433	88
TOTAL AM (INCL. GFE)	50505	51151	646
PAYLOAD SHROUD (CFE)	25473	26024	551
TOTAL AM & PS (INCL. GFE)	75978	77175	1197

* WEIGHT IN POUNDS

FIGURE 2.3-5 U-1 LAUNCH WEIGHT VERSUS MAXIMUM SPECIFICATION WEIGHT

2.4 THERMAL CONTROL SYSTEM

The Airlock thermal control system (TCS) provided temperature control for the Airlock and cooling to the MDA and OWS. It consisted of an active coolant system, ATM C&D Panel/EREP cooling system, battery cooling system, thermal coatings, thermal curtains, equipment insulation, AM wall heaters, and molecular sieve exhaust duct heaters. The active coolant system provided coldplates and heat exchangers for equipment and atmospheric cooling. Primary and secondary flow paths provided the required redundancy.

2.4.1 Design Requirements

The basic requirements of the TCS were to provide temperature control for the AM crew compartment, equipment, and structural surfaces during prelaunch, launch ascent, and orbital phases of the mission. In addition, interface requirements were to provide atmospheric cooling to the MDA and OWS, to provide temperature controlled coolant to the ATM and EREP panels, and to meet the thermal control interfaces to the MDA, OWS, ATM, CSM, IU, the experiments, and GSE, as shown in Figure 2.4-1. The AM/MDA interface requirements are presented in ICD 13M02521A and the AM/OWS interface requirements are presented in ICD 13M02519B. Additional requirements associated with the TCS design were to provide instrumentation intelligence and procedures as a basis for system operation.

2.4.1.1 Evolution

As the Skylab program evolved from the SSESM to the AAP concept, and ultimately to the Saturn V workshop concept significant changes in mission plan and systems design requirements were made. Consequently, the design requirements of the Airlock Module thermal control system also changed significantly throughout its development. This section presents a discussion of the significant changes to the Airlock Module thermal control system design and requirements.

A. External Design Requirements - The change to the Saturn V workshop concept resulted in several changes to the external thermal environment conditions. The change in orbit inclination angle from 35° to 50° increased the mission beta angle extremes from ±58 1/2° to ±73 1/2°. Combined with the change to the basic solar inertial attitude, this resulted in a more severe hot case external environment design condition.

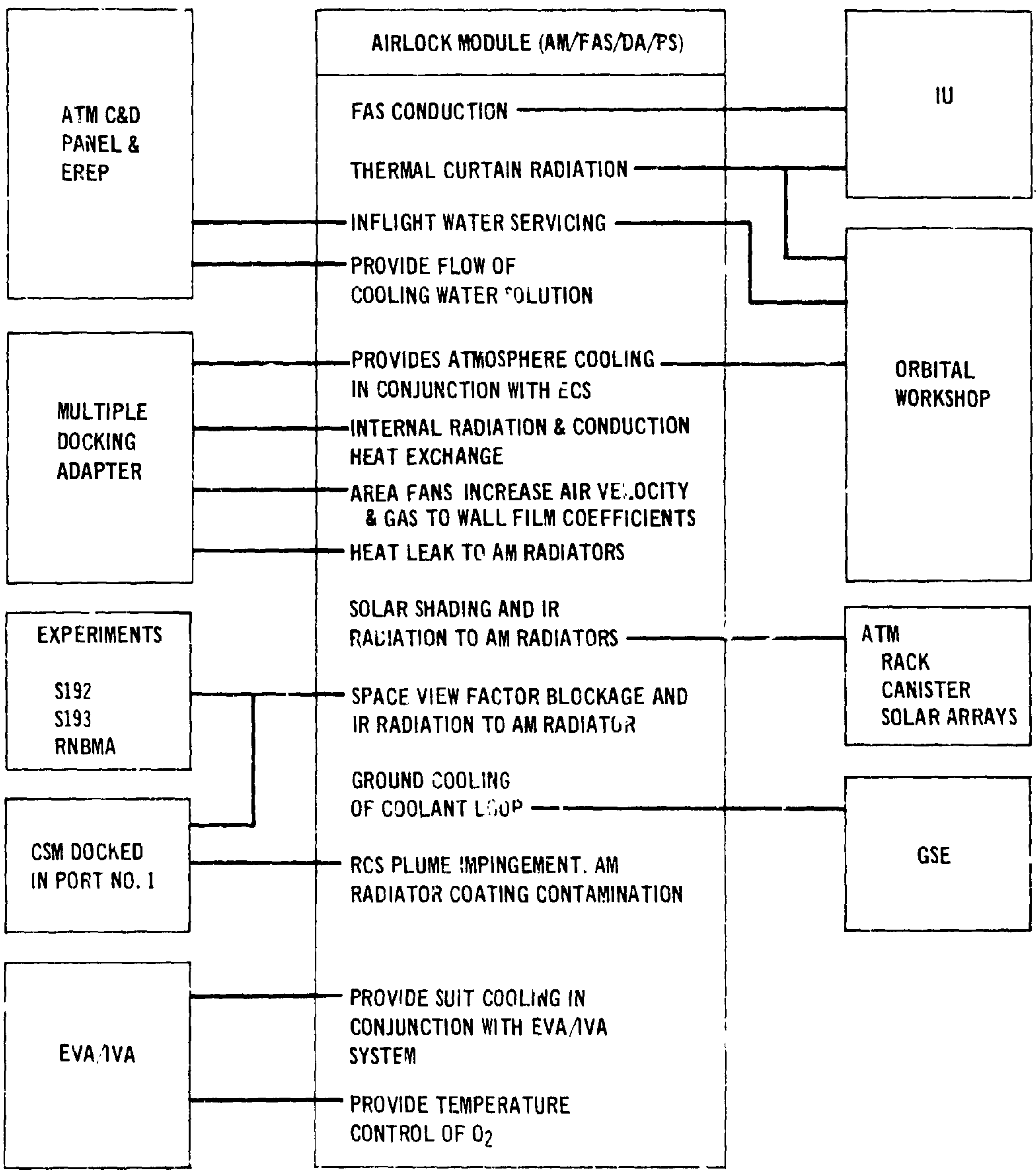


FIGURE 2.4-1 THERMAL CONTROL INTERFACE

The change to the basic solar inertial attitude also had an impact on the passive temperature control system, and resulted in the change from a multiple layer "superinsulation" concept to the thermal curtain insulation design. The addition of the EREP mode of operation required the incorporation of the Z-LV (Z-axis Local Vertical) vehicle orientation into the AM TCS design.

B. Internal Design Requirements - Several added requirements significantly affected the design of the active coolant subsystem of the AM TCS. In addition to the normal growth of heat loads associated with system definition, the requirement for reduced water delivery temperatures in the EVA/IVA suit cooling loops, as discussed in Section 2.6 had a major impact on AM coolant system design. Prior to the 45°F water delivery temperature requirement, the AM coolant loop design included a single 40°F TCV downstream of the radiator. To provide for the lower water delivery temperatures, a heat exchanger interfacing each water suit cooling loop with the associated AM active coolant loop was moved upstream of the 40°F TCV. To assure that temperature control would be maintained at the 40°F TCV, a thermal capacitor was added to the AM coolant loops immediately downstream of the radiator. The thermal capacitor was added to limit coolant supply temperatures to the EVA heat exchangers to a maximum of 28°F after it was concluded that improved radiator performance could not be achieved due to limited space available for increased radiator area, and also because of limitations on radiator thermal coating values (α/ε)

Concern over life of the AM batteries led to the requirement of providing lower coolant temperatures at the battery modules. In conjunction with a requirement to maintain a minimum cluster dew point temperature of 46°F, this resulted in the addition of two 47°F TCV's in each coolant loop and the design of the suit/battery cooling module. A second thermal capacitor was added to offset the reduced radiator performance due to the Z-LV orientation associated with EREP operation.

Design requirement changes that occurred during the program resulted in the need for expanded system capability. Examples of such changes in addition to those described above are the requirement for more OWS atmospheric cooling, addition of the ATM C&D/EREP water cooling system and the addition of various electronic equipment requiring cold plate cooling. Incorporation of the plumbing, cold plates, heat exchangers, and valves needed for meeting these requirements substantially increased system coolant flow resistance. This led to higher pressure levels in some portions of the loop, which exceeded component specification allowables during contingency situations requiring operation of two pumps in one coolant loop. System pressure levels were ultimately reduced by design changes such as modifying the radiator to a bifiler type design (i.e., parallel coolant flow paths), paralleling coolant flow paths in the battery and electronic modules, increasing plumbing size, and boring out standard plumbing connectors to larger internal diameters. Although the above methods eliminated the major portion of the pressure level problem, requalification of some of the off-the-shelf hardware to higher pressure levels was also required.

2.4.1.2 Flight Configuration

Thermal control system design requirements consist of external design requirements and internal design requirements.

A. External Design Requirements - The mission thermal design data are listed in Figure 2.4-2 per MDAC-ED Report F319. The original design vehicle orientation was the solar inertial. Short duration Z-LV (Z-axis Local Vertical) orientation for EREP and rendezvous as well as CMG desaturation maneuvers were accommodated. The EREP design maneuvers are illustrated by Figure 2.4-3 and the CMG desaturation maneuver by Figure 2.4-4. A fifth design orientation requirement was accommodated during the flight - namely the orientation necessary for study of the Kohoutek comet during the SL-4 mission. The Kohoutek comet viewing design maneuvers are described in Figure 2.4-5.

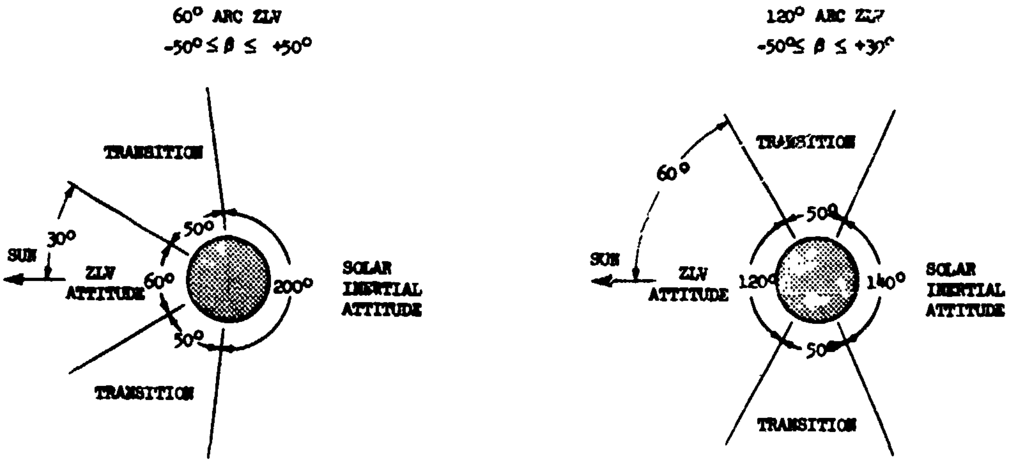
B. Internal Design Requirements - A summary of general design requirements for the active and passive subsystems of the Airlock Module thermal control system are shown in Figure 2.4-6. Detailed requirements of individual subsystems are discussed in the Systems Description Sections as indicated in Figure 2.4-6.

2.4.2 Integrated Thermal Analysis

An integrated thermal analysis was required to convert design requirements for the AM Thermal Control System (Section 2.4.1) to design requirements for each system (Section 2.4.3). The integrated thermal analyses were conducted using detailed mathematical thermal models. Since the AM was shaded partially by the ATM, was structurally attached to the IU and MDA, had seven of eleven radiator panels in the AM Coolant System mounted on a portion of the MDA, and provided cooling to the OWS and MDA with the atmospheric control system, the thermal effects of these structural and systems factors were included in the thermal model. In fact, the thermal model included a thermal representation of all pertinent structure and AM/MDA systems to permit calculation of realistic heat load division between systems. The AM/ATM, AM/MDA and AM/OWS interfaces are defined in ICD 13M20726, ICD 13M02521A, and ICD 13M02519B, respectively. Integrated thermal analyses also were required to thermally qualify the vehicle for flight. In addition, integrated thermal analyses as valuable mission support aids.

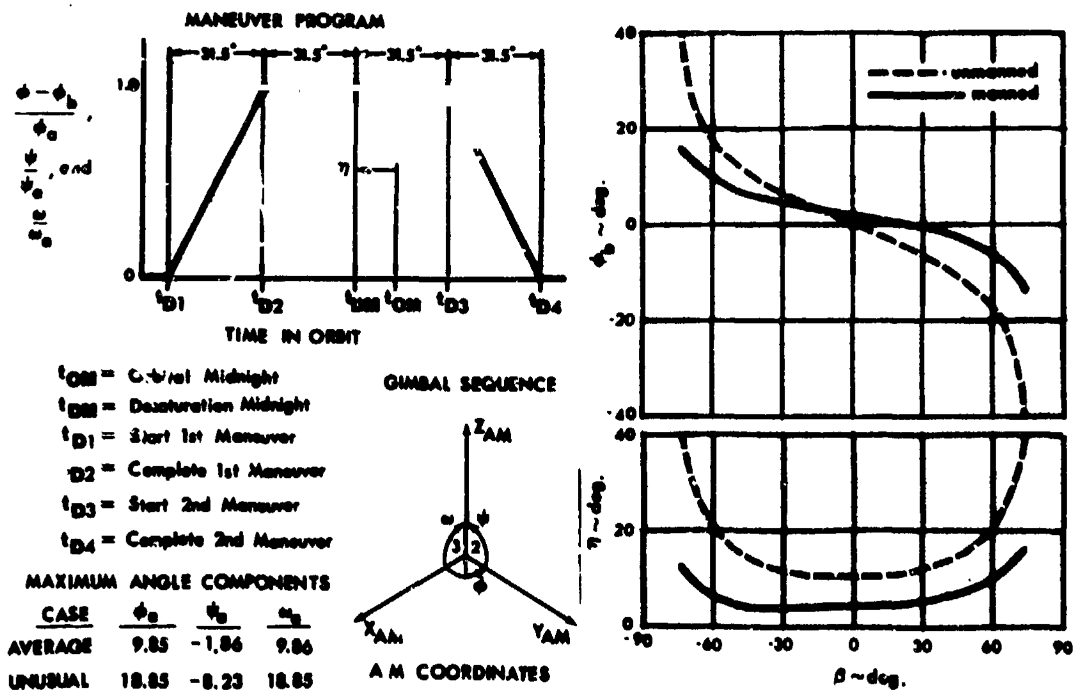
PARAMETER	MISSION DESIGN DATA
Ambient Pressure, Temperature and Density	0 to 120 Kilometers: U.S. Standard Atmosphere 1962 Above 120 Kilometers: Jacchia Model Atmosphere as specified in NASA Document SP-8021
Launch Time (SL-1)	14 May 1973 at 17:30 GMT
Launch Trajectory	Nominal per S&E-AERO-MFM-33-70 dated 13 March 1970 Off-Nominal per S&E-AERO-MFM-39-70 dated 20 March 1970
Solar Constant	429 $\pm$28 Btu/hr-ft^2
Earth Emitted Radiation	75 $\pm$11.33 Btu/hr-ft^2
Albedo (Earth)	0.30 $\pm$.102
Orbit Definition Altitude Beta Angles	 235 n.m. nominal (160 n.m. to 300 n.m.) -73.5° to +73.5°
Vehicle Orientation	Solar inertial orientation with the AM minus Z-axis towards the sun and the AM +X axis in the orbit plane and in the direction of the velocity vector at orbital noon will be the normal vehicle orientation except during earth resources experiment (EREP) performance periods and rendezvous, which utilize the Z-local vertical (Z-LV) mode. Rendezvous - The AM will be oriented in the X-IOP/Z-LV with the AM minus X-axis in the direction of the velocity vector. The transition from solar inertial to X-IOP/Z-LV will be initiated as early in a given orbit as orbital midnight. Return to solar inertial will be made at orbital midnight after a maximum of 2 orbits. Rendezvous may occur at Beta angles up to ±73.5° For $\|\beta\|$ > 50°, the vehicle will be rolled about the X-axis up to 23.5° maximum from the true Z-axis local vertical position. EREP - The EREP maneuvers used for AM thermal analyses are shown in Figure 2.4-3. During the Z-LV phase the AM will be oriented with the +Z axis towards earth center and the +X axis in the orbit plane (X-IOP/Z-LV) and in the direction of the velocity vector. The 60° arc ZLV orbits may occur singly or in pairs. The 120° arc ZLV orbits occur singly. At least 4 solar inertial orbits must follow each pair of 60° arc ZLV orbits and each 120° arc ZLV orbit. Single 60° arc ZLV orbits may be alternated with solar inertial orbits up to a maximum of 4 continuous sequences within a 24-hr period.
Beta Angle	Beta Angle (β) - Beta angle is defined as the geocentric angle between the sun and the Airlock at noon transit of the Airlock (the point of closest approach of the Airlock to the sun). Beta angle is positive if the Airlock orbit is clockwise when viewed from the sun.

FIGURE 2.4-2 THERMAL DESIGN DATA (ALL MISSION PHASES)



NOTE: A LINEAR RATE OF CHANGE IS ASSUMED IN THE TRANSITION FROM/INTO THE SOLAR INERTIAL ATTITUDE.

FIGURE 2.4-3 EREP DESIGN MANEUVERS



t_{OM} = Orbital Midnight
t_{DM} = Desaturation Midnight
t_{D1} = Start 1st Maneuver
t_{D2} = Complete 1st Maneuver
t_{D3} = Start 2nd Maneuver
t_{D4} = Complete 2nd Maneuver

MAXIMUM ANGLE COMPONENTS

CASE	ϕ_a	ψ_a	ω_a
AVERAGE	9.85	-1.86	9.86
UNUSUAL	18.85	-8.23	18.85

FIGURE 2.4-4 CONTROL MOMENT GYROS DESATURATION MANEUVERS

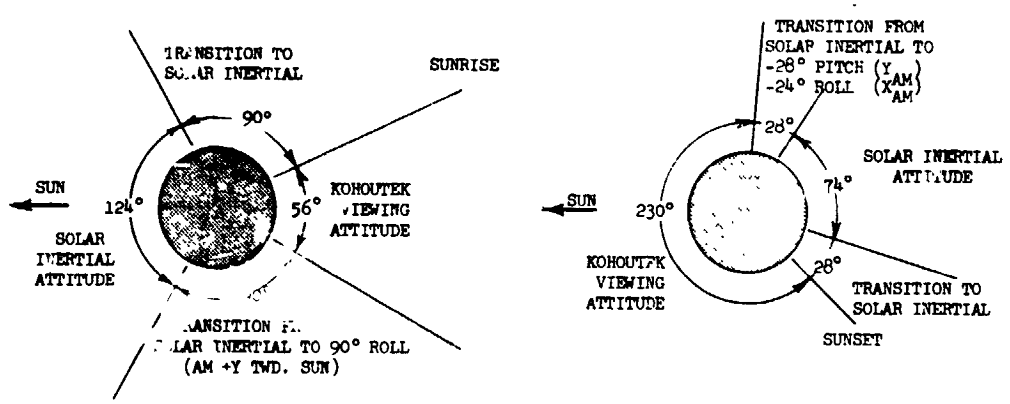


FIGURE 2.4-5 KOHOUTEK COMET VIEWING DESIGN MANEUVERS

SUBSYSTEM	GENERAL REQUIREMENT	DETAILED REQUIREMENT
• ACTIVE SUBSYSTEM		
• COOLANT SYSTEM	PRELAUNCH - REJECT HEAT VIA A GROUND COOLING HEAT EXCHANGER, GROUND COOLING CART, AND FAS FLY-AWAY UMBILICAL SYSTEM.	SEE SYSTEMS DESCRIPTION. SECTION 2.4.3.2
	ORBIT OPERATIONS - REJECT HEAT THROUGH A RADIATOR/CAPACITOR SYSTEM CAPABLE OF REJECTING NOT LESS THAN 12,000 BTU/HR FOR EVA OPERATIONS AND NOT LESS THAN 16,000 BTU/HR FOR OTHER NORMAL OPERATIONS.	SECTION 2.4.3.2
• ATM C&D/EREP COOLING		SECTION 2.4.3.3
• BATTERY COOLING		SECTION 2.4.3.4
• AM WALL HEATERS		SECTION 2.4.3.8
• MOLECULAR SIEVE EXHAUST DUCT HEATERS		SECTION 2.4.3.9
• PASSIVE SUBSYSTEM	IN CONJUNCTION WITH THE ACTIVE THERMAL CONTROL SUBSYSTEM, CONTROL AIRLOCK SYSTEM TEMPERATURES WITHIN ALLOWABLE LIMITS BY APPROPRIATE SURFACE COATINGS, INSULATION, AND EQUIPMENT LOCATION. PROVIDE SUITABLE THERMAL INTERFACES WITH OTHER VEHICLES OF THE ORBITAL CLUSTER PER APPLICABLE INTERFACE DOCUMENTS.	
• THERMAL COATINGS		SECTION 2.4.3.5
• THERMAL CURTAINS		SECTION 2.4.3.6
• EQUIPMENT INSULATION		SECTION 2.4.3.7

FIGURE 2.4-6 THERMAL CONTROL SYSTEM DESIGN REQUIREMENTS

Other structures and systems were included in the thermal model. The OWS forward skirt and dome were represented to complete the enclosure formed with IU and FAS. Electrical wall heaters and thermostats operation was simulated for both the AM and MDA to verify adequate heater power for cold mode operations. Heat leaks from the MDA walls to the AM radiator were calculated to define radiator/capacitor system heat loads. Convective heat transfer between MDA walls and equipment and the cabin atmosphere was also included as a part of the overall system heat balance to establish heat load split between the wall heating system, equipment, and atmospheric control system. The ATM C&D panel and EREP components were included in the model to provide a realistic distribution of heat loads to the cabin and the ATM/EREP water cooling loop. A thermal representation of the electrical equipment from the electrical power system, instrumentation system, caution and warning system, and communication system was included.

An operational plan for the cluster was assumed so that heat loads could be established. It was decided that all operating modes could be represented adequately for design with eight basic cases. They consisted of one prelaunch/launch ascent mode case and seven orbital operation cases.

2.4.2.1 External Design Heat Loads

The external heat load conditions for the seven orbital cases are shown in Figure 2.4-7. Analyses for the hot modes and the EVA/IVA mode were based on maximum beta angle and high external heating rates. Cold mode analyses utilized zero beta angle and low external heating rates for minimum orbital heating. Nominal heating rates were used for the rendezvous and EREP analyses, as indicated in Figure 2.4-7.

MODE	± β ANGLE ⚠1	ATTITUDE ⚠1	HEATING ⚠2
HOT AM/MDA	73 1/2°	SOLAR INERTIAL	HIGH
HOT OWS	73 1/2°	SOLAR INERTIAL	HIGH
COLD AM/MDA/OWS	0	SOLAR INERTIAL	LOW
ORBIT STORAGE	0	SOLAR INERTIAL	LOW
EVA/IVA	73 1/2°	SOLAR INERTIAL	HIGH
RENDEZVOUS	73 1/2°	RENDEZVOUS	NOMINAL
EREP	50°	EREP	NOMINAL

⚠1 BETA ANGLE AND ATTITUDES DEFINED IN FIGURE 2.4-2

⚠2 FOR SOLAR CONSTANT, EARTH IR, AND ALBEDO (SEE FIGURE 2.4-2):

3. PLUS TOLERANCES USED FOR HIGH FLUXES
4. MINUS TOLERANCES USED FOR LOW FLUXES
5. ZERO TOLERANCES USED FOR NOMINAL FLUXES

FIGURE 2.4-7 EXTERNAL DESIGN HEAT LOAD CONDITIONS – ORBITAL

2.4.2.2 Internal Design Heat Loads

The internal heat loads defined for the eight basic cases are shown in Figure 2.4-8. The compartment and coldplate loads were used as inputs to the thermal model to calculate structure and systems temperatures and heat flows. The AM compartment heat loads were based on the electrical equipment operation shown in Figure 2.4-9 for each operating mode. The AM/MDA wall heater loads shown in Figure 2.4-8 were predicted based on thermal model output, and were included as a part of the gross system heat loads. The external heat leaks were also calculated with the use of the thermal model, and were used to determine the net radiator heat load.

	OPERATING MODE							
SOURCE	HOT AM/MDA	HOT OWS	COLD AM/MDA/OWS	EVA/IVA	EREP	ORBIT STORAGE	PRE LIFT-OFF	RENDEZVOUS
1. COMPARTMENT LOADS								
MDA (2)	(1056)	(798)	(798)	(762)	(1056)	(0)	(0)	(0)
AM (TOTAL) (3)	(2003)	(1392)	(792)	(1353)	(1432)	(0)	(0)	(0)
AFT	361	276	100	100	276	0	0	0
LOCK	296	139	0	269	9	0	0	0
FWD	0	0	0	0	0	0	0	0
STS	1373	977	692	984	1147	0	0	0
OWS ATMOSPHERE COOLING	(1740) (4)	(1900) (4)	(55)	(0) (16)	(920) (12)	(0)	(0)	(0)
METABOLIC (5)	(1245)	(530)	(0)	(0)	(1565)	(0)	(0)	(0)
SENSIBLE								
MDA	480	0	0	0	690	0	0	0
AM	235	0	0	0	345	0	0	0
LATENT	530	530	0	0	530	0	0	0
2. AM/MDA WALL HEATER LOAD	(0)	(0)	(1858)	(0)	(0)	(1200)	(0)	(1790)
3. COLDPLATES, HX'S ETC.	(5668)	(4695)	(5094)	(9659)	(6064)	(3203)	(1885)	(3782)
EVA/IVA HX'S	0	0	0	5064 (6)	0	0	0	0
ATM C&D PANEL/EREP HX	1126 (7)	153 (8)	153 (8)	153 (8)	1042 (9)	0	0	0
ELECTRONICS MODULES (10)	1116	1116	1116	1116	1116	315	387	944
PCG'S (EIGHT) (11)	2750	2750	3200	2650	3230	2550	1160 (15)	2500
TAPE RECORDERS	102	102	51	102	102	51	51	51
COOLANT PUMP MODULE	574	574	574	574	574	287	287	287
4. TOTAL HEAT LOADS								
GROSS SYSTEM HEAT LOAD	(11,712)	(9315)	(8597)	(11,774)	(11,037)	(4403)	(1885)	(5572)
EXTERNAL HEAT LEAK (13)	1,177	715 (14)	2343	574 (14)	1,500	1307	-5957	1672 (14)
RADIATOR HEAT LOAD	10,535	8600	6254	11,200	9,537	3096	7842 (17)	3900

(1) BTU/HR

NOTES:

(1) NOMINAL EQUIPMENT AND METABOLIC HEAT LOADS FOR SUSTAINED OPERATION AT MISSION MODE INDICATED. (4 HR EVA LIMIT.)
(2) MDA COMPARTMENT HEAT LOADS PER AM/MDA ENVIRONMENTAL CONTROL DATA, S&E-ASTN-PL(72-130).
(3) BASED ON AM EQUIPMENT LOADS SHOWN ON FIGURE 2.4-9.
(4) BASED ON 83°F OWS RETURN GAS TEMPERATURE.
(5) CREW METABOLIC SENSIBLE LOADS PER S&E-ASTN-PL(72-214), BASED ON TOTAL METABOLIC LOAD OF 500 BTU/HR PER CREWMAN, CLO=0.35, V(GAS)=40 FT/MIN. LATENT METABOLIC HEAT LOADS EXCLUDE 220 BTU/HR (MOLECULAR SIEVE VENTING).
(6) 3130 BTU/HR (ONE EVA/IVA LOOP) + 1730 BTU/HR (OTHER EVA/IVA LOOP) + 204 BTU/HR (PUMPS).
(7) ATM C&D PANEL AVERAGE LOAD OF 310 WATTS (1058 BTU/HR) + PUMP LOAD (68. BTU/HR).
(8) ATM C&D PANEL AVERAGE LOAD OF 25 WATTS (85 BTU/HR) + PUMP LOAD (68. BTU/HR).
(9) ORBIT AVERAGE HEAT LOAD BASED ON 25 WATT ATM C&D PANEL LOAD DURING STANDBY, AND EREP EQUIPMENT LOAD PROFILE SHOWN IN FIGURE 2.4-31.
(10) BASED ON NOMINAL ELECTRONIC EQUIPMENT OPERATION; INCLUDES NONCOLDPLATED EQUIPMENT.
(11) ORBIT AVERAGE HEAT LOADS PER FIGURE 2.4-33.
(12) BASED ON 70°F OWS RETURN GAS TEMPERATURE.
(13) INCLUDES LOSSES TO CSM; EXCLUDES HEAT LEAK TO RADIATOR.
(14) ESTIMATED VALUE.
(15) BASED ON BATTERIES ON TRICKLE CHARGE.
(16) OWS POWERED DOWN
(17) REPRESENTS PRELAUNCH GCHX LOAD WITH RADIATOR IN BYPASS. BASED ON GCHX HEAT LOADS MEASURED DURING U-1 SEDR D3-E75 SIMULATED FLIGHT TESTS. TOTAL LOAD AT GSE/AM INTERFACE WITH GROUND COOLANT SUPPLY PER 65ICD9542, -15°F @ 900. LB/HR.

FIGURE 2.4-6 INTERNAL DESIGN HEAT LOADS

COMPARTMENT	LOAD	HOT AM/MDA	HOT OWS	COLD MODE	EVA/IVA	EREP	ORBIT STORAGE	PRE LIFT-OFF	RENDEZVOUS
		OPERATING MODE							
STS	FANS	(563.)	(563.)	(437.)	(400.)	(563.)	(0.)	(0.)	(0.)
	STS CABIN HX	126.	126.	0.	0.	126.			
	STS/OWS DUCT	37.	37.	37.	0.	37.			
	MOLECULAR SIEVES	400.	400.	400.	400.	400.			
	LIGHTS	(529.)	(144.)	(0.)	(309.)	(309.)	(0.)	(0.)	(0.)
	COMPARTMENT	220. (1)	0.	0.	0.	0.			
	INST. PANEL	165. (1)	0.	0.	165.	165.			
	PANEL METER LIGHTS	144. (1)	144.	0.	144.	144.			
	CONTROLS	(8.)	(8.)	(8.)	(8.)	(8.)	(0.)	(0.)	(0.)
	MOLECULAR SIEVES	5.							
	O_2/N_2 SYSTEM	3.							
	SENSORS	(82.)	(82.)	(82.)	(82.)	(82.)	(0.)	(0.)	(0.)
	FIRE	43.							
	PRESSURE	26.							
	OTHER	13.							
	MISC. EQUIPMENT	(61.)	(61.)	(51.)	(61.)	(61.)	(0.)	(0.)	(0.)
	SPEAKER INTERCOM	10.		0.					
	DIGITAL DISPLAY UNIT	39.		39.					
	DIGITAL CLOCK	12.		12.					
STS EQUIPMENT LOAD		1243.	858.	578.	860.	1023.	0.	0.	0.
ELECTRICAL LOSSES		20.	9.	4.	14.	14.	0.	0.	0.
TOTAL STS EQUIPMENT LOAD		1263.	867.	582.	874.	1037.	0.	0.	0.
MOLECULAR SIEVE GAS LOAD		110.	110.	110.	110.	110.	0.	0.	0.
TOTAL STS COMPARTMENT LOAD, BTU/HR		1373.	977.	692.	984.	1147.	0.	0.	0.
FWD	TAPE RECORDERS (3)	0.	0.	0.	0.	0.	0.	0.	0.
LOCK	LIGHTS								
	COMPARTMENT	252. (1)	126. (2)	0.	252. (1)	0.	0.	0.	0.
	PANEL METERS	9.	9.	0.	9.	9.	0.	0.	0.
	ELECTRICAL LOSSES	8.	4.	0.	8.	0.3	0.	0.	0.
	TOTAL LOAD	(269.)	(139.)	(0.)	(269.)	(9.3)	(0.)	(0.)	(0.)
AFT	OWS CABIN HX FANS	176.	176.	0.	0.	176.	0.	0.	0.
	LIGHTS	166. (1)	83. (2)	83. (2)	83. (2)	83. (2)	0.	0.	0.
	FIRE SENSORS	14.	14.	14.	14.	14.	0.	0.	0.
	ELECTRICAL LOSSES	5.	3.	3.	3.	3.	0.	0.	0.
	TOTAL LOAD	(361.)	(276.)	(100.)	(100.)	(276.)	(0.)	(0.)	(0.)

(1) LIGHTS ON BRIGHT
(2) LIGHTS ON DIM
(3) ON COLDPLATES (51. BTU/HR PER RECORDER)

FIGURE 2.4-9 AM COMPARTMENT HEAT LOADS (BTU/HR AT 28 VOLTS)

2.4.3 System Description

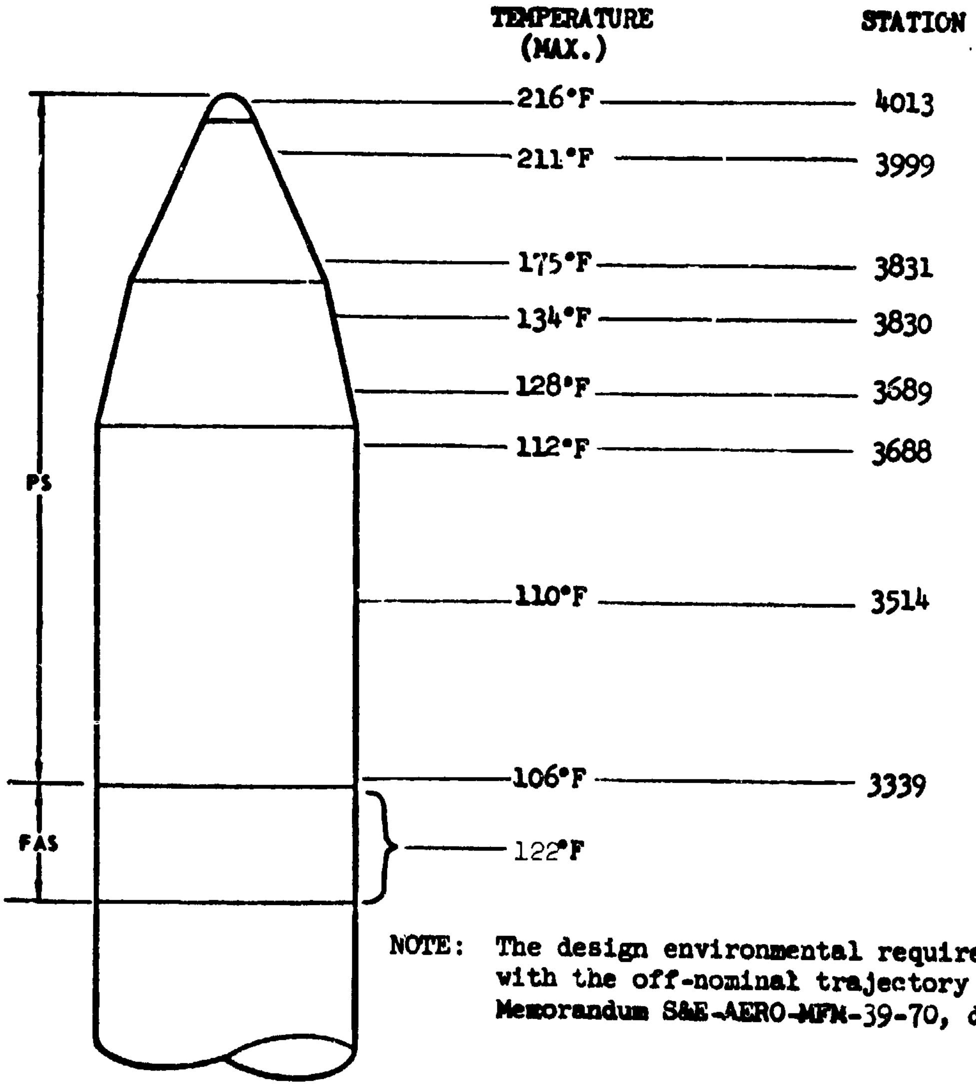
The following description of the TCS subsystems reflects the as-flown configuration. The TCS subsystems made up the overall TCS to provide for temperature control of: AM structure and equipment, AM crew compartments, suit cooling system, water solution for ATM C&D/EREP cooling, and atmospheric gas for OWS and MDA cooling.

Both active and passive techniques were used in the TCS subsystems to provide the necessary temperature control. The payload shroud provided for temperature control of structure and equipment during prelaunch, launch, and ascent. The temperatures of AM structure and crew compartment surfaces were controlled during orbital operations by the thermal coatings, thermal curtains, and equipment insulation subsystems utilizing passive techniques, and by the AM wall heating subsystem using active techniques.

Equipment temperature control utilized the active techniques of the coolant subsystem, battery cooling subsystem, and AM wall heating subsystem, in addition to the passive techniques of the thermal coatings, thermal curtains, and equipment insulation subsystems. Temperature control of suit cooling water in the EVA/IVA Suit System (Section 2.6) was provided by the coolant subsystem in conjunction with equipment insulation. Similarly, the ATM C&D/EREP cooling water temperature was controlled through heat exchange with the coolant subsystem, and by the thermal coatings and thermal curtains subsystems. The atmospheric gas temperature control was provided by the exchange of heat between the coolant subsystem and the atmospheric control subsystem in the Environmental Control System heat exchangers, as described in Section 2.5.

2.4.3.1 Payload Shroud

The payload shroud supported the prelaunch purge and protected against aerodynamic heating of payload during launch ascent. The external temperatures during launch and ascent, Figure 2.4-10, were predicted using solar heating at launch time, the off-nominal launch trajectory and standard atmosphere listed in Figure 2.4-2.



NOTE: The design environmental requirements are compatible with the off-nominal trajectory data contained in MSFC Memorandum S&E-AERO-MFM-39-70, dated 20 March 1970.

FIGURE 2.4-10 EXTERNAL SURFACE TEMPERATURE PROFILE DURING LAUNCH AND ASCENT

Results of the thermal analyses showed that the maximum payload shroud skin temperature remained below 220°F; the maximum temperature of the separation joint bellows and ordnance remained below 150°F. All of these temperatures were below the allowable limits.

The prelaunch purge aspect of the payload shroud performance is discussed further in Section 2.5, Environmental Control System. Other data pertaining to the payload shroud is presented in Payload Final Technical Report, G4679A.

2.4.3.2 Coolant System

The coolant system, illustrated schematically in Figure 2.4-11, provided temperature control of EVA/IVA, ECS, and equipment heat loads by exchanging heat with coolant fluid whose temperature and flowrate were controlled for this purpose. The coolant system consisted of primary and secondary coolant loops containing pumps, inverters, radiator, heat exchangers, coldplates and valving controls. The two separate coolant loops provided redundancy in that each loop was capable of removing and dissipating the anticipated waste heat.

A. Coolant Pump Subsystem - Coolant was circulated for heat transfer. Each coolant loop had two pump assemblies. One pump assembly in each coolant loop contained two pump/motor units, two check valves, and a reservoir. The other pump package contained one pump/motor unit, one check valve, and a reservoir. Each loop had an inverter assembly containing three inverters to provide AC power to the three pump/motor units in that loop. Planned operation was for one pump in one loop to be powered during prelaunch and unmanned orbital storage, while one pump in each loop would be powered for normal manned operations. Pump and inverter selection was by either DCS command or on-board switch selection. The on-board control was from ECS control panel 203 shown on Figure 2.4-12 and from circuit breaker panel 200.

   Because of only one coolant loop operating during unmanned operations, and the possibility of long times between ground station passes, an automatic pump switchover system was provided to automatically switch pump operation to the standby loop in case of a coolant loop failure. The system could be enabled by DCS command, and would

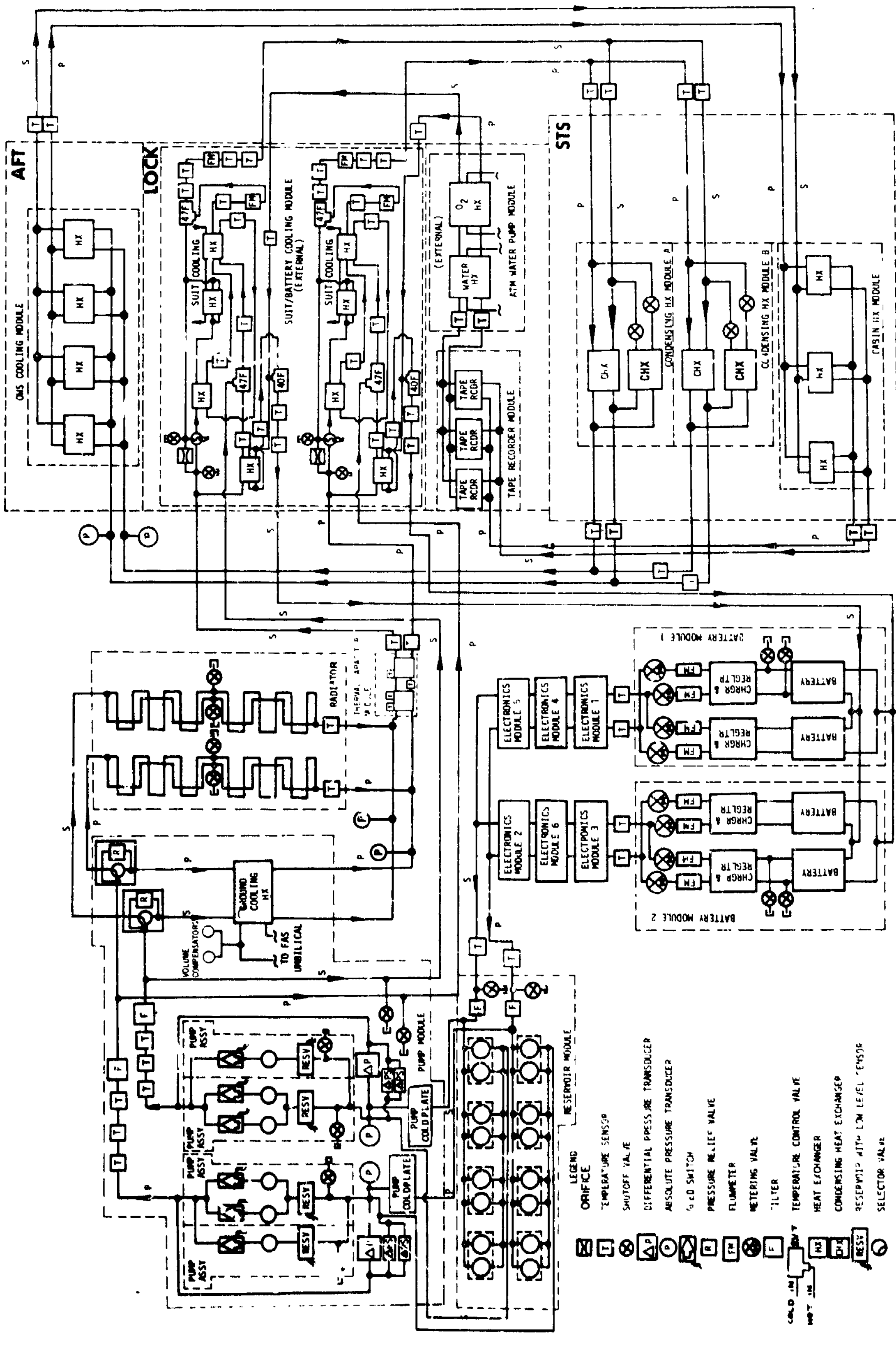


FIGURE 2.4-11 COOLANT SYSTEM

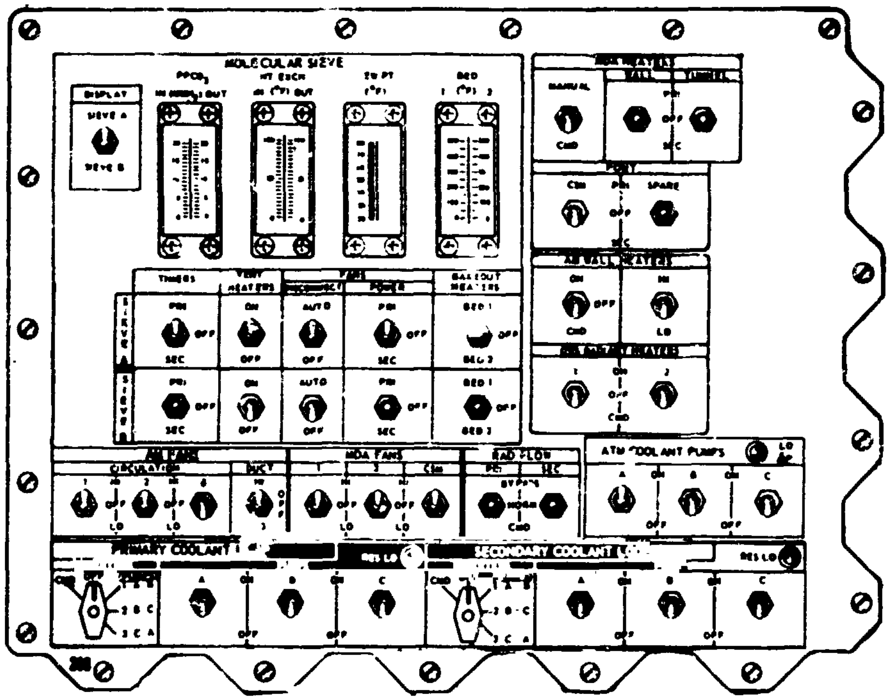


FIGURE 2.4-12 ECS CONTROL PANEL 203

result in switching of power from the failed loop to the corresponding pump inverter in the other loop if the pressure rise across the pump decreased to 18 ±1 psid or if the downstream 47°F TCV outlet temperature fell below 38 ±2°F.

During manned operations, with normal two coolant loop operation, coolant system parameters were monitored by the caution and warning system to provide a caution signal if certain conditions occurred. Coolant pump outlet temperatures were monitored to warn of equipment coldplate outlet temperatures above 120 ±2°F, and coolant pump flow was monitored to warn of low flow conditions.

(1) Pumps - To provide a relatively constant flowrate for a wide range of conditions, gear type pumps were used in conjunction with a three phase induction motor. The range of pump performance available is shown on Figure 2.4-13. The pump characteristic limits were determined by combining the highest flowrate pump/motor combination in a package with the highest frequency inverter, and the lowest flowrate pump/motor combination in a package with the lowest frequency inverter.

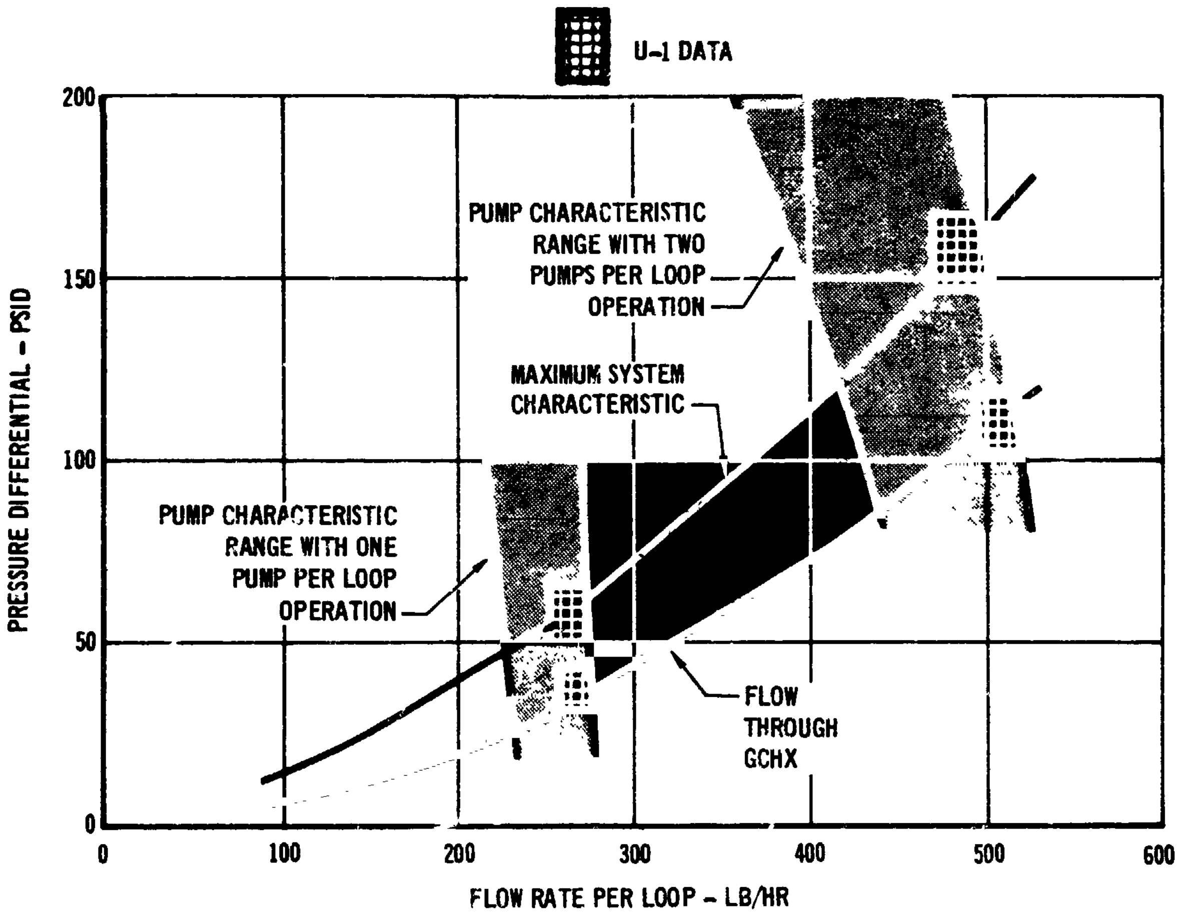


FIGURE 2.4-13 COOLANT SYSTEM FLOW PERFORMANCE

The system pressure drop characteristic was based on calculations using 50°F isothermal MMS-602 coolant with full flow through the radiator. A combination of specification control drawing (SCD) values, development data, and acceptance test data was used for pressure drop characteristics of vendor components. The filters were assumed to be free of dirt. Moody charts and equivalent pressure loss coefficients were used for analysis of smooth tubes, fittings, and bends.

(2) Inverters - For reliability the circuitry within an inverter assembly was arranged to allow each inverter to power either of two pump motor units in a loop. Inverter No. 1 powered pumps A and B, inverter No. 2 powered pumps B and C, and inverter No. 3 powered pumps C and A.

(3) Coolant Reservoirs - The reservoirs in the coolant system established the base pressure for the coolant system. The system base pressure varied with reservoir coolant volume

and coolant temperature. Each coolant loop contained ten reservoirs. Eight were contained within the four reservoir assemblies in the reservoir module. The other two, mentioned above, were contained in the two pump packages. Figure 2.4-11 shows how the reservoirs were connected into the coolant loops.

Figure 2.4-14 presents typical coolant reservoir characteristics. The initial fill was set to provide pressure greater than 5 psia, to prevent pump cavitation, and less than 47.5 psia so as to not exceed maximum allowable operating pressures for the temperature range of 40°F to 150°F. This fill also provided a good margin on reservoir volume at maximum allowable coolant leakage rates. The plan was to fill each reservoir to a level which corresponds with (neglecting elevation effects on pressure) the upper line of the shaded area on Figure 2.4-14. This was to be accomplished by filling each loop completely at 70°F then removing 2500 cc of coolant. The maximum allowable coolant loss per reservoir during prelaunch and flight is also indicated on the figure.

The actual coolant volume at which both the primary and secondary coolant loops on U-1 were launched was less than specified by Figure 2.4-14 as the minimum prelaunch coolant volume. This condition resulted from breaking into the coolant loops at KSC. The primary loop was reserviced after replacement of coolant lines damaged in shipment of the vehicle to KSC. The secondary loop had a piggyback pressure transducer added at D223.

The lower than planned coolant volumes were considered adequate because the prelaunch data indicated the leakage rate was very low with respect to the design allowable and ample coolant was available to last the entire duration of the mission with both loops leaking at their maximum allowable design rate.

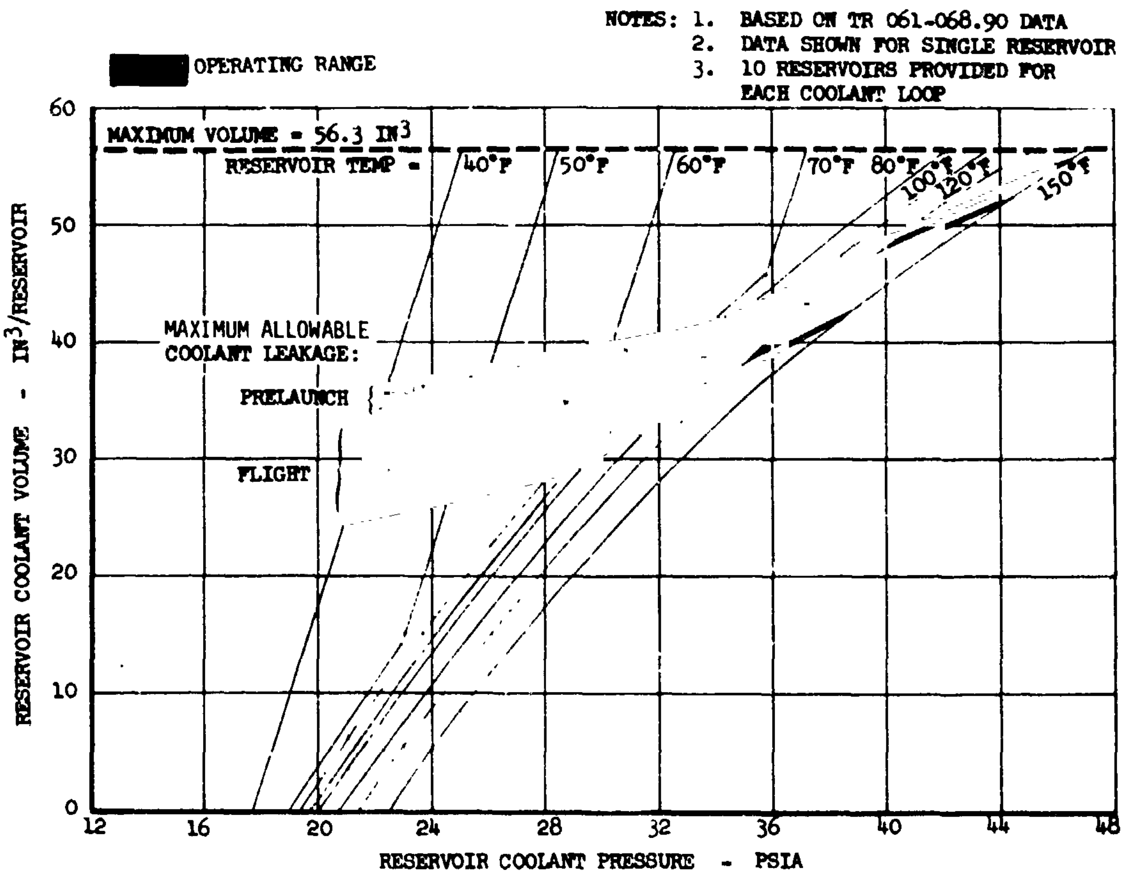


FIGURE 2.4-14 TYPICAL COOLANT RESERVOIR CHARACTERISTICS

Flight coolant mass is discussed in Section 2.4.5.2.

B. Heat Loads - Temperature control of the heat loads was through their heat transfer with the coolant. The coolant system design heat loads presented in Section 2.4.2 may be grouped into the three regions.

(1) Suit Cooling - Suit cooling heat loads were introduced into the coolant system through suit cooling heat exchangers in the suit/battery cooling module. Basically the load was the sum of suit cooling load, the water pump heat load, and heat gain from environment.

(2) ECS - ECS heat loads were added to the coolant system through condensing heat exchangers, cabin heat exchangers for OWS and AM/MDA, three tape recorders, an ATM C&D Panel/EREP heat exchanger, an oxygen heat exchanger, and heat leaks. The portion of the compartment loads leaked from the equipment section of the coolant system (e.g., via coldplates mounted on compartment walls) were entered into the integrated thermal model as equipment heat load.

(3) Equipment - The active coolant system utilized coldplates to control the temperatures of equipment which had small contact areas, high heat dissipation requirements and small allowable temperature ranges. Three tape recorders, two battery modules, six electronics modules, and two coolant pump inverter assemblies were coldplate mounted. Due to their location in the coolant loop, the tape recorder coldplate and the ATM C&D/EREP cooling system heat load were considered to be part of the ECS heat load.

The coldplated equipment and coldplate surface contact areas are listed in Figure 2.4-15. The coldplate design heat removal capability was based on a component base/coolant fluid conductance of 50 Btu/hr-ft^2°F. Conductance was based on use of a heat transfer compound per MDAC-E Process Specification 13618. The heat transfer compound was used on all coldplates except

ITEM	COLDPLATE PART NUMBER	COLDPLATE LOCATION: DRAWING TITLE	COLDPLATE LOCATION: DRAWING NUMBER	EQUIPMENT INSTALLED ON COLDPLATE: QTY	EQUIPMENT INSTALLED ON COLDPLATE: DRAWING NUMBER	EQUIPMENT INSTALLED ON COLDPLATE: DRAWING TITLE	TOTAL CONTACT AREA OF EQUIPMENT ON COLDPLATE IN 2	EQUIPMENT WEIGHT LB/UNIT	HEAT DISSIPATION BTU/HR/UNIT	ALLOWABLE COLDPLATE TEMPERATURE RANGE: MIN (°F)	ALLOWABLE COLDPLATE TEMPERATURE RANGE: MAX (°F)
1	61A830102	BATTERY MODULE	61A820023	1	61V820051	BATTERY	221 0	120.1	170.7	40	74
2	61A830102	BATTERY MODULE	61A820029	1	61V820051	BATTERY					
3	61A830102	BATTERY MODULE	61A820029	1	61V820051	BATTERY					
4	61A830102	BATTERY MODULE	61A820029	1	61V820051	BATTERY					
5	61A830102	BATTERY MODULE	61A820029	1	61V820051	BATTERY					
6	61A830102	BATTERY MODULE	61A820029	1	61V820051	BATTERY					
7	61A830102	BATTERY MODULE	61A820029	1	61V820051	BATTERY					
8	61A830102	BATTERY MODULE	61A820029	1	61V820051	BATTERY	221 0	120.1	170.7	40	74
9	61A830103	BATTERY MODULE	[illegible]	1	61V820052 & 53	CHARGER & REGULATOR	210 0	26.4	124.5	40	100
10	61A830103	BATTERY MODULE	[illegible]	1				FOR EACH	FOR EACH		
11	61A830103	BATTERY MODULE	[illegible]	1				CHARGER	CHARGER		
[illegible]	[illegible]	BATTERY MODULE	[illegible]	1							
[illegible]	[illegible]	BATTERY MODULE	[illegible]	1				13.7	56.9 *		
[illegible]	[illegible]	BATTERY MODULE	[illegible]	[illegible]				FOR EACH	FOR EACH		
[illegible]	[illegible]	BATTERY MODULE	[illegible]	1	61V820052 & 53	CHARGER & REGULATOR	200 0	REGULATOR	REGULATOR	40	100
[illegible]	[illegible]	ELECTRONICS MODULE NO 1	[illegible]	4	61B350	SUIT COMPR PWR SUP	125 0	4.5	45	40	120
[illegible]	[illegible]	ELECTRONICS MODULE NO	[illegible]	[illegible]	[illegible]	CPJU	[illegible]	33.5	31	40	120
[illegible]	[illegible]	ELECTRONICS MODULE NO	[illegible]		[illegible]	VHF TM 10 WATT XMTR	64 0	2.3	191	40	138
				1	52 85712	TELEMETRY 2 WATT XMTR	15 7	2.3	56	40	120
[illegible]	61A830275	ELECTRONICS MODULE NO 3	[illegible]	[illegible]	52 85712	HI LEVEL MULTI	89 0	3.2	0.2	30	120
				2	52 85713	LO LEVEL MULTI	29 6	2.6	0.49	30	120
				1	61V880000	INSTR PKG 2A	94 4	16.6	19.1	36	120
21	[illegible]	ELECTRONICS MODULE NO 3	[illegible]	1	[illegible]	PCM INTERFACE BOX	187 3	62.7	45	36	120
				1	61V880000	INSTR PKG 2A	94 4	15.6	19.1	36	120
22	61A830255	ELECTRONICS MODULE NO 3	61A820035	[illegible]	52 85711	PCM PROGRAMMER	270 0	20.1	21	40	120
23	61A830217	ELECTRONICS MODULE NO 3	61A820035	[illegible]	52 85713	LO LEVEL MULTI	74 0	2.6	0.49	30	120
24	61A830169	ELECTRONICS MODULE NO 4	61A820036	3	61C850037	DC DC CONVERTER	97 35	7 6	158	36	138
25	61A830257	ELECTRONICS MODULE NO 5	61A820042	1	61V880000	INSTR PKG	91 2	16.0	4	40	120
				1	61B850012	C&W UNIT	165 0	49.0	322	40	120
26	61A830293	ELECTRONICS MODULE NO 5	61A820042	1	61C350096	VHF TRANSCEIVER	14 6	13.2	96	40	120
27	61A830139	ELECTRONICS MODULE NO 5	61A820042	1	61C850505	RTTA	5 6	2.5	14	40	120
28	61A830323	ELECTRONICS MODULE NO 5	61A820042	1	61A880000	INSTRUMENTATION PACKAGE	73 5	18.2	10.2	36	120
				1	61B850012	HI LEVEL AUDIO AMP C & W	21 3	12.3	24	40	120
29	61A830324	ELECTRONICS MODULE NO 6	61A820049	1	61C850013	TRACKING LIGHT ELECT PKG	264 0	9.0	201	40	120
30	61A830183	COOLANT PUMP MODULE	61A830050	3	61B850009	COOLANT PUMP PWR SUP	139 5	9.5	47	40	120
31	61A830183	COOLANT PUMP MODULE	61A830050	3	61B850009	COOLANT PUMP PWR SUP	133 [illegible]	9.5	47	40	120
32	61A830116	TAPE RECORDER MODULE	61A800018	1	52 85713	TAPE RECORDER	100 0	13.9	52	36	100
33	61A830116	TAPE RECORDER MODULE	61A800018	1	52 85713	TAPE RECORDER	100 0	13.9	52	36	106
34	61A830116	TAPE RECORDER MODULE	61A800018	1	52 85713	TAPE RECORDER	100 0	13.9	52	36	100

*2500 WATT PCG LOAD, $\beta = 0^{o}$ ORBIT.

FIGURE 2.4–15 COLDPLATE MOUNTED EQUIPMENT

Those for the tape recorders which were adequately cooled without the compound.

All of the coldplated equipment listed in Figure 2.4-15 was qualified for 120°F (or greater) coolant inlet temperatures, except for the tape recorders and the battery module chargers and regulators which were qualified for 100°F operation, and the batteries which were qualified for 74°F continuous operation. PCG cooling is discussed further in Section 2.4.3.4

The coolant loop sequence of coldplates is shown in Figure 2.4-16. In order to reduce the pressure drop, flow was paralleled through the three tape recorder coldplates and through the two streams of a battery module and three electronics modules. All coldplates, except the coolant pump inverter coldplates, contained two coolant passages.

C. Heat Sink - The excess heat removed by the Airlock coolant system was disposed of by the ground cooling system prior to launch, and the radiator/capacitor system while in orbit. During the interim, heat was stored in the thermal capacitor. The transfer from the ground cooling system to the radiator/capacitor system was initiated by the normal DCS command mode. The radiator selector valve was also controllable from ECS control panel 203 shown on Figure 2.4-12 after crew arrival.

(1) Ground Cooling - For ground operations prior to launch, heat was dissipated from the coolant loops to ground coolant equipment through the ground cooling heat exchanger. The system is shown in Figure 2.4-17. The GSE heat exchanger was in parallel with the vehicle radiators, with valving control to perform a switchover to the radiators accomplished by DCS command approximately 10 minutes prior to launch.

The most stringent temperature requirement occurred during pre-lift-off when the wax in the thermal capacitor had to be frozen and maintained through launch. The thermal capacitor coolant inlet temperature as a function of the ground coolant temperature

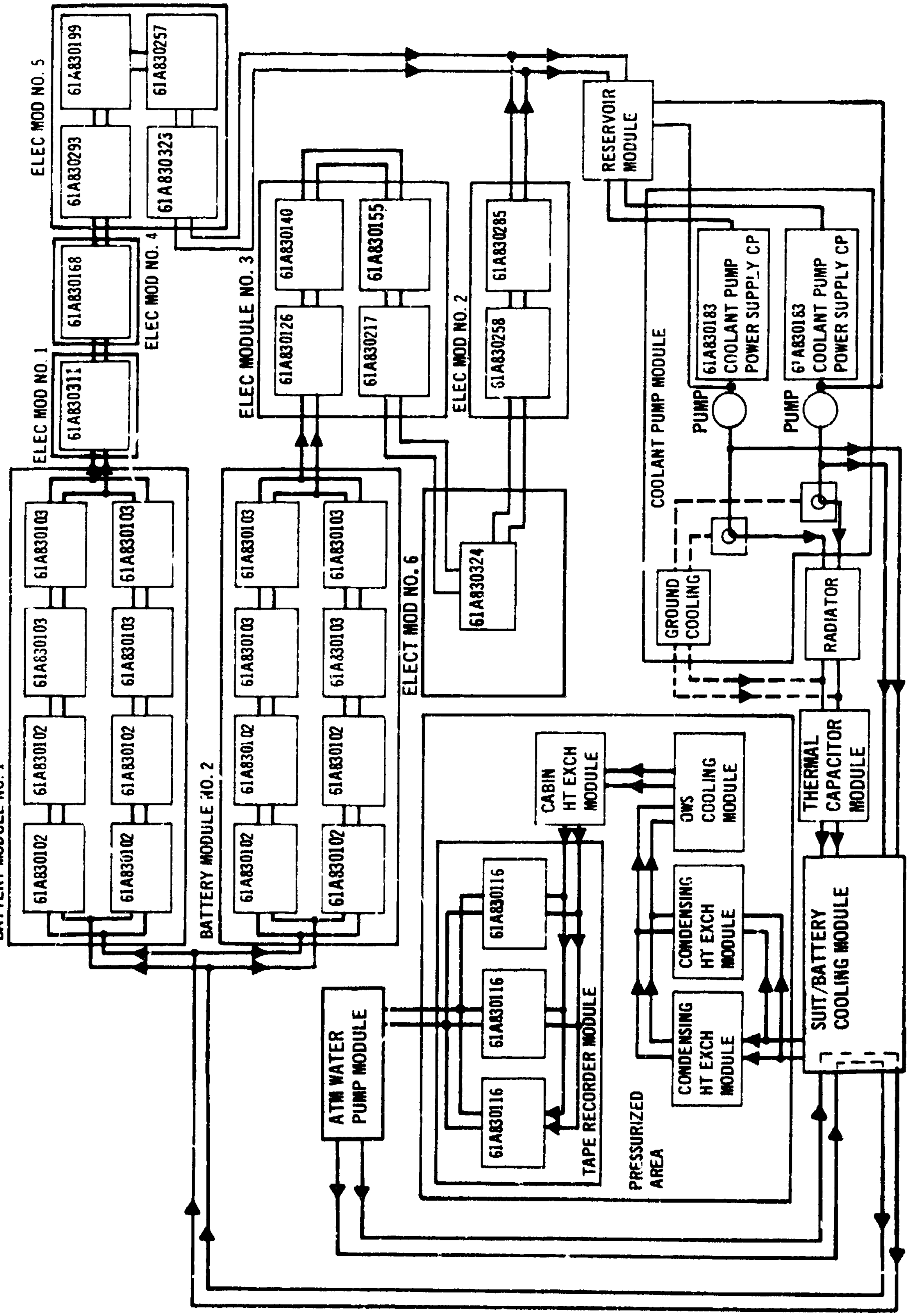


FIGURE 2.4-16 COLDPLATE LOCATIONS

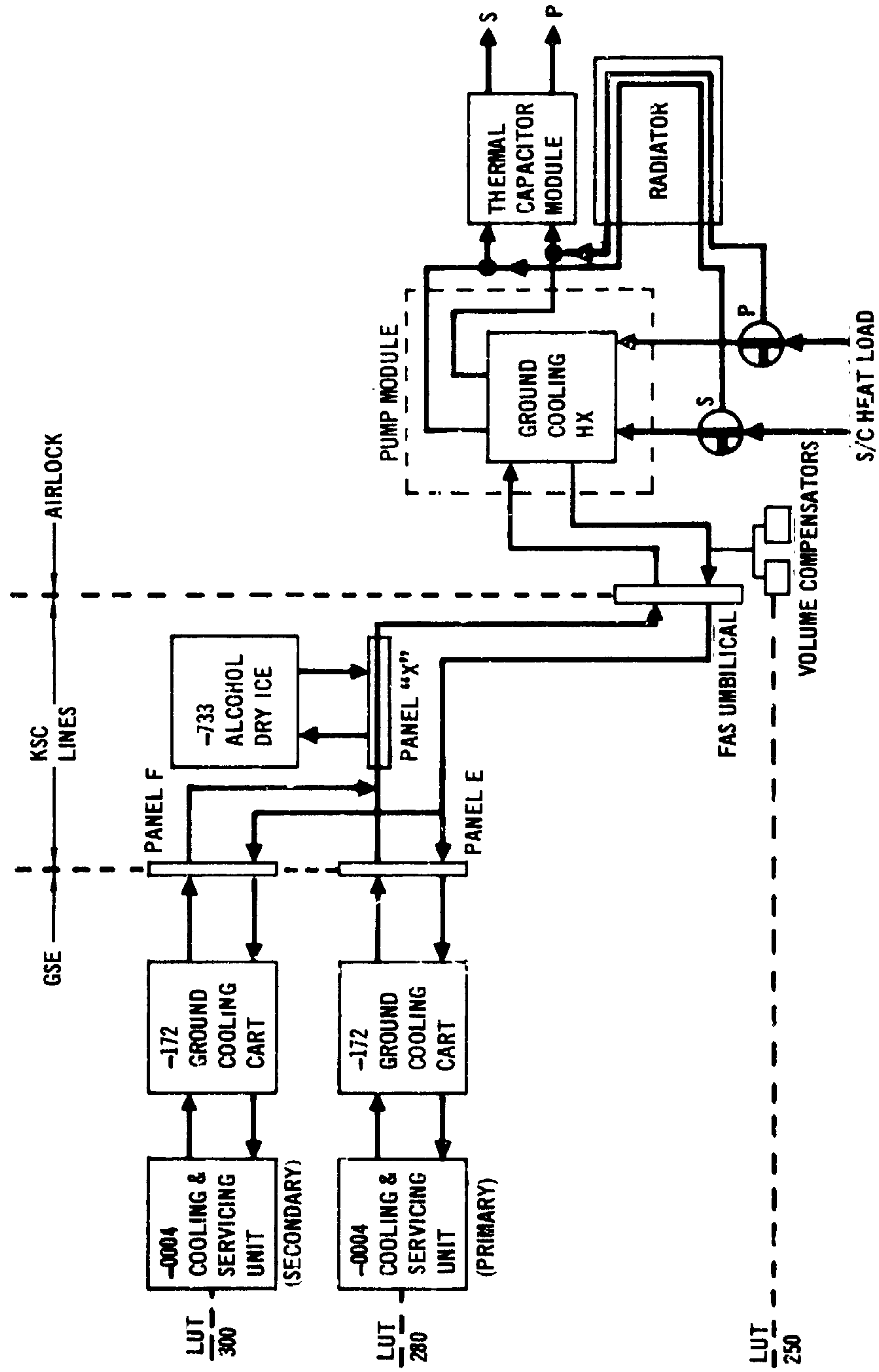


FIGURE 2.4-17 PAD AND VAB GROUND COOLING SYSTEM

supplied to the AM/GSE interface at the FAS is shown in Figure 2.4-18. This data was generated using the pre-lift-off equipment loads plus heat leaks to the coolant lines, ECS, and electrical equipment. The heat leaks were based on data determined during test (SEDR D3-E75), and modified for the expected 80°F effective heat leak environment at the launch site. Figure 2.4-18 shows a capacitor module inlet temperature of approximately 12°F for an AM/GSE interface supply temperature of -15°F. The GSE cooling equipment capacity was sized to meet the requirements, taking into consideration the heat leaks of the launch site interconnecting lines. The prelaunch total spacecraft heat load (equipment plus leaks) was approximately 9000 Btu/hr. No minimum Redline was established since minimum temperatures attainable by ground cooling equipment were well above the TCV qualification minimum temperature of -100°F. The 18°F maximum Redline was enforced by Launch Mission Rule during the time period from 30 minutes before lift off until termination of ground cooling. The rules applied specifically to the thermal capacitor skin temperatures (C262, C263, and C264), the capacitor No. 2 primary coolant inlet temperature (C265), and the capacitor module primary coolant outlet temperature (C244).

Coolant loop analysis was used to determine that satisfactory first orbit temperatures would result if ground cooling and switchover to vehicle radiator occurred at the T-10 minute point. Figure 2.4-19 defines the ground cooling requirements for a launch hold after termination of ground cooling. This data was part of the Redline requirement.

Also during prelaunch operations, the Mission Rules Redlines for the condensing heat exchanger inlet temperatures (C209 and C217) were set at 42.7°F minimum and 51.0°F maximum. This assured that the downstream 47°F TCV in the primary coolant loop was in control.

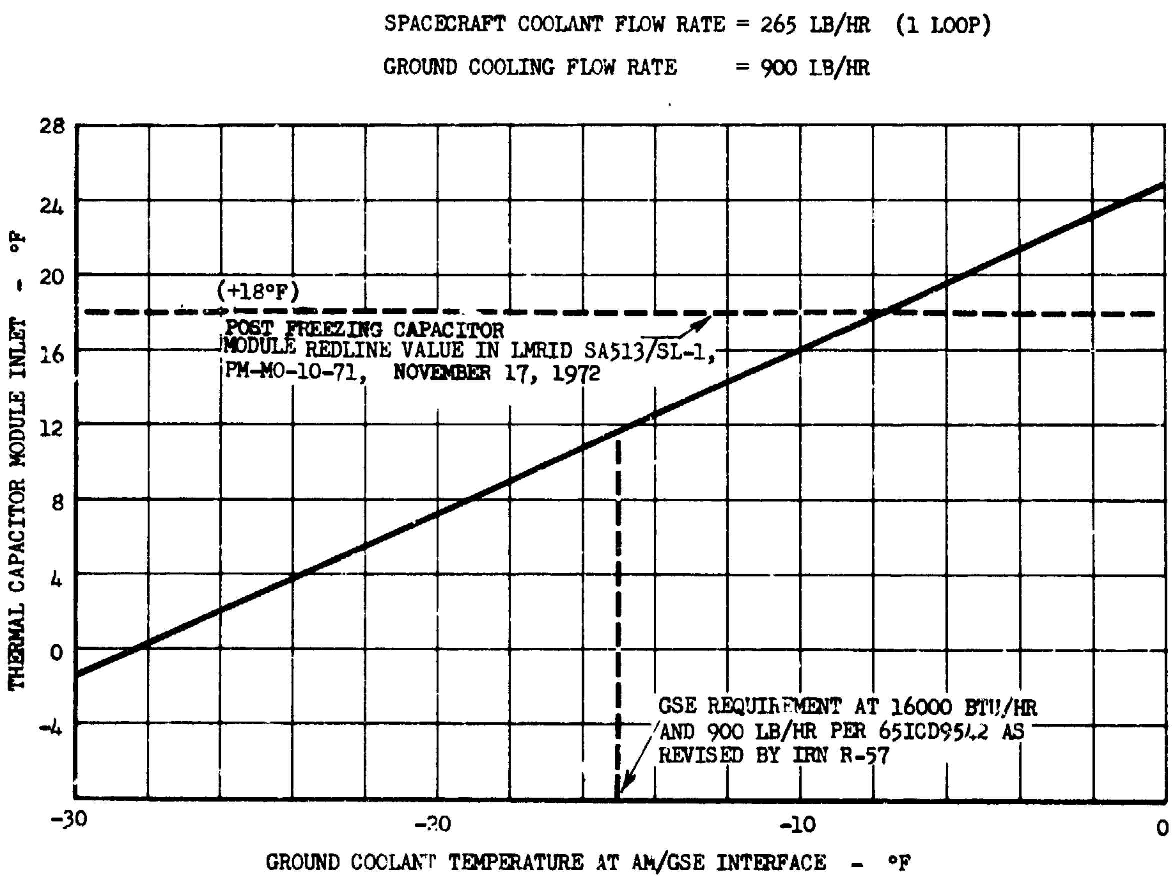
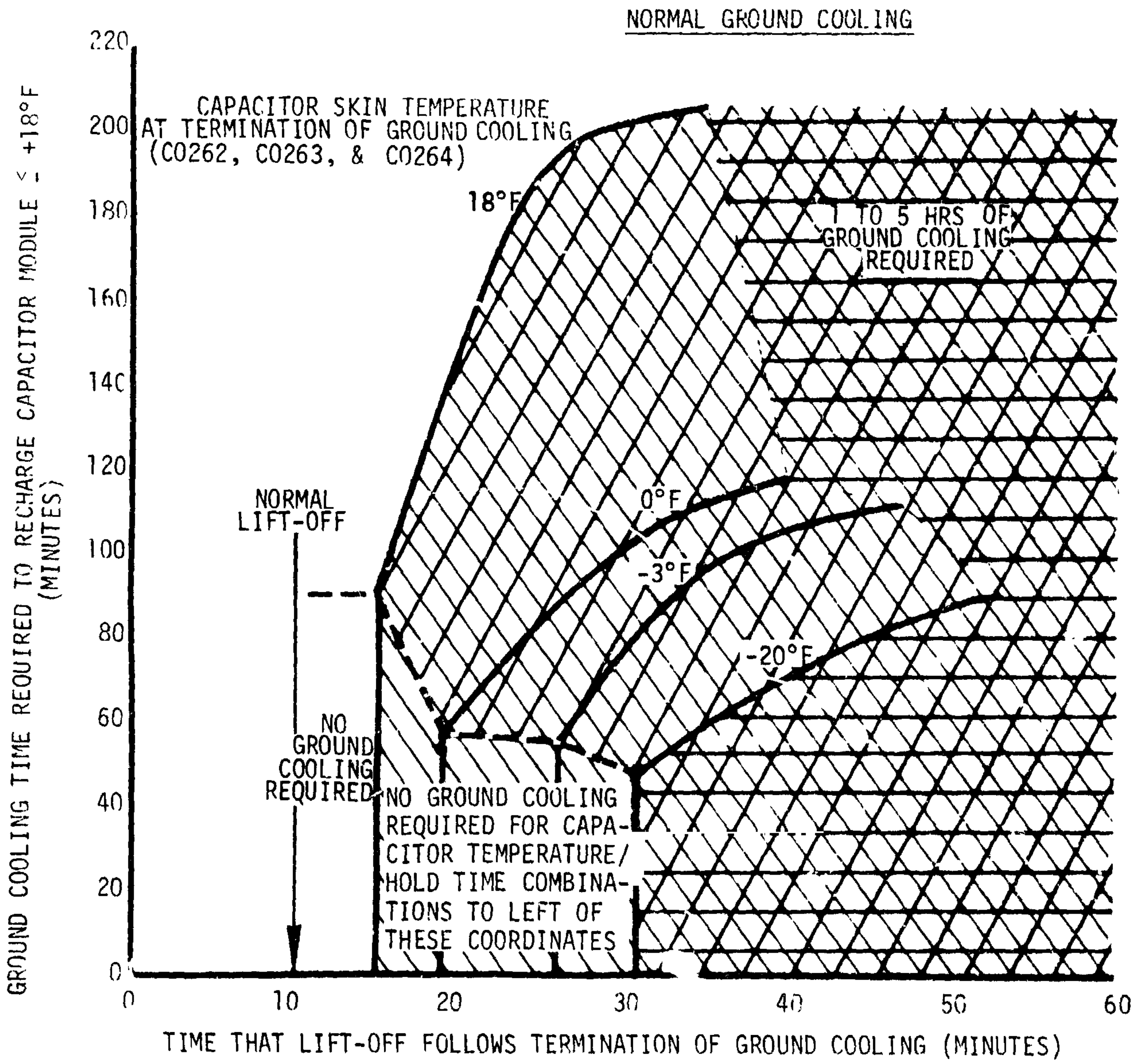


FIGURE 2.4-18 PRE-LIFTOFF COOLING REQUIREMENTS



NOTES:
1) NOMINAL GROUND COOLING TERMINATION AT T-10 MIN.
2) RECHARGING ASSUMES GROUND COOLANT SUPPLIED AT TEMP. WHICH PRODUCED CAPACITOR MODULE TEMP. AT T-10 MIN.
3) ANALYSIS BASED ON NOMINAL PRELAUNCH EQUIPMENT OPERATION.

FIGURE 2.4-19 GROUND COOLING REQUIREMENTS FOR A HOLD AFTER TERMINATION OF NORMAL GROUND COOLING

Two volume compensators were provided in the flight portion of the ground cooling system for redundant protection against overpressurization due to thermal expansion of the coolant trapped by prelaunch disconnection of the ground cooling umbilicals. With a maximum interface pressure of 111 psia, the expansion volume provided in one compensator was sufficient to accommodate expansion of the coolant in the full system, plus coolant which could fill the second compensator should it fail. Pressure at a guage in the operating coolant and servicing unit after T-10 minutes was limited to 80 psia by Mission Rule to ensure a maximum pressure at the compensator below 100 psia. The performance characteristics of one of the compensators is shown in Figure 2.4-20.

(2) Radiator/Capacitor - The coolant system radiator rejected the waste heat to space while in orbit. The capacitor module supplemented the Airlock space radiator during high heat load periods associated with EVA/IVA and EREP mission operations in orbit and provided cooling during the initial launch phase prior to radiator cooldown.

Specification radiator/capacitor system total heat rejection requirements were 16,000 Btu/hr for solar inertial, non-EVA, one pump per loop, two loop operation without CMG desaturation and 12,000 Btu/hr for solar inertial, 4200 Btu/hr maximum EVA, one pump per loop, two loop operation without CMG desaturation.

Radiator/capacitor performance versus load criteria was considered to be satisfactory if the performance met the load based on operation of one pump per loop, two loops operational. The radiator/capacitor provided cooling at a reduced level with one pump, one loop operation. Operation of more than two pumps was not planned.

Radiator/capacitor capacity for solar inertial attitude, with EVA operations and normal operations, including the effects of experiment view factor blockage (see Figure 2.4-1) are shown in Figure 2.4-21.

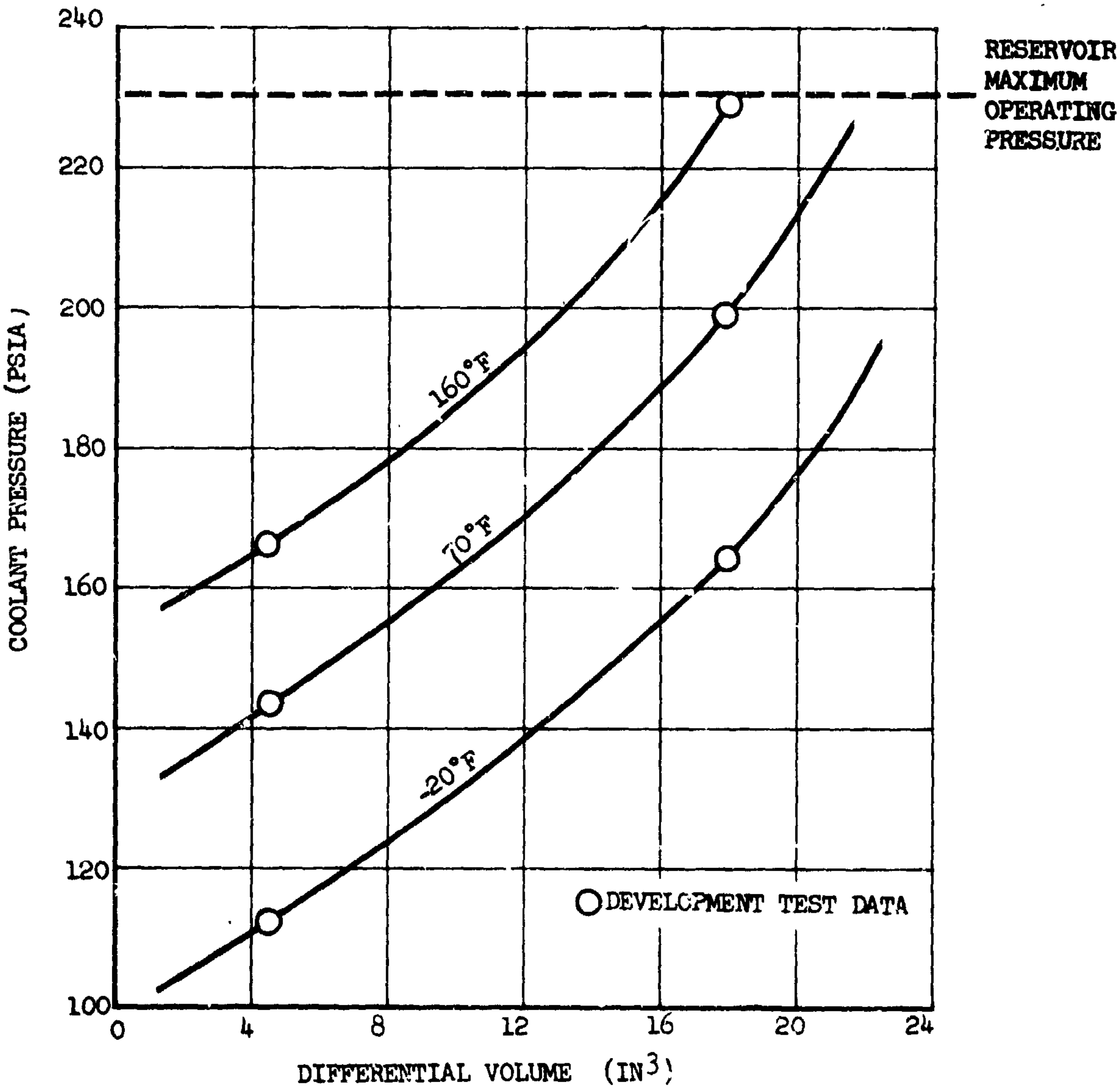


FIGURE 2.4-20 GROUND COOLING SYSTEM COOLANT VOLUME COMPENSATOR CHARACTERISTICS CURVES

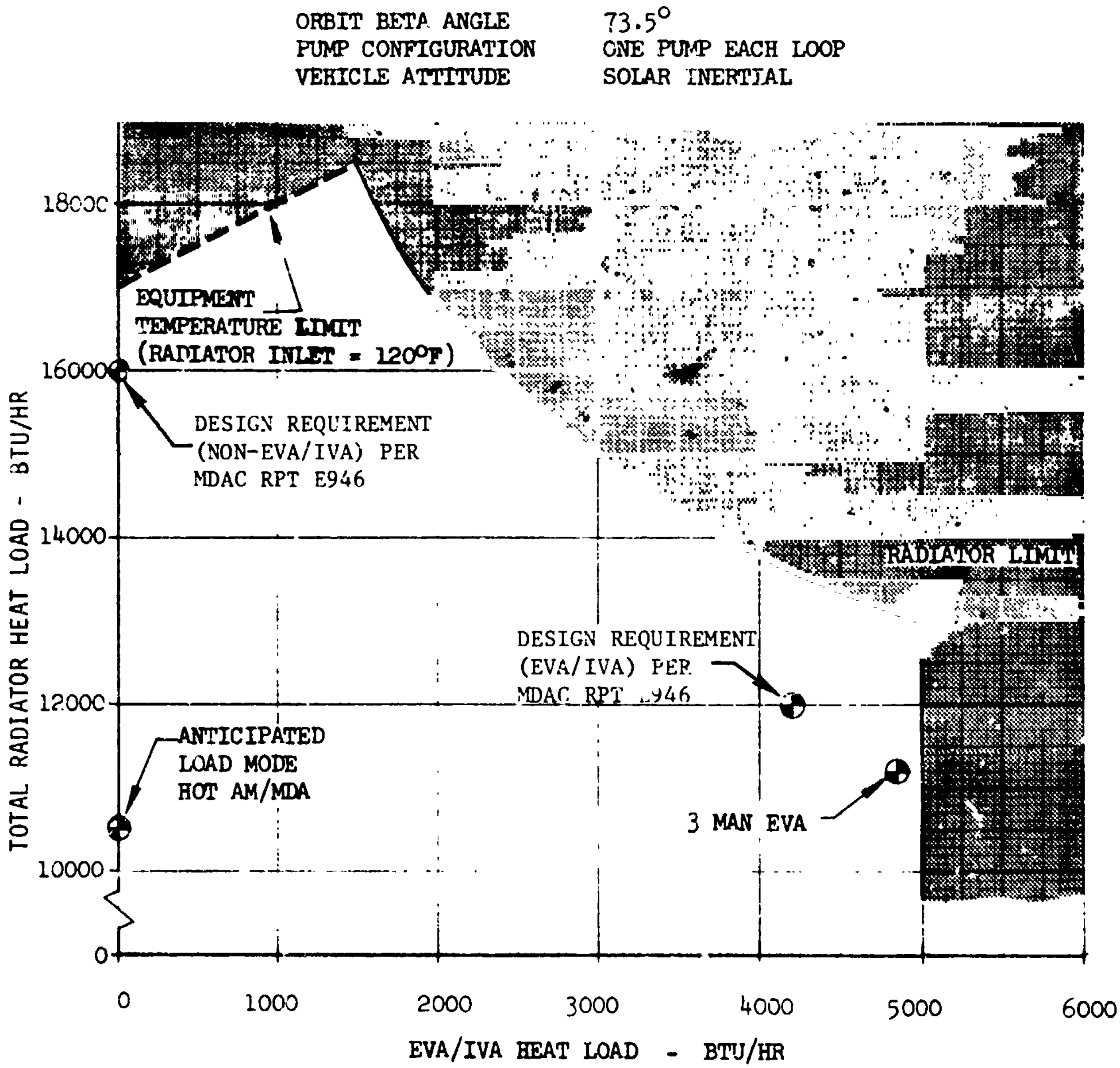


FIGURE 2.4-21 RADIATOR CAPACITY

Figures 2.4-22 and 2.4-23 show the radiator/capacitor capacity available during EREP. For comparison purposes, anticipated heat loads are shown in Figures 2.4-21, -22, and -23. Consequently, both anticipated and specification requirements could be met with the system designed.

- Radiator - To provide the attitude flexibility of a series radiator with reduced pressure drop two parallel flow paths, or files, were used. The radiator consisted of eleven panels, shown in Figure 2.4-24. Four panels were mounted on quarters of the STS between Stations 152.75 and 200, four panels were mounted on quarters of the lower MDA between Stations 200 and 280.57, and three panels were mounted on the upper MDA between Stations 280.57 and 364.10.

  Each STS panel consisted of a 0.050 inch thick magnesium skin, onto which was welded four patterns or files of magnesium tee extrusions. The bulb of the tee was hollow, forming the coolant passage. Two files per panel formed the primary coolant passage and the other two formed the secondary coolant passage. The MDA panel configurations were similar to the STS panels, except that 0.032 inch thick skins were used. The panel skins were bolted to fiberglass stringers which were riveted to the pressure wall. Spiral turbulators (42 total) were installed in both files of the primary and secondary crossover lines between all STS and MDA radiator panels, except for STS panel crossovers between -Z and -Y, and +Z and +Y. In these areas, turbulators were installed in only one of the two files. The eleven radiator panels had a total surface area of 432 square feet.
- Capacitor - The thermal capacitor module located on AM truss No. 3 contained two thermal capacitors within an insulated enclosure to provide a phase change heat sink (fusion temperature $\approx$ 22°F). During orbit operation, the

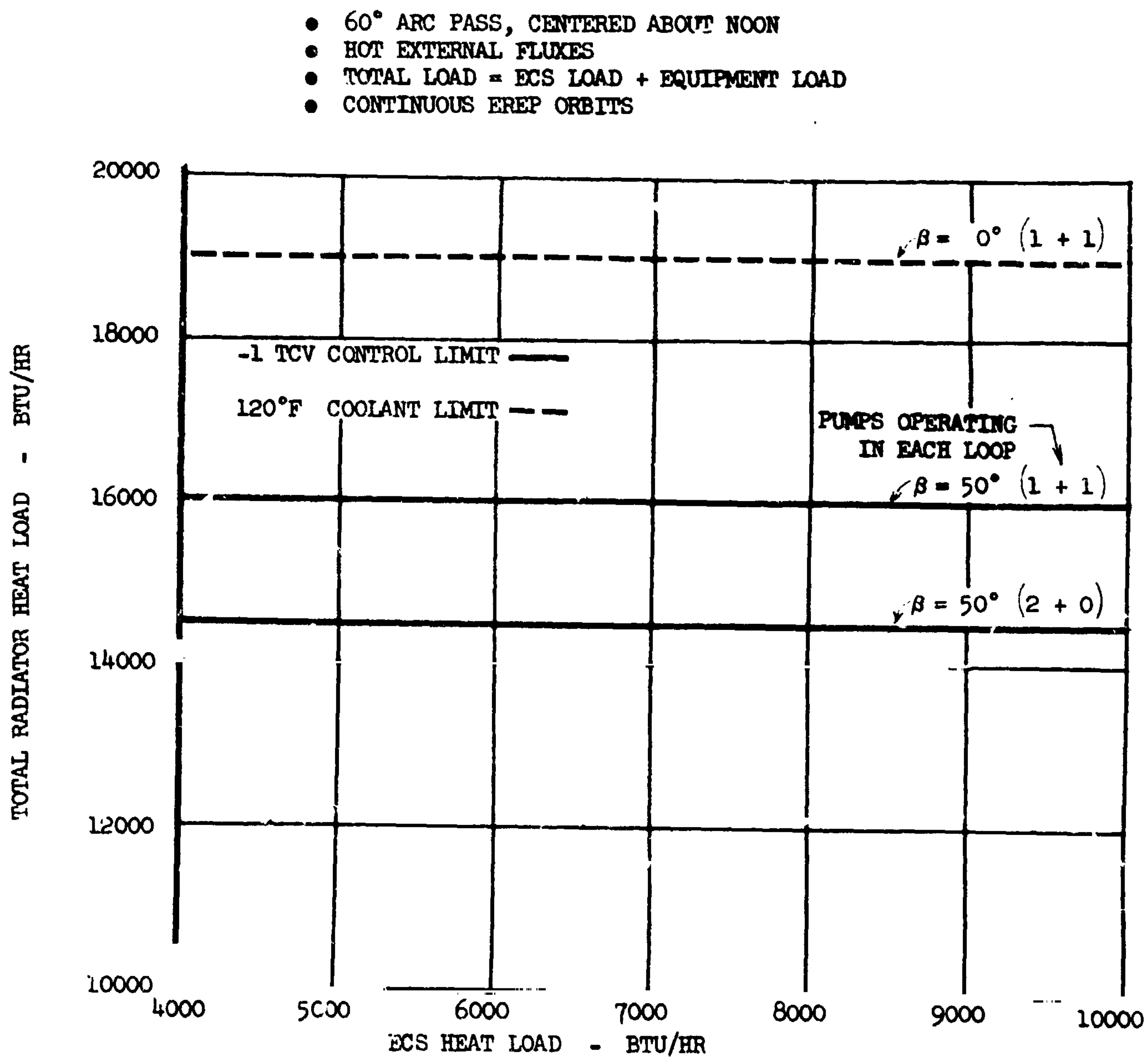


FIGURE 2.4-22 RADIATOR PERFORMANCE FOR EREP MANEUVERS (60° ARC PASS)

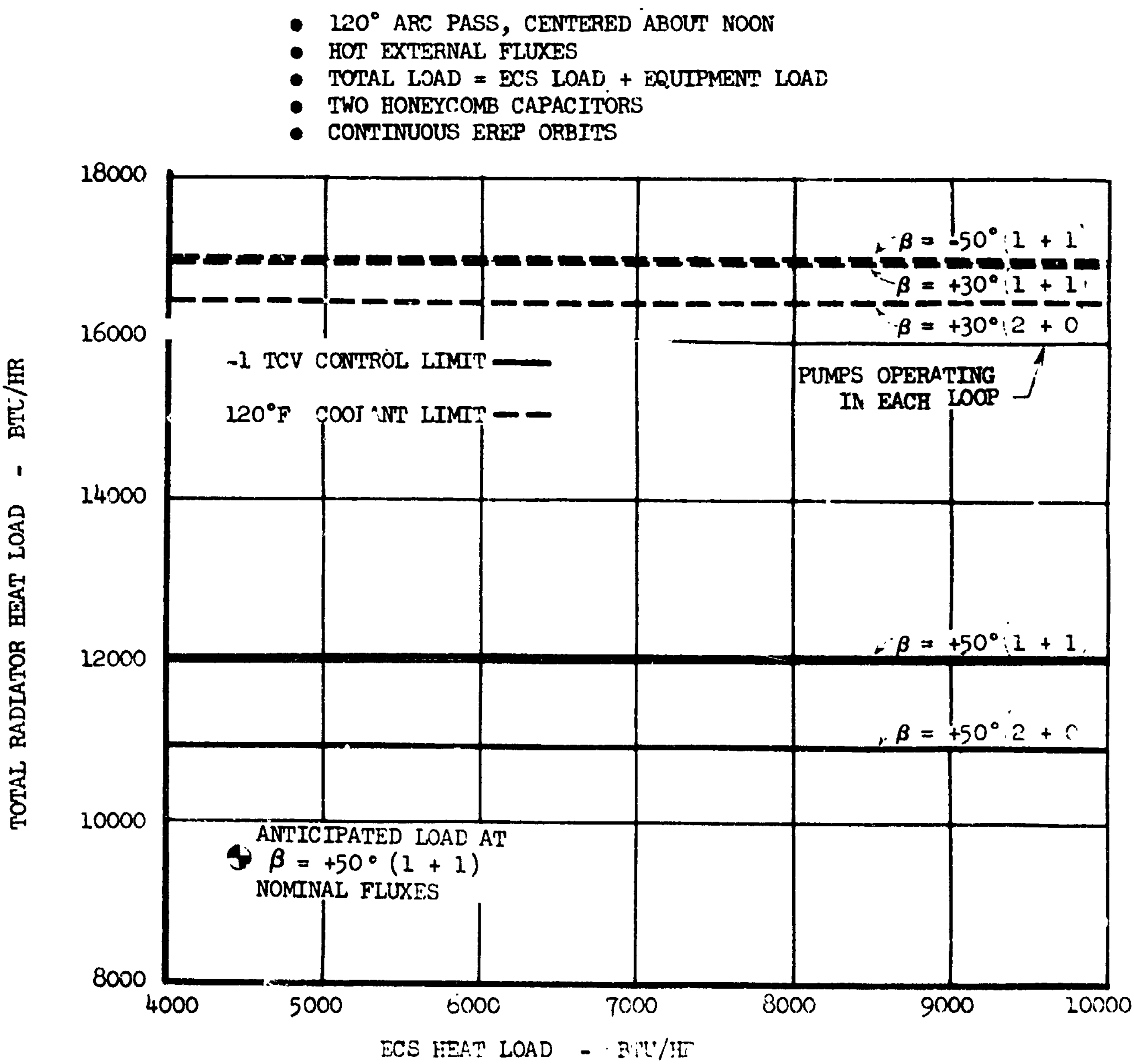


FIGURE 2.4–23 RADIATOR PERFORMANCE FOR EREP MANEUVERS (120° ARC PASS)

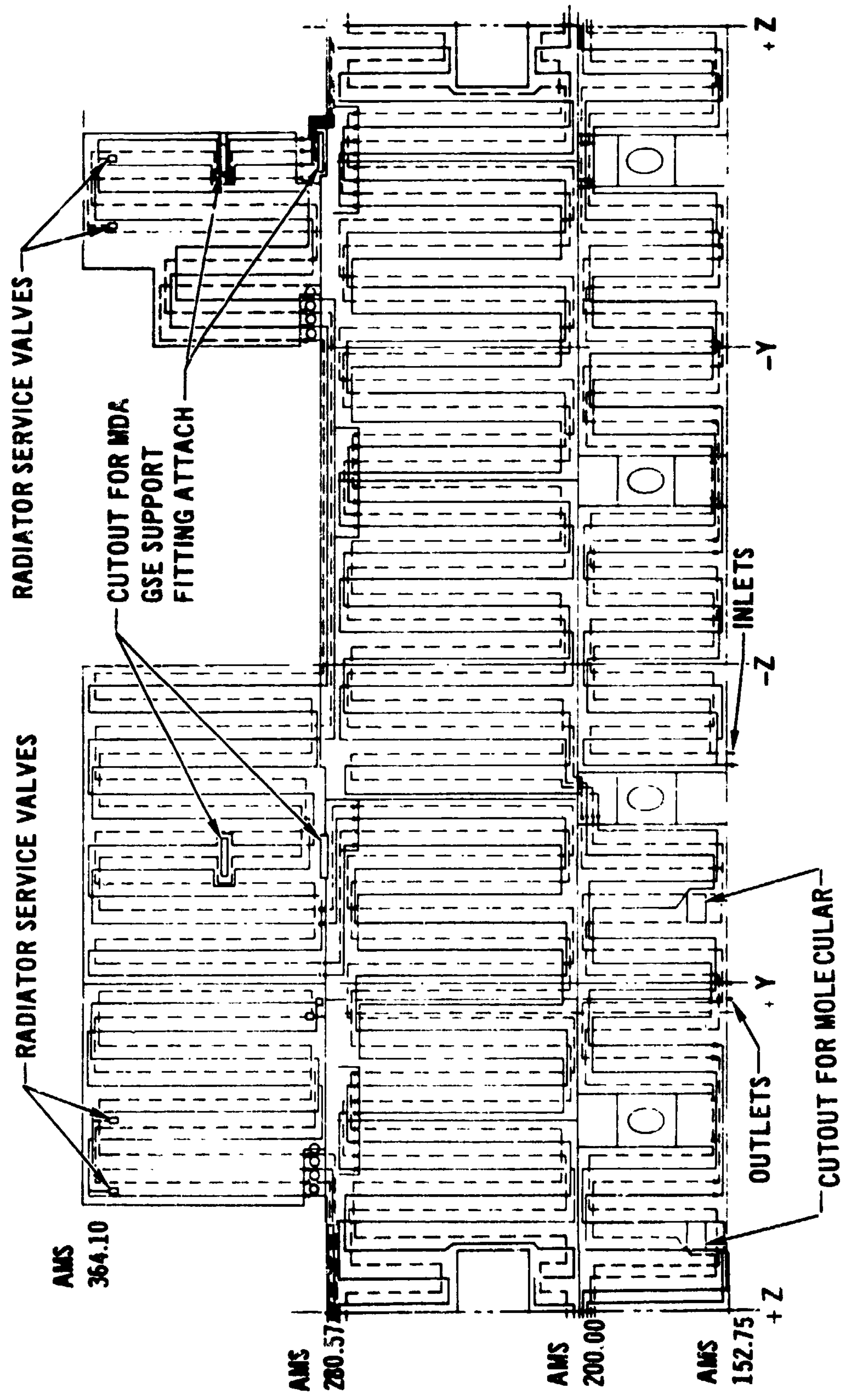


FIGURE 2.4-24 RADIATOR STRETCHOUT – LOOKING OUTBOARD

capacitors were designed to be normally frozen or "charged" on the cold side of the orbit corresponding to the cyclic external heating environment. They were designed to be melted or "discharged" on the hot side of the orbit to a degree depending upon how much the system heat load (internal and external) caused the radiator outlet coolant temperature to exceed the 22°F tridecane wax melt temperature. Each capacitor consisted of two segments. Each segment consisted of a dual passage coldplate (one passage for primary and the other for secondary coolant) bonded between two 1-1/4 x 12 x 18 inches wax chambers, each containing approximately 5 lbs of tridecane wax in an isolated 1/8 inch cell honeycomb matrix. Figure 2.4-25 shows some of the construction features of the capacitor module and a schematic of the coolant flow routing in the capacitor module.

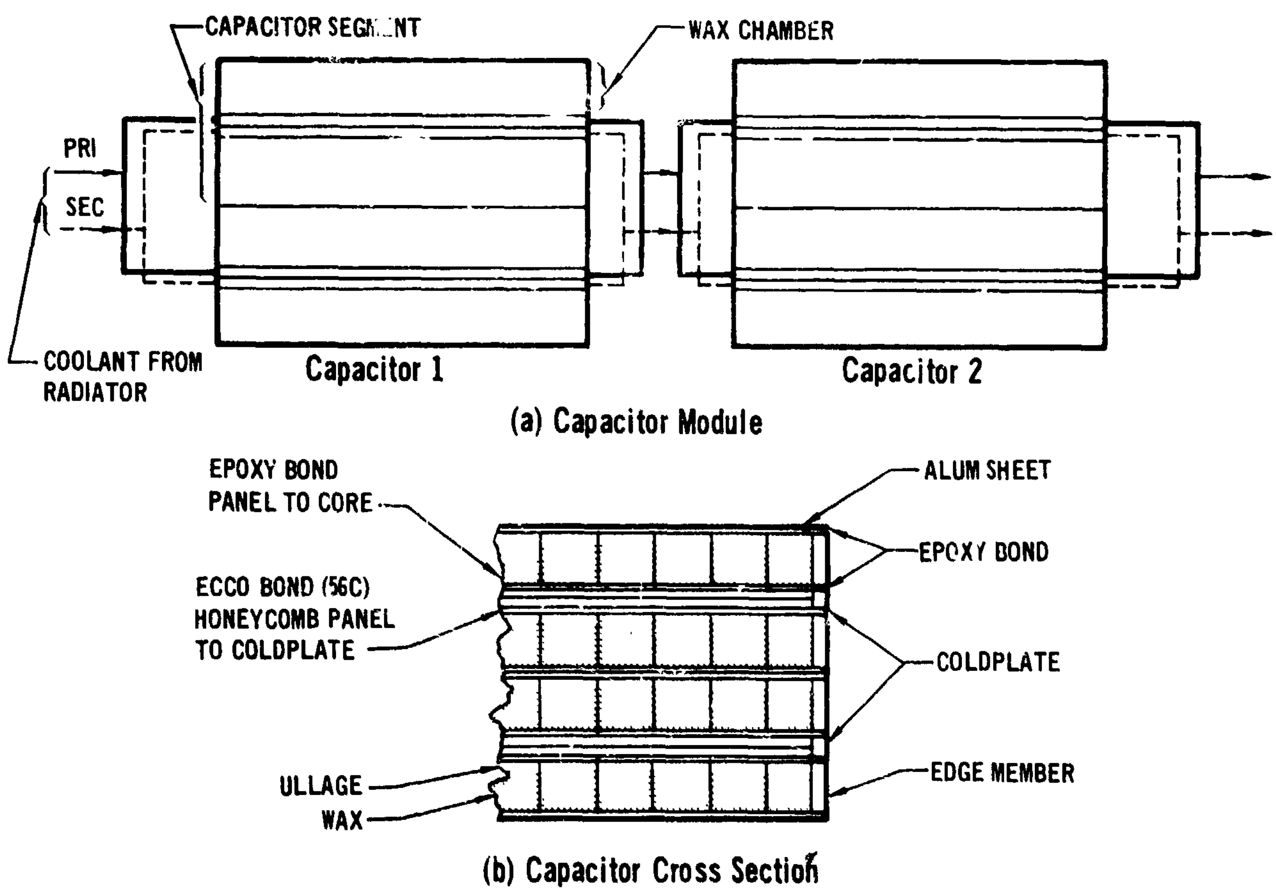


FIGURE 2.4-25 THERMAL CAPACITOR

D. Temperature Control - Temperatures in each coolant loop were controlled by three temperature control valves (TCV). The coolant system temperature control areas were ranked according to their temperature level priority, determining the manner in which the control valves were connected to form the system. In order of priority, these areas were the EVA/IVA, ECS, and equipment. The cooling system supported heat loads from these areas, distributed per the performance limits shown in Figure 2.4-26. Below an EVA/IVA heat load of 1500 Btu/hr, the ECS and equipment heat loads were limited by the 120°F maximum equipment coolant outlet temperature.

(1) Temperature Control Valves - In each of the coolant loops there were two temperature control valves (TCV) with 47°F nominal control points and one TCV with a 40°F nominal control point (refer to Figure 2.4-11). Each valve proportioned coolant flow from its hot and cold inlets to provide a coolant outlet temperature within an operating band about the nominal control point. The control range for the 47°F TCV was ±2°F, and for the 40°F TCV was $^{+2°}_{-4°}$ F. As shown in the figure, one 47°F TCV (designated the upstream valve) supplied coolant to the hot inlet of the other TCV (designated the downstream valve). The downstream TCV delivered coolant to the ECS section of the coolant loop. This valve was the primary TCV and

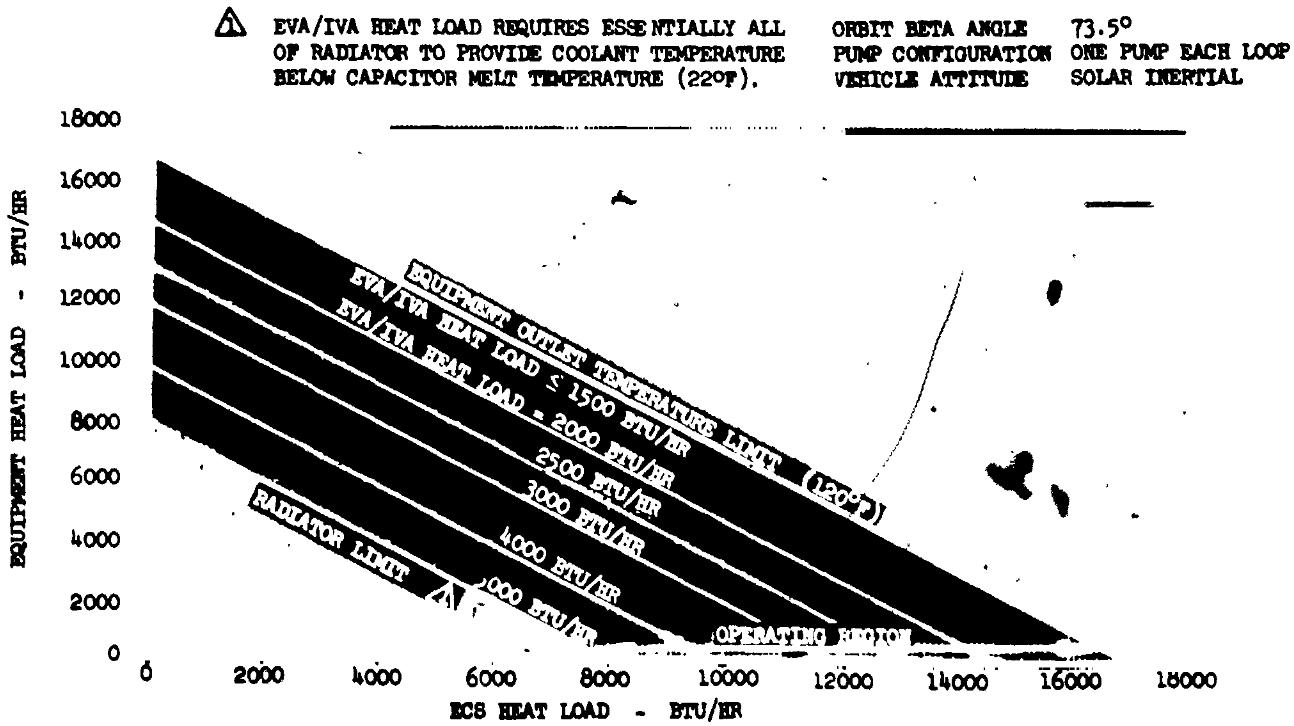


FIGURE 2.4-26 COOLANT SYSTEM PERFORMANCE

maintained control automatically for all heat heat loads within the system capacity limits shown in Figure 2.4-26. The 40°F TCV supplied coolant to the equipment section of the coolant loop. This valve was in control, except for combinations of high EVA/ECS heat loads and warm capacitor module outlet coolant conditions. For these conditions high coolant flow was required by the downstream 47°F TCV cold inlet; the flow from the upstream 47°F TCV was proportionally reduced. Since coolant flow to the upstream 47°F TCV cold inlet was the source of supplemental cooling for the 40°F TCV, the reduced flow allowed the 40°F TCV to go out-of-control on the high side. The upstream 47°F TCV thus served as an intermediary, regulating the cooling available at the battery cooling heat exchanger after the demands of the downstream 47°F TCV were met.

The downstream 47°F TCV outlet temperatures in both loops were monitored by the caution and warning system to warn of condensing heat exchanger inlet temperature below 38 $\pm$1.75°F. No high temperature caution and warning was provided for this parameter since a double failure (both upstream and downstream 47°F TCV's) would have to occur.

(2) EVA/IVA Heat Exchanger Coolant Flow Valves - In each coolant loop a flow selector valve was provided to bypass coolant around heat exchangers used for suit cooling when water solution was not being circulated in the suit cooling loop. These valves are discussed in Section 2.6.2.2.

E. In-flight Coolant Loop Reservicing - The coolant reservicing equipment provided the ability to top off both coolant loops with Coolanol 15 to replenish coolant lost through leakage. The hardware used for reservicing is shown on Figure 2.4-27. The basic method involved pressurizing the coolant supply tank with 35 psig N_2, and forcing coolant into the loop through a line-piercing saddle valve.

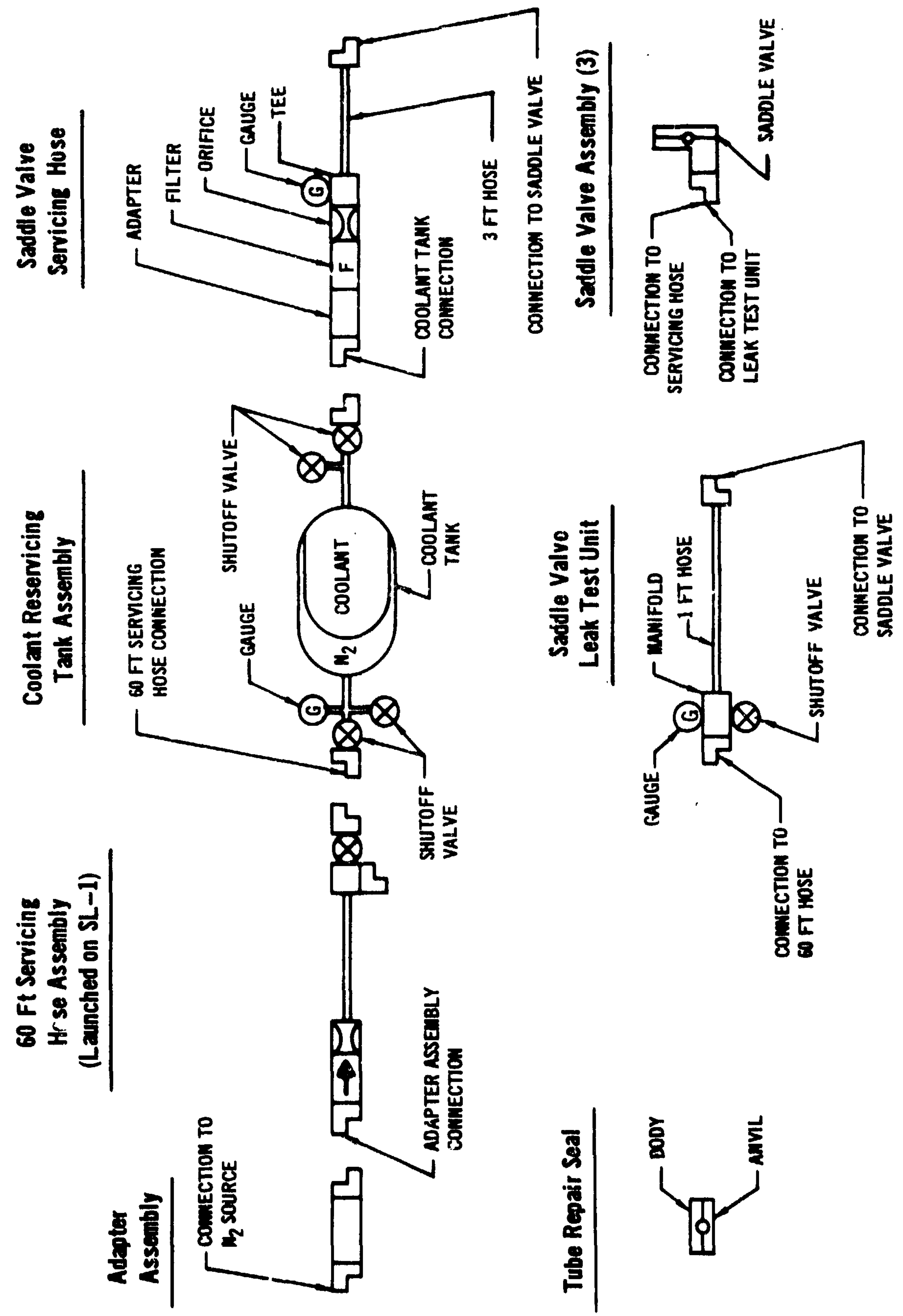


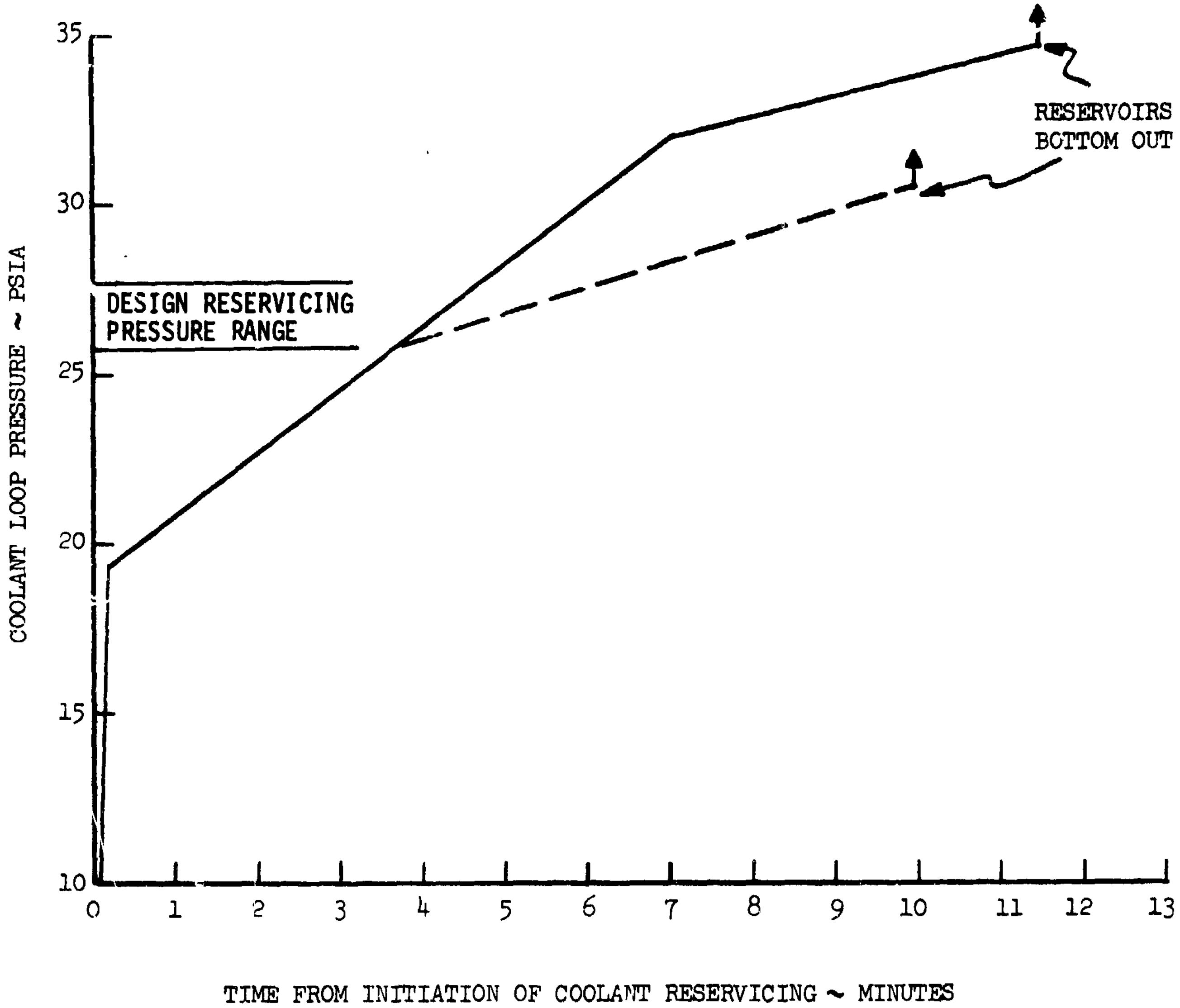
FIGURE 2.4-27 SL-4 COOLANT RESERVICING

Hardware and procedures were developed during the SL-3 mission for reservicing the coolant loops due to the gradual loss of coolant from both loops. The reservicing hardware (except for the 60 ft hose assembly) and procedures were launched with SL-4, and the primary coolant loop was reserviced during the SL-4 mission. Primary loop reservicing is discussed further in Section 2.4.5.

The supply tank was designed to be launched on SL-4 with 42 lbs of coolant and 180 in^3 of pressurized nitrogen. The tank was to be maintained at a positive pressure prior to and during launch by initially aerating the coolant to a dissolved gas content of 340 ppm by weight and pressurizing the tank to 26.7 psia for launch.

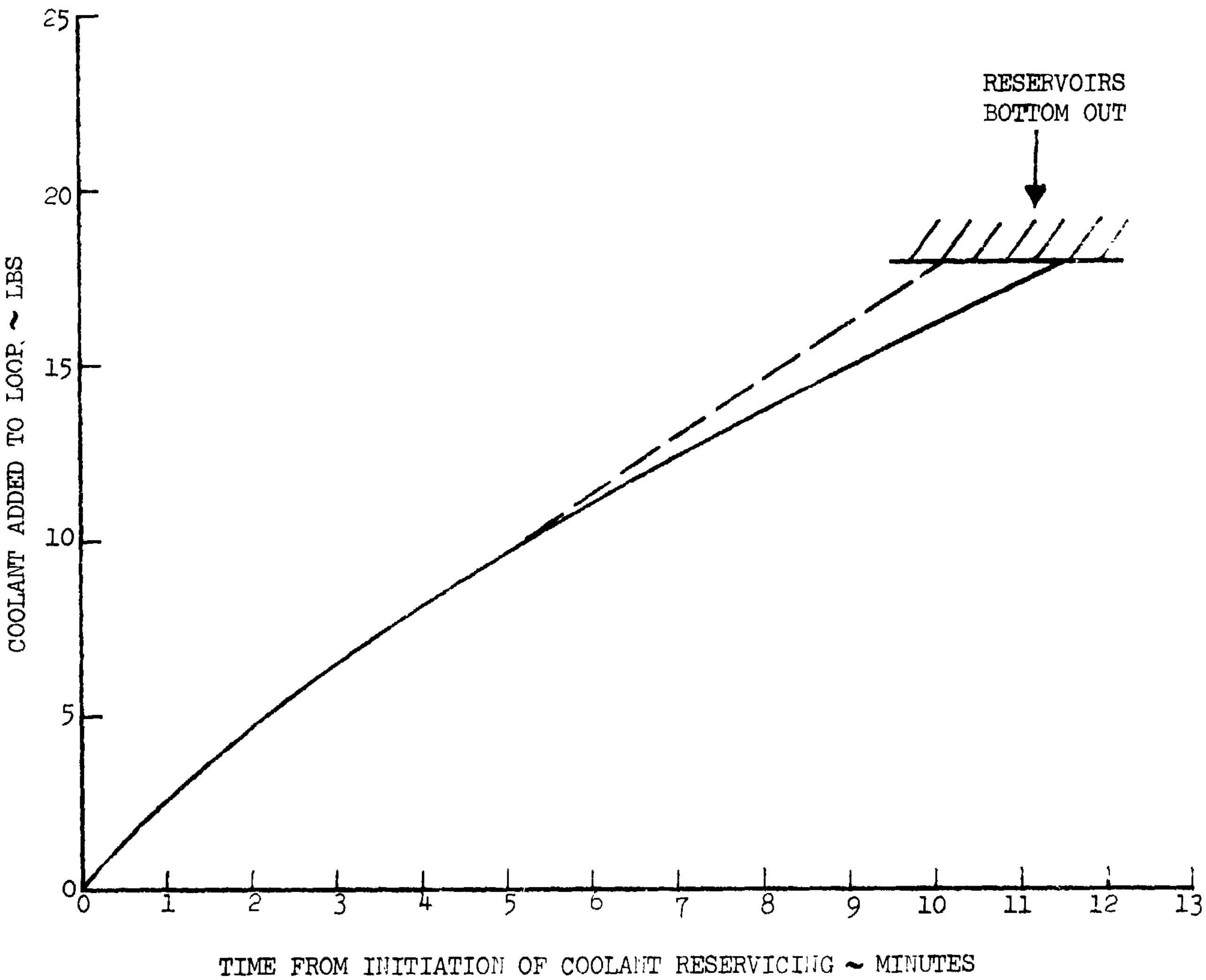
Coolant servicing was planned to be accomplished by: (1) attaching the saddle valve to the coolant line; (2) performing two leakage checks on the installed saddle valve, one with N_2 gas and the other with Coolanol; (3) piercing the coolant line by turning the saddle valve stem until it bottomed on saddle valve body, then retracting to stop; (4) opening coolant supply valve on the reservicing tank to establish flow into the loop; (5) closing the flow valve when the desired pressure level (21 to 23 psig) was indicated by the gage on the 3 ft servicing hose; (6) turning saddle valve until it bottomed again on saddle valve body; and (7) disconnecting the servicing hoses.

Figures 2.4-28 and 2.4-29 show design values of loop pressure increase and coolant addition during reservicing. The actual coolant mass added is discussed in Section 2.4.5.2(E). The combined resistances of the 3 ft servicing valve orifice and the saddle valve resulted in an approximate flowrate of 2 lb/min.



- INITIAL FREE GAS VOLUME = 15 IN^3
- NO FREE GAS DISSOLUTION DURING RESERVICING
- ΔP (SADDLE VALVE TO RESERVOIR MODULE) = 13.6 PSID AT 270 LB/HR
- RESERVICING WITH SN-8 SADDLE VALVE - MAX FLOW
- PRIOR TO RESERVICING, LOOP PRESSURE = 3.5 PSIA

FIGURE 2.4-28 COOLANT RESERVICING PRESSURE CHARACTERISTICS



——— RESERVOIR TEMP 65°F

— — — RESERVOIR TEMP 55°F

- INITIAL FREE GAS VOLUME = 15 IN^3
- NO FREE GAS DISSOLUTION DURING RESERVICING
- ΔP (SADDLE VALVE TO RESERVOIR MODULE) = 13.6 PSID AT 270 LB/HR
- RESERVICING WITH SN-8 SADDLE VALVE - MAX FLOW
- PRIOR TO RESERVICING, LOOP PRESSURE = 3.5 PSIA

FIGURE 2.4-29 COOLANT RESERVICING MASS CHARACTERISTICS

2.4.3.3 ATM C&D Panel/EREP Cooling

The ATM C&D panel/EREP cooling system provided temperature control of the ATM C&D panel and EREP equipment by cooling and circulating water solution through the MDA/STS interface. A schematic of the system is shown in Figure 2.4-30. ICD requirements included the supply of water to the MDA/STS interface at a temperature of 49°F to 78°F. Maximum allowable heat addition from the MDA panels was 1335 Btu/hr, based upon the 78°F water delivery temperature. The total allowable heat load of 1437 Btu/hr transferred to the coolant system, with 78°F water delivery temperature, consisted of 1335 Btu/hr from the MDA panels and 102 Btu/hr from the water pumps. The 78°F water solution temperature was predicted based on an 83°F maximum OWS atmospheric temperature. Design electrical load profiles for transient EREP operation are shown in Figures 2.4-31 and 2.4-32.

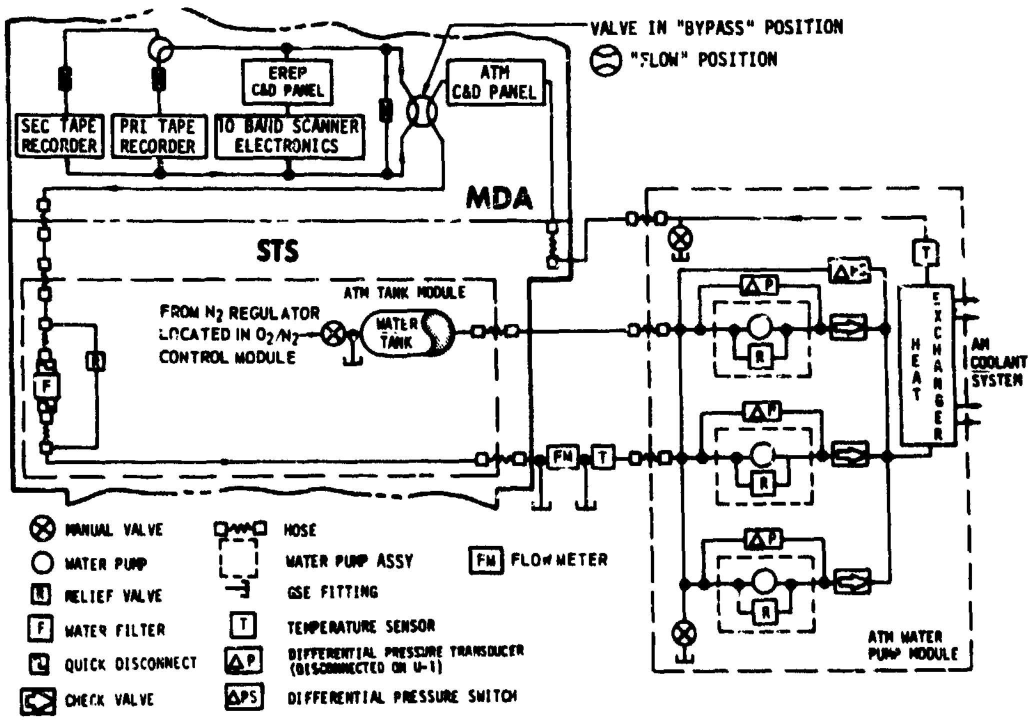


FIGURE 2.4-30 ATM C&D PANEL/EREP COOLING SYSTEM

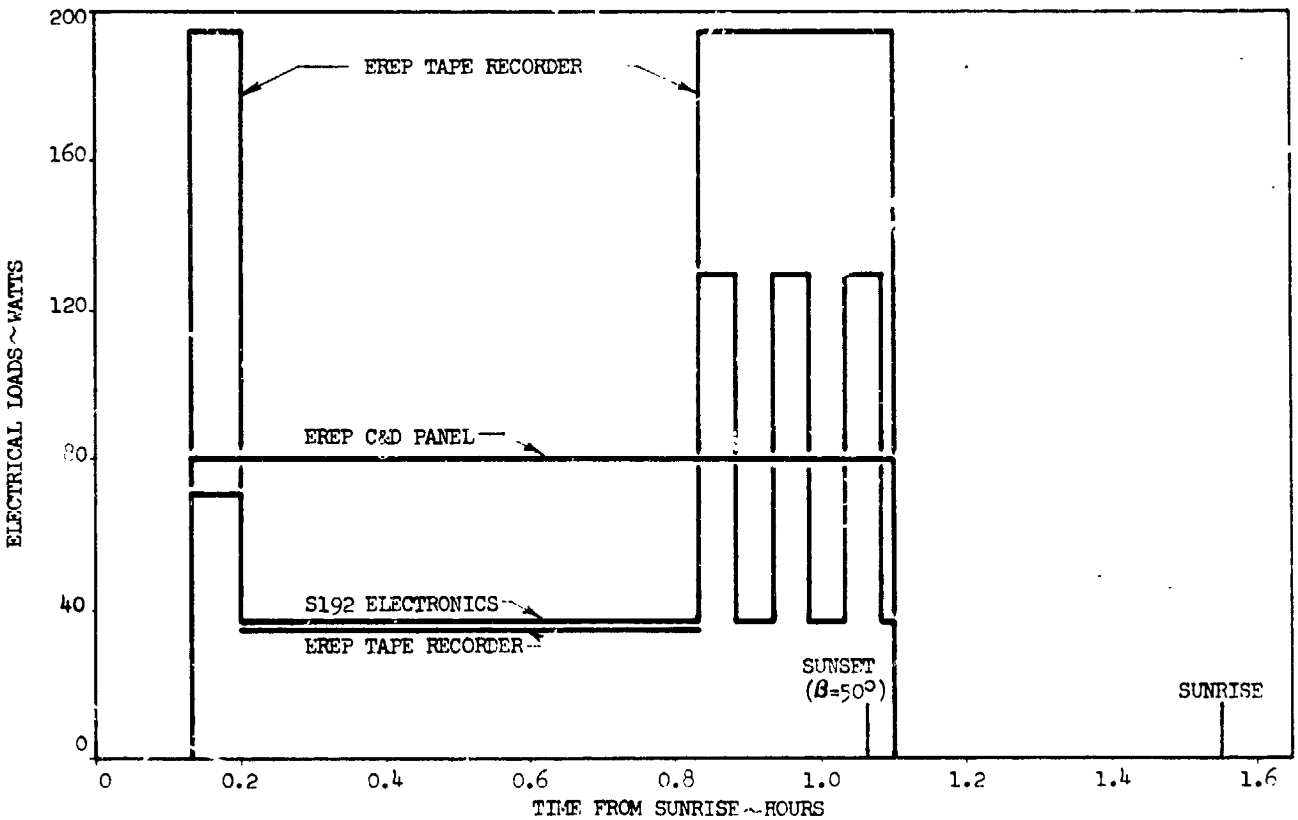


FIGURE 2.4-31 EREP ELECTRICAL LOADS (60° ARC PASS)

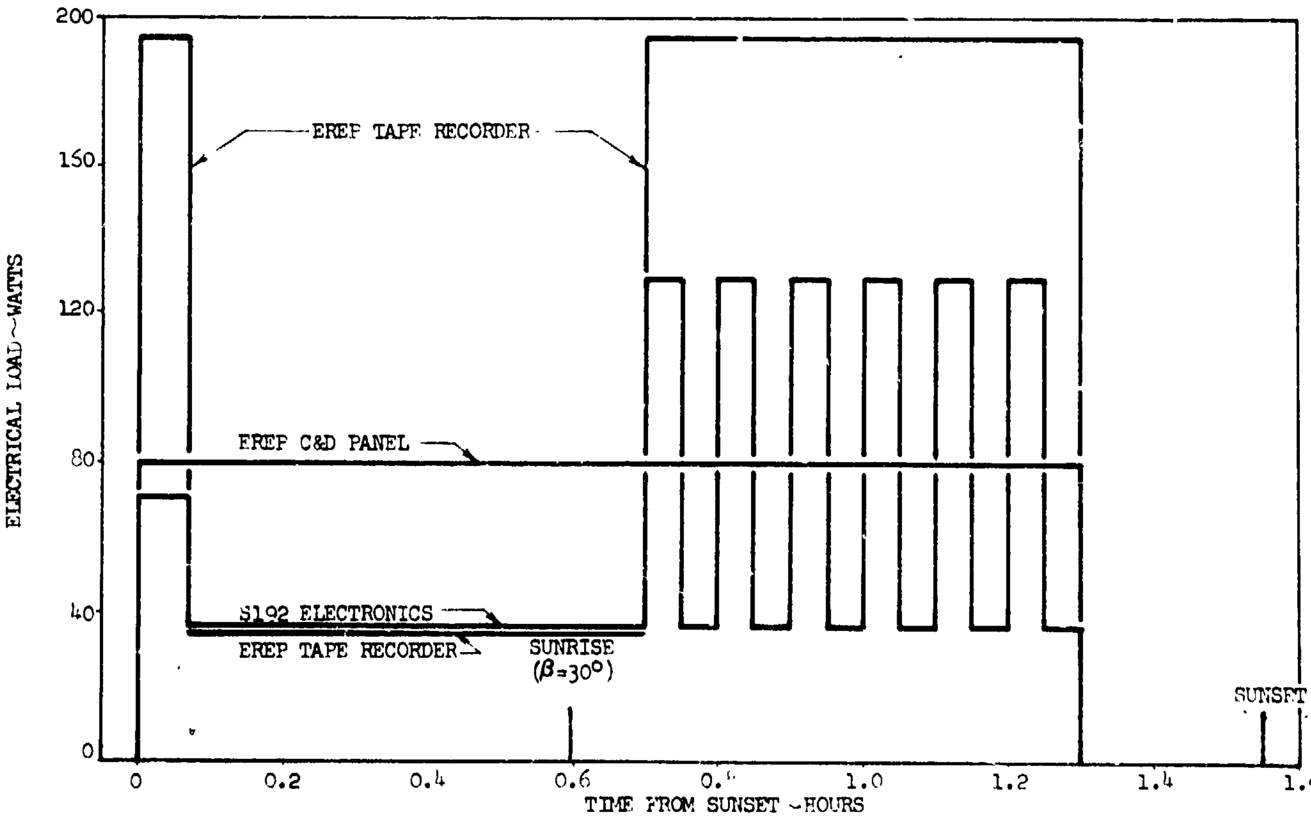


FIGURE 2.4-32 EREP ELECTRICAL LOADS (120° ARC PASS)

The water flowrate delivered to the MDA/STS interface was to be 220 lb/hr minimum. The pressure drop on the MDA side of the interface was to be less than 6.75 psid at 220 lb/hr. An abnormally low pressure rise across the pump would have been indicated by illumination of the LOΔP light on the ECS control panel 203. The design water delivery pressure was not to exceed 37.2 psia.

The water loop was serviced prior to flight with a mixture prepared in accordance with Process Bulletin 3-302 (Rev. E), containing 97% deaerated MMS-606 water, 2% by weight dipotassium hydrogen phosphate, 0.2% by weight sodium borate, and 500 PPM Roccal. The system was designed to allow in-flight reservicing. The system also contained an in-flight replaceable filter. The filter was installed on panel 235. It was to be replaced at the beginning of each mission with a preserviced filter brought up in the command module. The water system was protected against freezing during orbital storage as follows: the ATM tank module by its location within the heated vehicle, the ATM pump module by its location under the thermal curtains and its heat transfer with the tunnel wall, and the AM lines by insulation isolating them from cold environment while providing controlled heat exchange with warm coolant lines.

The system was deactivated for orbital storage and could be deactivated for EVA/IVA, if required to decrease coolant system heat loads during manned operations. The water-to-coolant heat exchanger and water pumps formed part of the ATM water pump module. The system was operated from the ECS control panel 203 and circuit breaker panel 202.

2.4.3.4 Battery Cooling

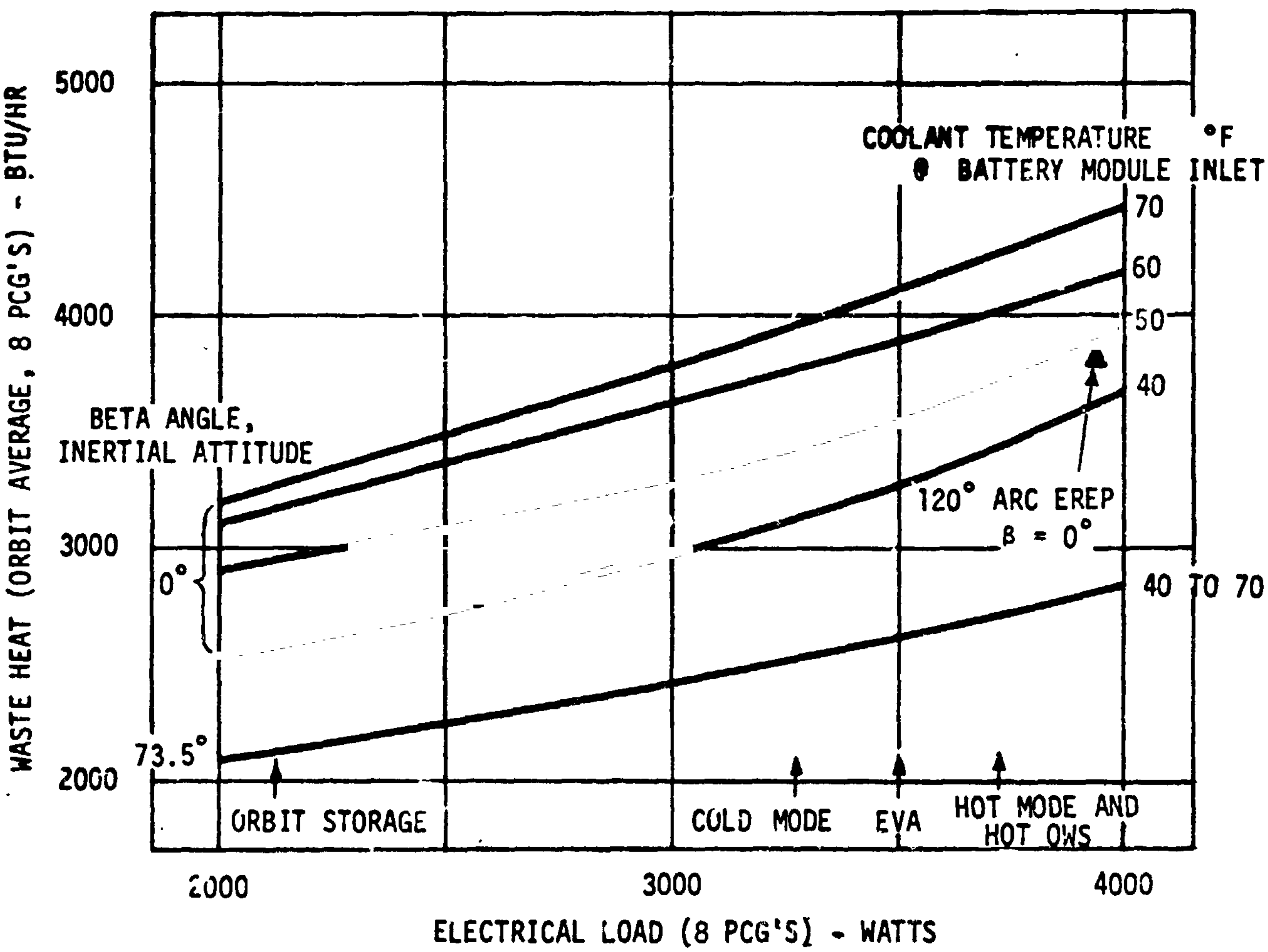
The battery cooling system reduced the temperature of coolant entering the power conditioning equipment to as close to 39 $\pm$3°F as permitted by the total coolant loop heat load and by radiator performance. The system for one coolant loop consisted of a three-passage heat exchanger and a 40°F TCV. This equipment was part of the suit/battery cooling module, as indicated in Figure 2.4-11. The electrical load of eight power conditioning groups (PCG) is shown in Figure 2.4-33, based upon the as-designed two OWS solar array system wing configuration. Due to the loss of solar array wing number two from the OWS during ascent, PCG waste heat predictions were made for operation of wing number one. Figure 2.4-34 presents waste heat predictions for the one wing solar array configuration with typical flight operating conditions noted.

For pre-lift-off operation, the temperature on top of the second battery cell case was predicted to be 47°F, as shown in Figure 2.4-35 for continuous battery charging and 40°F battery module coolant inlet temperature. The continuous charging condition is represented by the β = 73.5° curve.

For orbit operations the second battery top of cell temperature was predicted to be approximately 56°F during hot AM/MDA operations, per Figure 2.4-36. Similar top of cell temperatures were expected for hot OWS, EREP, and EVA modes. Transient effects, however, were expected to slightly reduce the top of cell temperatures during EREP and EVA. With a 40°F battery module coolant inlet temperature, the top of cell temperatures were expected to range from 46°F to 50°F during manned cold mode or unmanned rendezvous and orbital storage mode operations (refer to Figure 2.4-36).

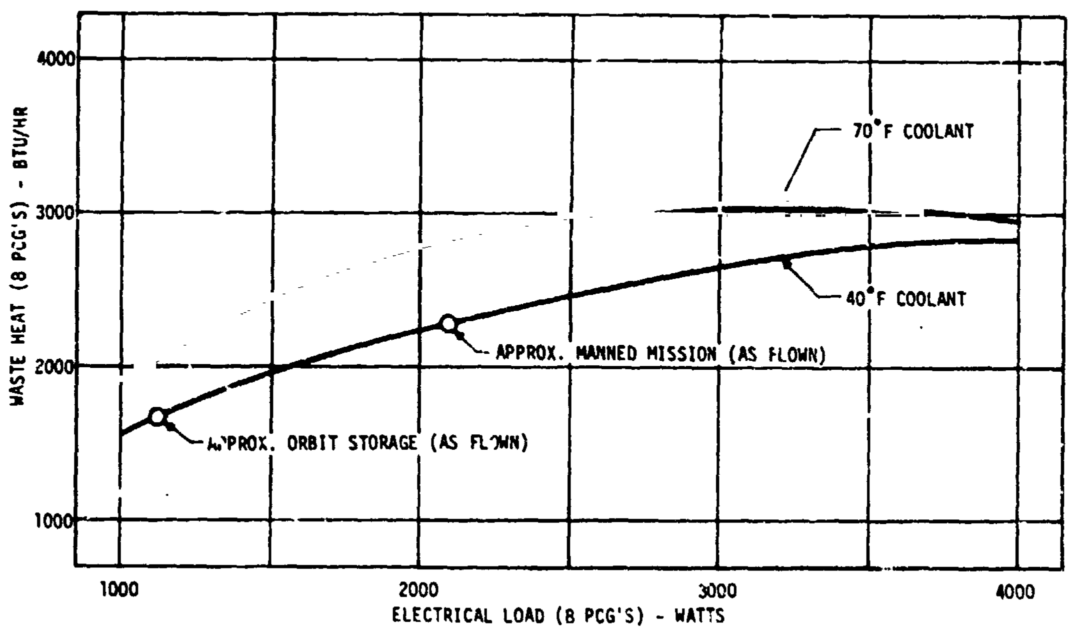
2.4.3.5 Thermal Coatings

Coatings were used to control the transfer of heat. The external thermal coating design values of the orbital vehicle are shown in Figures 2.4-37 and 2.4-38. The external surface coatings employed were aluminum, black, and white paints. The white paint, with a low ratio of solar absorptivity (α) to emissivity (ε), provided low effective sink temperatures and resulted in higher heat rejection rates. The hot case design value used for the radiator surface accounted for degradation during the mission due to UV exposure, meteoroids, exhause plume impingement, etc. Both black and white paints were used on the forward skirt and black paint was used on IU, FAS, and MDA.



1. QUALIFICATION TEST DATA (BATTERY S/N 18)
2. REGULATOR BUS VOLTAGE SET AT 30V
   +50 MV ON PCG 1 AND 5, AND
   -50 MV ON PCG 2, 3, 4, 6, 7 AND 8
3. PCG WASTE HEAT DISSIPATED APPROXIMATELY 77% TO COOLANT AND 23% BY RADIATION AND CONDUCTION TO STRUCTURE

FIGURE 2.4-33 POWER CONDITIONING GROUP WASTE HEAT – TWO SOLAR ARRAY WINGS



1. QUALIFICATION TEST DATA (BATTERY S/N 18)
2. REGULATOR BUS VOLTAGE SET AT 30V
   +50 MV ON PCG 1 AND 5, AND
   -50 MV ON PCG 2, 3, 4, 6, 7, AND 8
3. PCG WASTE HEAT DISSIPATED APPROXIMATELY
   77% TO COOLANT AND 23% BY RADIATION AND
   CONDUCTION TO STRUCTURE
4. BETA = 0°; SOLAR INERTIAL
   ATTITUDE, SUMMER SUN

FIGURE 2.4-34 POWER CONDITIONING GROUP WASTE HEAT – SOLAR ARRAY WING #1

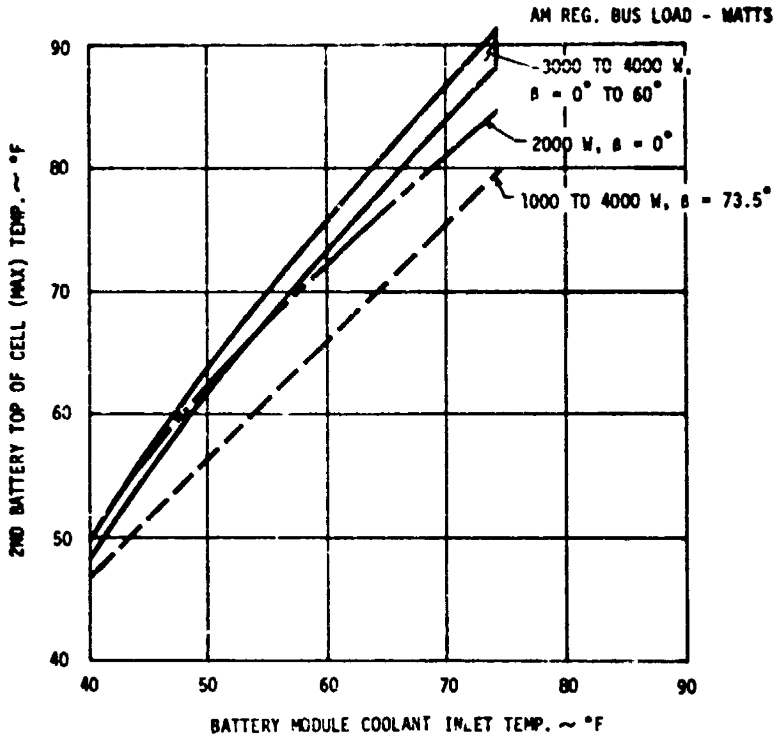


FIGURE 2.4-35 PREDICTED BATTERY TEMPERATURES – TWO SOLAR ARRAY WINGS

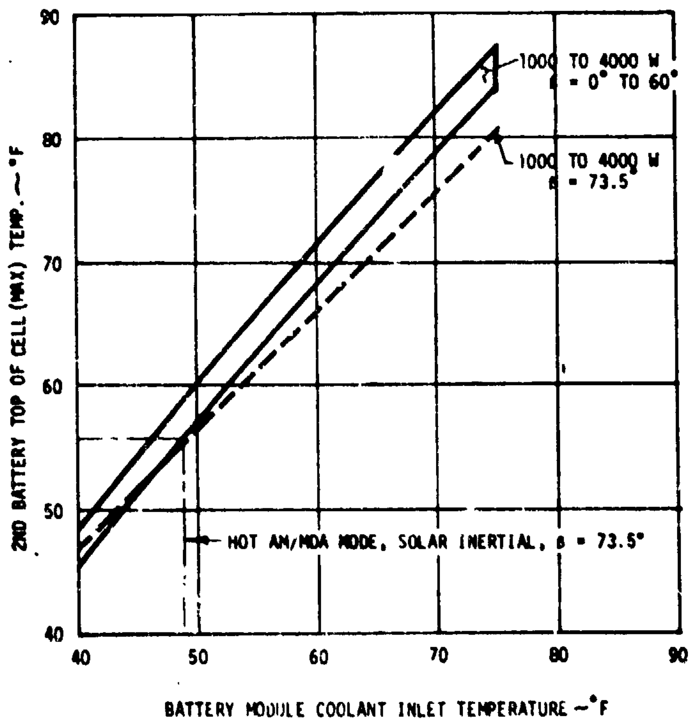


FIGURE 2.4-36 PREDICTED BATTERY TEMPERATURES – SOLAR ARRAY WING #1

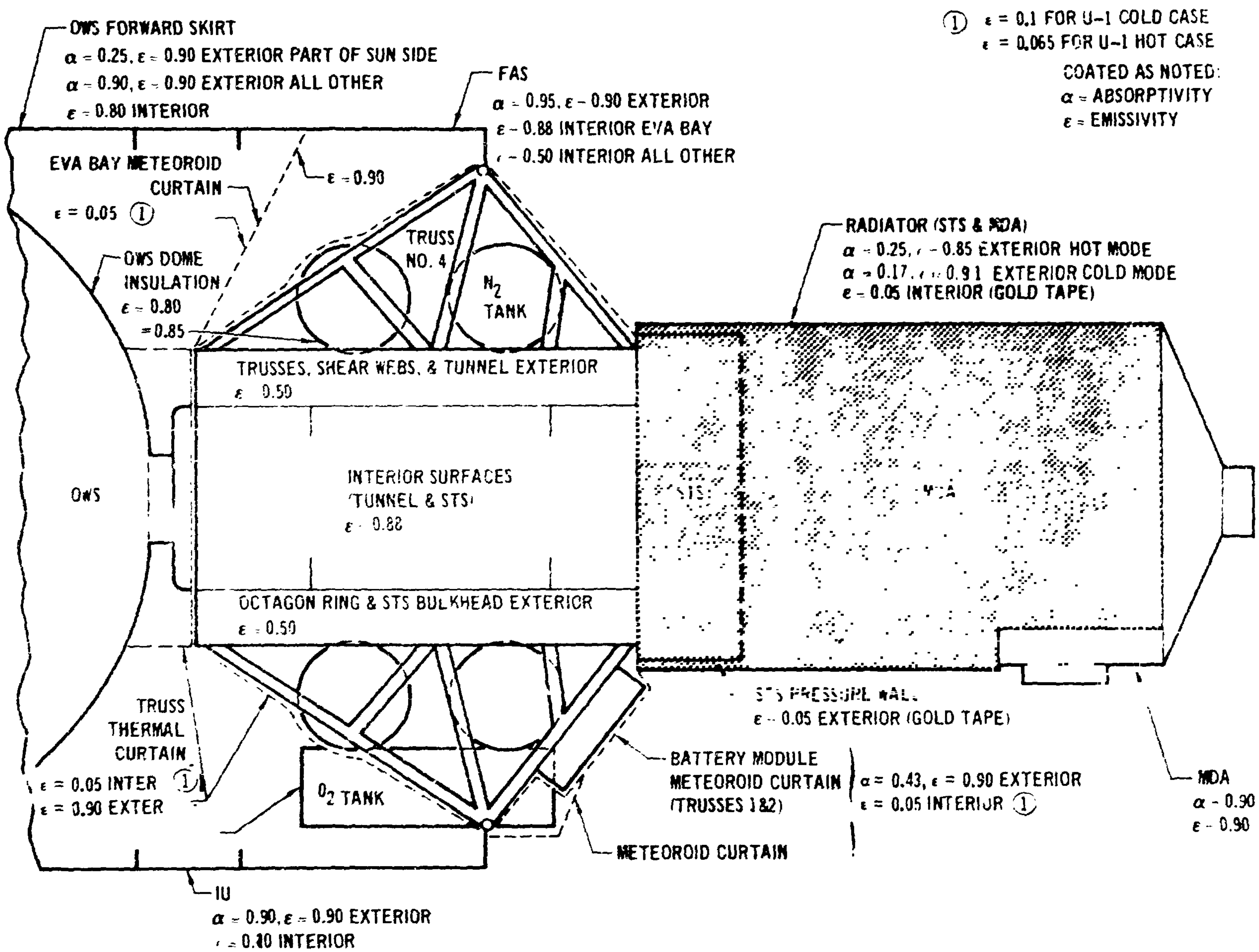


FIGURE 2.4-37 AM/MDA THERMAL COATING DESIGN VALUES

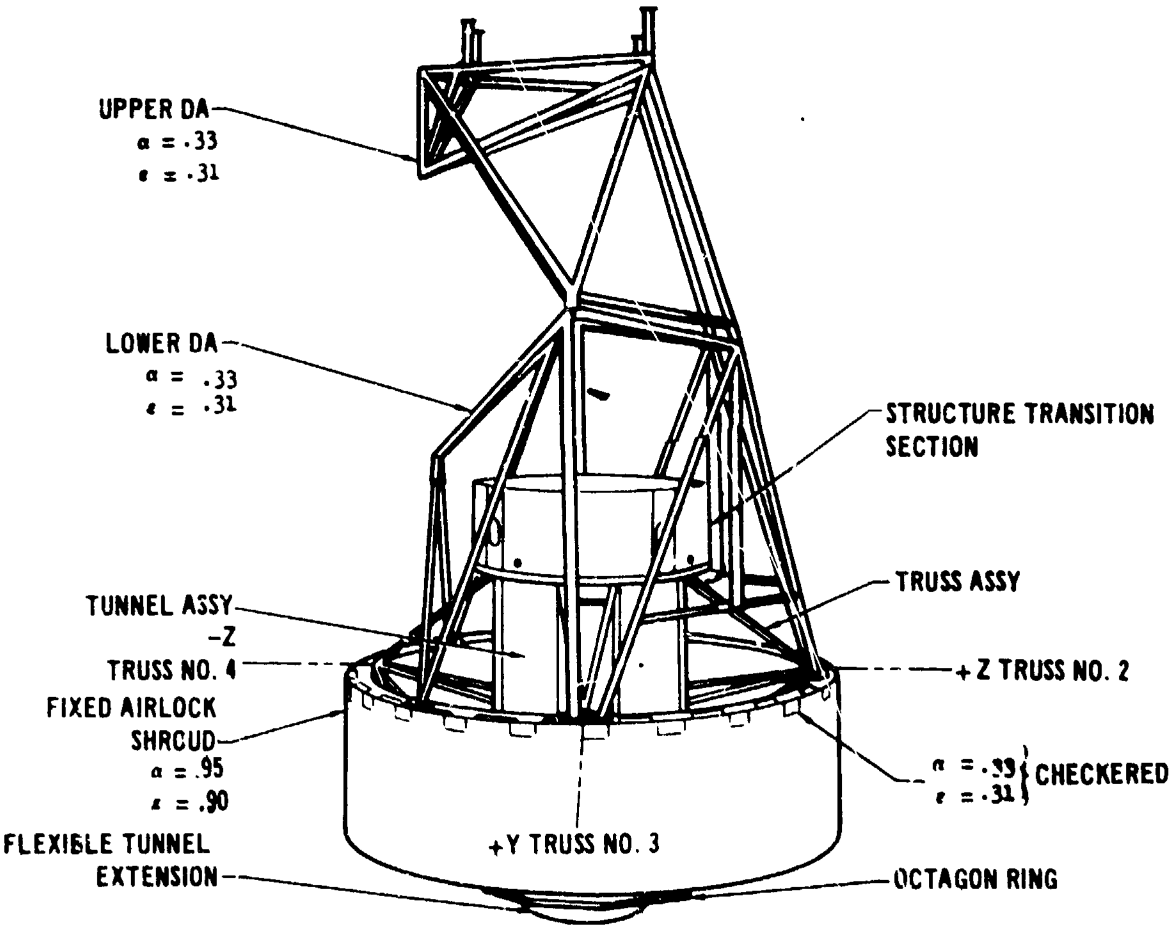


FIGURE 2.4-38 DA AND FAS THERMAL COATING DESIGN VALUES

An aluminum paint (α = .33, ϵ = .31) was used on the DA and on square markings provided around the top of the FAS to improve visibility during docking (see Figure 2.4-38). For analysis purposes the ATM rack sides and base were assumed to be insulated with α/ϵ = .3/.9 on the exterior surface of the insulation. The ATM solar arrays had an α/ϵ of .79/.87 on the active side. The internal surfaces within the enclosure of the meteoroid curtain include the AM trusses, tunnel outside surface, and the STS bulkhead which were coated with an aluminized paint to give an emissivity of 0.5. This emissivity was a design requirement for an early configuration at which time the trusses were painted. The emissivity was 0.5 on the internal surface of the FAS (except in the EVA quadrant where it was 0.88). The OWS forward skirt and dome had a 0.8 emissivity. The battery module components, exterior of the cylindrical section of the STS under the radiator, and the backsides of the radiator had low emissivity surfaces.

Protective care was exercised during the assembly, storage, and shipment to maintain thermal control surface quality. Radiator coating properties were monitored by measurements taken during various phases of assembly and installation. Individual absorptance and emittance measurements of the white paint on the radiator exterior surface were made for each square foot of surface area. Optical quality coatings were protected from contamination during handling and shipping. Personnel in contact with the coatings wore clean gloves. Storage and installation was in a clean dry environment. Protective covers were used for storage and shipping. Measurements and visual inspections of passive thermal control surfaces were made at KSC to verify that cleanliness had been maintained. Absorptivity measurements of the AM/MDA radiator were made prior to the vehicle leaving the VAB. Gold taped surfaces were visually inspected and emittance checks were made of any contaminated areas to determine emittance values and assess need for repair.

2.4.3.6 Thermal Curtains

The thermal curtains served to minimize heat loss and to isolate the AM from the variable orbital environment. They also minimized the electrical heater power required to make up AM heat losses during the cold mode and orbital storage operations. The AM thermal insulation system utilized thermal and meteoroid curtains, as illustrated in Figure 2.4-39.

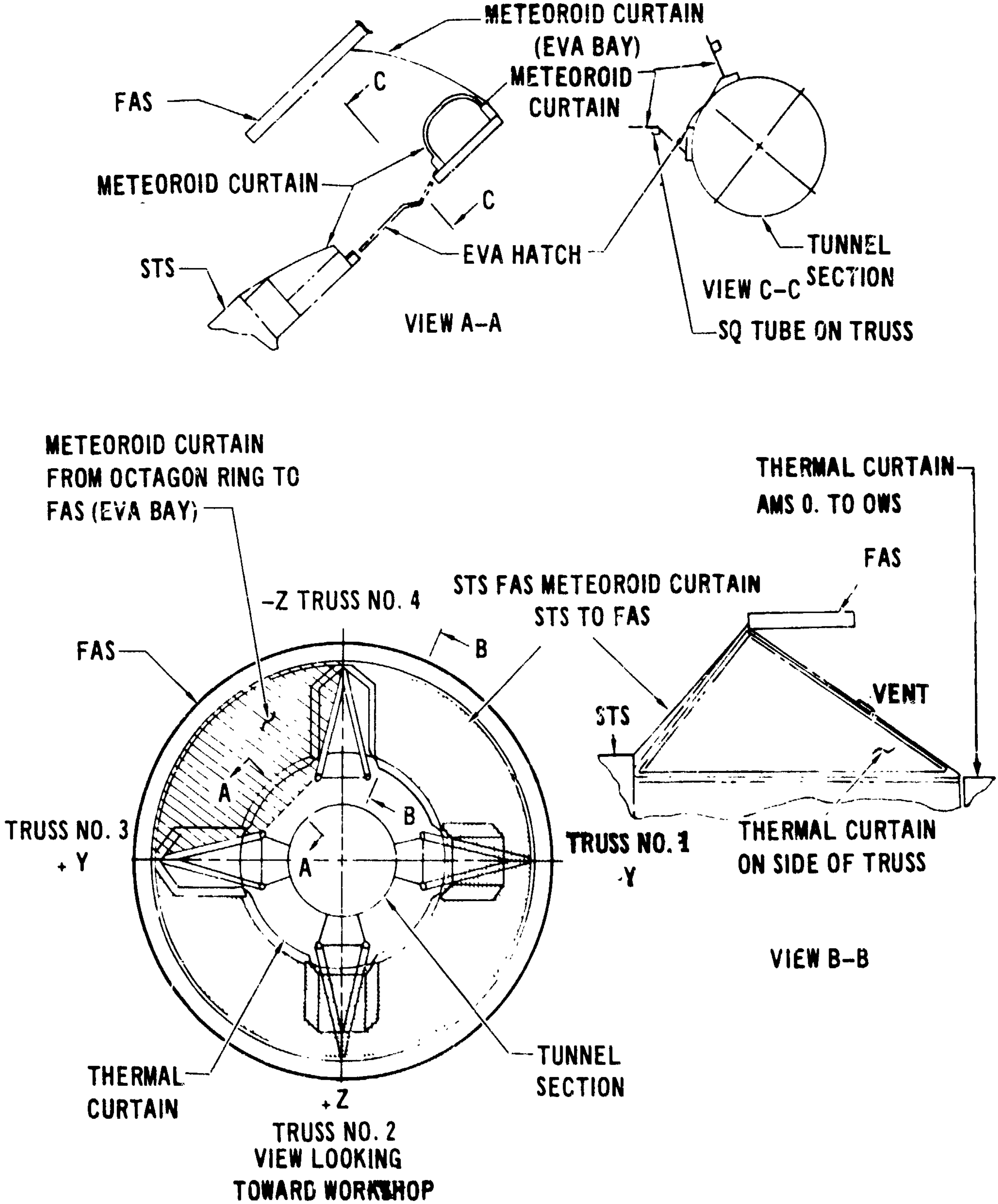


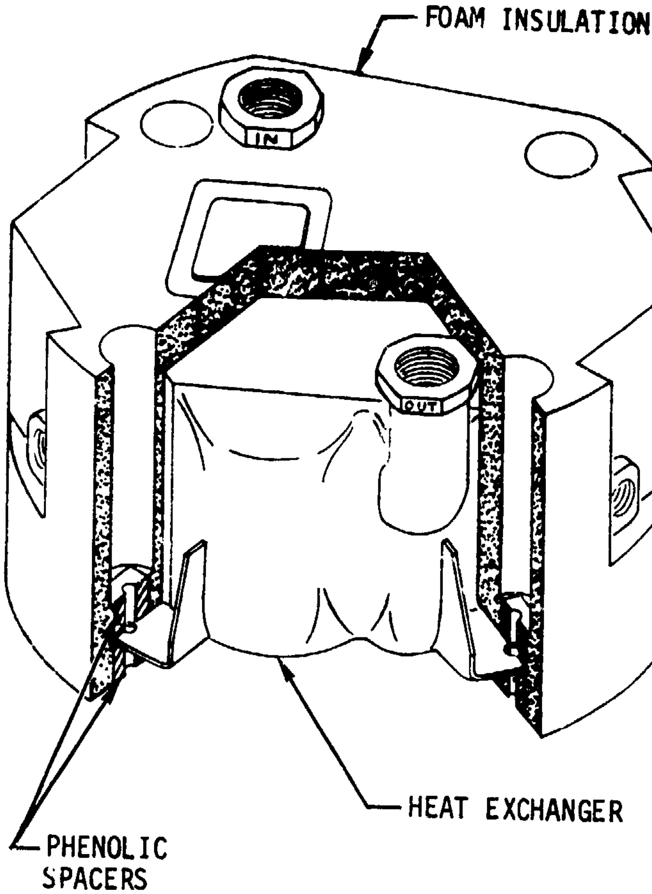
FIGURE 2.4-39 VEHICLE THERMAL INSULATION

The thermal curtains consisted of a single layer of fiberglass, Viton rubber impregnated on one side and gold coated on the other side. The thermal curtains were installed with the black Viton side external except for the quadrant covering the suit/battery cooling module. The meteoroid curtain was similar to the thermal curtain except it was thicker and had an off-white fiberglass cloth exterior facing.

Emissivity of the thermal and meteoroid curtain gold surfaces was measured after trial fit on the vehicle to assure desired values. All flight vehicle curtains were doubly bagged individually in heat sealed polyethylene bags. The bagging was done in white rooms at dew point temperatures below 60°F. The bagged curtains were then maintained at temperatures greater than 60°F to prevent condensation. Visual inspection of the curtains was made at KSC prior to vehicle installation. Emissivity test checks were made if there was evidence of contamination. Gold surface damage was repaired by gold tape.

2.4.3.7 Equipment Insulation

Insulation was used to limit the transfer of heat to equipment and in some cases provide acoustic suppression. Bulkhead fittings were insulated from support structure by fiberglass washers. Lines in the suit cooling, coolant, and ATM C&D Panel/EREP cooling systems were insulated from structure by fiberglass washers. All heat exchangers except the condensing heat exchangers and the ATM C&D/EREP heat exchanger were covered with low density foam insulation. The condensing heat exchangers were covered with mosite except over the water separator plate assemblies. The thermal capacitor module was insulated with glass fiber batt, and covered with a rubberized fiberglass cloth vapor barrier with a flap-type vent valve to provide launch ascent venting. The outer surface of the fiberglass cloth was overlayed with low emittance gold tape. Typical examples of equipment thermal insulation are shown on Figure 2.4-40. External water and coolant lines were routed together where practical and wrapped with Microfoil insulation. The ground coolant supply and return (FAS to GCHX) and the interfacing GCHX spacecraft line insulation consisted of 1 inch wide, 1/2 inch thick, 8 lb/ft^3 glass fiber insulation strips enclosed in heat sealed plastic. The spirally wrapped plastic bags were overlayed with aluminum foil tape followed by Mylar tape to seal the surfaces and provide high emissivity



INSULATION INSTALLATION
GROUND COOLANT HEAT
EXCHANGER
(COOLANT SYSTEM)

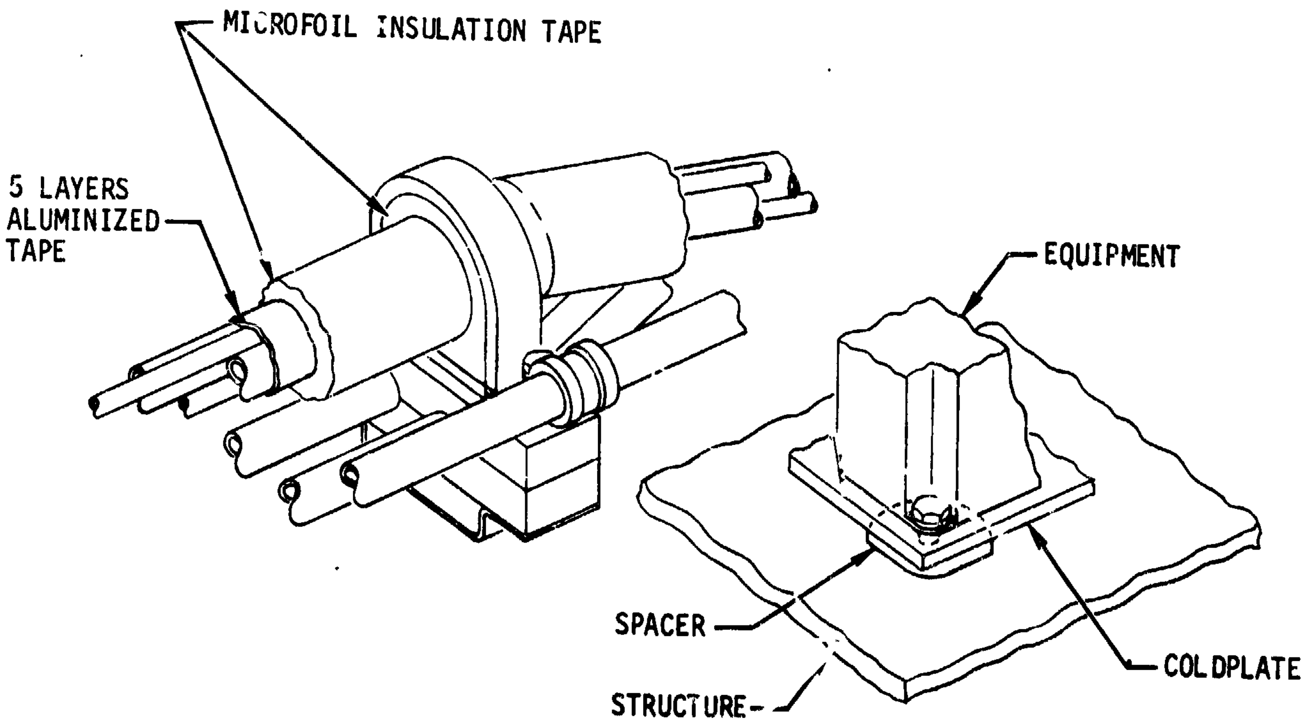


FIGURE 2.4-40 EQUIPMENT THERMAL INSULATION

exterior surface. Mosite insulation was used on internal water and coolant lines as required to limit condensation and heat leak during prelaunch and orbit. The suit/battery module water lines were not insulated because analysis indicated no potential water freezing problem in this subsystem (the tunnel wall and thermal curtain minimum temperatures in this area were expected to remain above the water freezing point). The internal portion of the condensate transfer line to the OWS was deliberately tied to the structure and was not insulated.

2.4.3.8 AM Wall Heaters

Fifteen electrical heaters were provided on the AM walls to maintain temperatures. Wall heater locations are shown in Figure 2.4-41.

Each heater had 42°F, 62°F, and 85°F (nominal closing settings) thermostats. The closing temperature design tolerance was ±5°F about the nominal setting. The opening temperature was 0.5°F to 8°F above the closing setting. The 42°F and 62°F thermostats were located approximately midspan between heater elements. The 85°F thermostats were located immediately adjacent to heater elements. The 62°F thermostats provided primary minimum wall temperature control during all mission phases. Wall temperature control by the 42°F and 62°F thermostats was selected by manual or DCS commands. The selection of the 85°F thermostats was manual only. The 85°F thermostats also provided the overall wall temperature limit for the 42°F and 62°F heater thermostats. The thermostats were operated from ECS control panel 203. The (On/Off/Cmd) AM wall heater switch normally was to remain in the CMD position. The (Hi/Lo) AM wall heater switch was inactive when the (On/Off/Cmd) switch was in Cmd; but was to be left in the Lo position. Both the 42°F and 62°F thermostats were to be activated by DCS during prelaunch and remain activated through the entire mission.

Each of the 15 heaters was rated at 15 ± 1.5 watts at 28 VDC. Tests on actual heater elements indicated that 14.4 watts average heat dissipation would occur with 28V at the heater terminals. Even though the orbit storage minimum bus voltage was 28V, the heater terminal voltage was lower due to line losses. Consequently, an average heater dissipation was estimated at 12.4 watts.

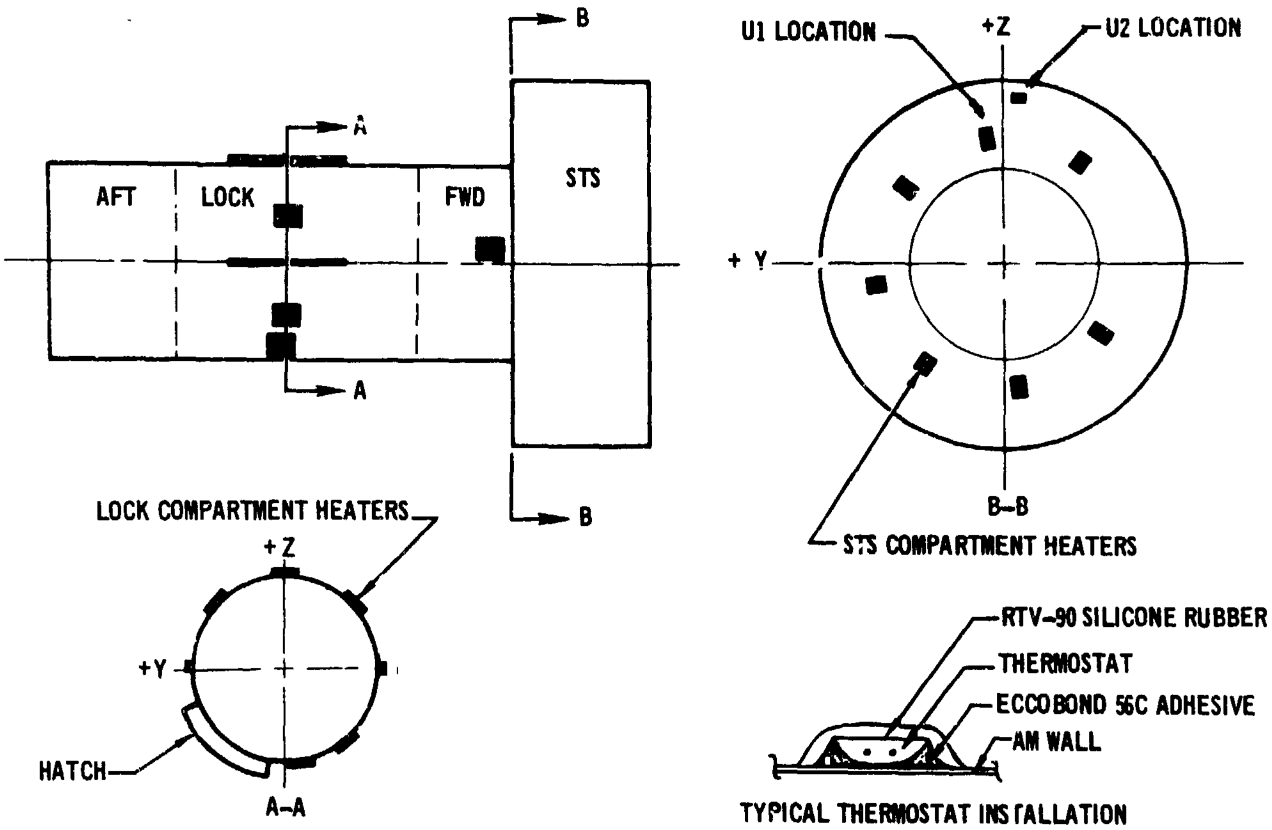


FIGURE 2.4-41 WALL HEATER LOCATION/THERMOSTAT INSTALLATION

The heaters were sized to provide a minimum wall temperature of 40°F during orbit storage. With all the AM wall heaters operating and the MDA wall heaters operating with the 45°F thermostat control, the AM wall temperatures were predicted to remain at 44°F to 53°F. With only the MDA heaters operating, the AM wall temperatures were predicted to range from 41°F to 49°F.

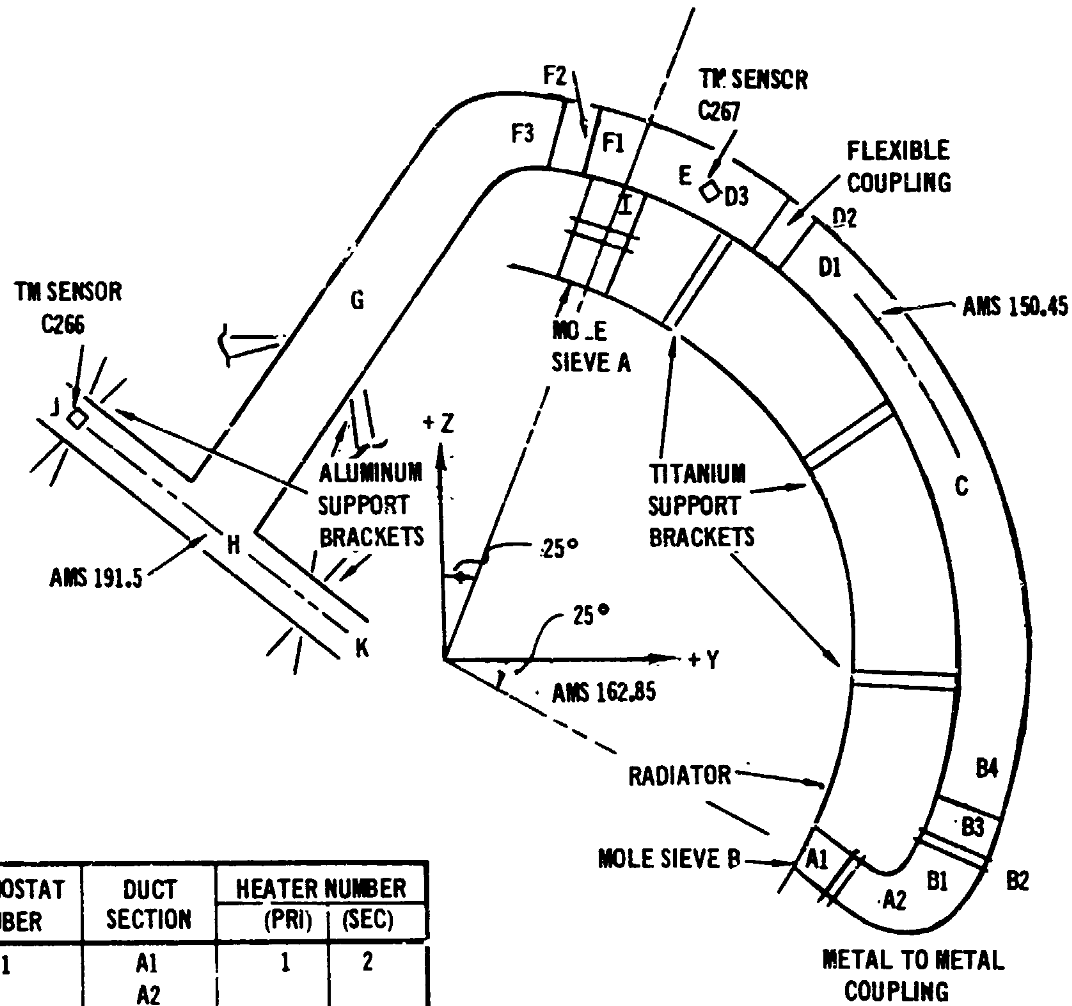
Four STS/Fwd and four lock tunnel heaters were powered from AM Bus 1, circuit 1. Four STS/Fwd and three lock tunnel heaters were powered from AM Bus 2, circuit 2. Circuit breakers on panel 200 provided protection for each circuit.

2.4.3.9 Molecular Sieve Exhaust Duct Heaters

Heaters were provided to prevent freezing of water vapor during molecular sieve operation. The heaters were located on the seven separate duct sections shown in Figure 2.4-42. Primary and secondary heaters were mounted to the duct at each section. Primary heater thermostats were set to activate at 50°F, nominal. The secondary heaters acted as a redundant system and were activated by nominal 42°F thermostats. The closing design tolerance was ±5°F about the nominal. The opening range was 0.5°F to 8°F above the closing setting.

Heater thermostats 3, 4, 5, 6, and 7 were activated manually whenever mole sieve A was operated. All heater thermostats were activated manually whenever mole sieve B was operated. Heater power was controlled from ECS control panel 203 and circuit breaker panel 200. Two temperature sensors for telemetry (C266 and C267) were mounted on the duct to monitor system performance.

Primary and secondary heaters each had a total capacity of 62.4 watts at 28 VDC. The heaters were sized based on tradeoffs of heater power, insulation thickness, and external surface radiation properties. To minimize the heater power required, Microfoil insulation tape (0.5 inch thick) was wrapped around the duct. A low emissivity tape (Schjeldahl G1015) was then wrapped around the insulation. Perforations in the tape allowed venting during launch ascent.



THERMOSTAT NUMBER	DUCT SECTION	HEATER NUMBER (PRI)	(SEC)
1	A1 A2 B1* B2 B3	1	2
2	B4 C* D1 D2	3, 5	4, 6
3	D3* E F1 F2	7	8
4	F3 G* H	11, 13	12, 14
5	I*	9	10
6	J*	17	18
7	K*	15	16

*THERMOSTAT LOCATION

FIGURE 2.4-42 MOLECULAR SIEVE OVERBOARD EXHAUST DUCT HEATERS

2.4.4 Testing

Testing was performed to provide information needed by engineering for design, to qualify a particular part numbered component, to verify that the particular part and serial numbered components operated properly, to verify that U-1 and U-2 modules and systems functioned properly, and to support verification that the vehicle was ready for flight. Postlaunch tests were conducted to provide information needed for real time mission planning. Information on the test philosophy is presented in Section 5 of this report.

2.4.4.1 Development Tests

Development tests were performed on components and systems to obtain data on the functional characteristics needed to support the design process. Test requirements were specified by Test Request (TR).

A. Performance Tests - Performance tests were conducted to establish the performance of new components and systems. Some were conducted by vendors to satisfy requirements identifed in Specification Control Drawings (SCD). Those tests conducted by MDAC-E are summarized below:

- TITLE — ATM C&D Panel Cooling Subsystem Development Test

  BACKGROUND — The ATM C&D panel water cooling loop transferred heat dissipated by the panel, located in the MDA, to the AM coolant loop.

  OBJECTIVE — Establish the performance of the water cooling module, to assure that ATM C&D panel heat loads could be rejected to the Airlock coolant loop and that cooling water could be supplied to the AM/MDA interface within specified temperature limits.

  RESULTS — The temperature of the water, at the inlet of the C&D panel simulator, ranged from 45.4°F during operation with lowest coolant supply temperature (40°F) to 84.6°F when the coolant was supplied at 80°F. The system operated satisfactorily within predicted limits during all conditions tested, Reference TR 061-068 41.

- TITLE Space Radiator Convection Heat Transfer Element
  BACKGROUND Analytical prediction of radiator performance required inputs of fluid-to-wall convection heat transfer coefficient.
  OBJECTIVE Determine the fluid-to-wall convection heat transfer coefficient for the Airlock space radiator configuration.
  RESULTS Temperature distribution, flow, and pressure drop data was determined for 11 test conditions, Reference TR 061-068.67.
- TITLE Ground Cooling Heat Exchanger Test
  BACKGROUND Additional performance data was needed to assist in defining cooling system modifications.
  OBJECTIVE Obtain performance data on ground cooling heat exchanger.
  RESULTS Performance data was obtained for each of the three heat exchanger passages, Reference TR 061-068.75.
- TITLE Coolant System Thermal Development Test
  BACKGROUND Performance verification was necessary for operating conditions expected during an actual mission.
  OBJECTIVE Develop a coolant system and verify its operation during conditions defined for orbital position of the spacecraft, heat loads of equipment, and astronauts metabolic heat load.
  RESULTS The coolant system operated satisfactorily during normal and emergency modes of operations tested. The system maintained stable control of all temperatures throughout the loop, Reference TR 061-068.76.
- TITLE Coolant Reservoir Dry Cycle Test
  BACKGROUND Dry pressure cycling occurred when the Airlock coolant system was leak checked with gas.
  OBJECTIVE Verify that dry cycling would not adversely affect operation of the coolant reservoirs.
  RESULTS The coolant reservoirs operated satisfactorily during cycle and burst test. Examination of chamber walls revealed no abnormal wear, Reference TR 061-068.79.

- TITLE — Coolant Reservoir Performance Test

  BACKGROUND — At low temperatures the Freon-114 pressurant in the coolant reservoirs existed in a two-phase state and the performance characteristics were uncertain.

  OBJECTIVE — Determine the pressure-volume characteristics of the coolant reservoir in the temperature range of 40°F to 120°F.

  RESULTS — Pressure-volume characteristics of a coolant reservoir were determined over the required temperature range for use in subsequent analysis, Reference TR 061-068.90.

- TITLE — Stretch Pressure Test of the Cabin Heat Exchanger

  BACKGROUND — Strength data was needed to evaluate an overpressure condition of the cabin heat exchanger.

  OBJECTIVE — Determine the effect of overpressure on flowrate characteristics of a cabin heat exchanger and determine its rupture pressure.

  RESULTS — The cabin heat exchanger was pressurized to 230 psig. The pressure drop was 1.77 psi and 1.82 psi at 220 lb/hr, respectively, before and after pressurization. The unit was pressurized at 1000 psi without rupture, Reference TR 061-068.91.

- TITLE — OWS Thermal Capacitor (Undecane filled) Melting/Freezing Characteristics

  BACKGROUND — A plate fin thermal capacitor failed at MDAC-W.

  OBJECTIVE — Determine the melting and freezing characteristics of the plate fin capacitor when filled with Undecane wax and determine stress limits when filled with Tridecane.

  RESULTS — None of the configurations tested produced acceptable results under all simulated flight conditions, Reference TR 061-068.92.

- TITLE — Honeycomb Thermal Capacitor Development Testing

  BACKGROUND — A new thermal capacitor design was initiated.

  OBJECTIVE — Evaluate the structural integrity of the 61A830371-1 thermal capacitor segment assembly.

  RESULTS — Strains did not exceed the 1500 microinches per inch allowable limit during any of the conditions tested. Visual inspection of the specimen during the test and after test revealed no structural deformation or leakage, Reference TR 061-068.92.01.

- TITLE — Honeycomb Thermal Capacitor Thermal Performance Tests - Tridecane Wax

  BACKGROUND — A prototype thermal capacitor design was initiated.

  OBJECTIVE — Determine thermal performance characteristics of the prototype honeycomb thermal capacitor.

  RESULTS — The thermal performance of the capacitor was satisfactory for all test conditions, Reference TR 061-068.92.02.

- TITLE — Honeycomb Thermal Capacitor Undecane Development Testing

  BACKGROUND — A new thermal capacitor design was initiated.

  OBJECTIVE — Evaluate the structural integrity of the 61A830371-5 thermal capacitor segment assembly.

  RESULTS — Strains did not exceed the 1500 microinches per inch allowable limit during any of the conditions tested, Reference TR 061-068.92.03.

- TITLE — Honeycomb Design AM Thermal Capacitor "Pre-Qual" Thermal Evaluation

  BACKGROUND — Evaluation of a production thermal capacitor unit was required.

  OBJECTIVE — Determine thermal performance characteristics of a production thermal capacitor segment.

  RESULTS — The thermal performance characteristics were satisfactory for all test conditions, Reference TR 061-068.96.

- TITLE Development Testing Thermal Capacitor Backup Designs

  BACKGROUND Use of alternate fluids for thermal capacitor servicing was investigated.

  OBJECTIVE Determine the structural and performance characteristics of a thermal capacitor serviced with Artech fluid.

  RESULTS Excessive surface strains occurred and the test was aborted. It was concluded that Artech fluid is not suitable for use in the thermal capacitor, Reference TR 061-068.98.

B. Endurance Test - An endurance test, designated ET-1 and documented by Report TR 061-068.35, was conducted to verify that system components had the endurance to function properly during a complete mission. The test hardware included more than 70 Airlock flight configuration components assembled into functional systems. The test was designed to load the components and make them perform under conditions expected during flight. The test followed the proposed Skylab mission plan which consisted of 3 Active Phases and 2 Orbital Storage Phases covering a real time period of 8 months.

All components initially assembled into the ET-1 Thermal Control System functioned adequately except the 40°F temperature control valve. Coolant temperature at the valve outlet port cycled between 32°F and 48°F (specification limit was 40 $^{+2}_{-4}$°F) when the temperature of coolant entering the cold port of the valve was less than 31°F. Temperature of the cold coolant was kept above 32°F and testing was continued. The flight coolant system configuration was changed to improve temperature control.

2.4.4.2 Qualification Tests

Qualification test documentation is available for all Airlock components and systems. Test results are summarized in MDC Report G499, Volume V.

2.4.4.3 Component Tests

Tests were conducted to prove the components and systems function properly.

A. Acceptance Test - An acceptance test had to be passed at the Vendor's plant before shipment to MDAC-E. Acceptance test requirements were specified in the Acceptance Test Procedures. These procedures were prepared by the Vendor and approved by MDAC-E.

B. Preinstallation Acceptance Test - A preinstallation acceptance (PIA) test had to be passed at the MDAC-E plant to prove that the hardware arrived in good condition prior to going into the crib which supplied parts for U-1, U-2, and spares. PIA test requirements were defined by MDAC-E Service Engineering Department Report (SEDR).

2.4.4.4 System Tests

System tests were conducted to verify that modules and systems operated properly. System test requirements were specified by SEDR.

A. Major Subassemblies - Major subassemblies were tested prior to installation during vehicle buildup. A tabulation of subassemblies tested prior to installation is shown in Figure 2.4-43.

SEDR	TITLE	SYSTEM
D3-G51	MISC. FLUID SYSTEM FUNCTIONAL TESTS	OXYGEN, NITROGEN, COOLANT
D3-M51	MISC. AM FLUID SYSTEM MANUFACTURING TESTS	MISCELLANEOUS
D3-G54	STS H/X SUBASSEMBLY FUNCTIONAL TEST	COOLANT, VENTILATION
D3-G66	OWS COOLING SUBASSEMBLY FUNCTIONAL TEST	COOLANT, VENTILATION
D3-G68	CONDENSING HEAT EXCHANGER MODULE	COOLANT, VENTILATION
D3-F41	MDA RADIATOR LEAKAGE (AT MMC)	COOLANT
D3-G41	STS/MDA RADIATOR PANELS LEAK AND FLOW TEST	COOLANT
D3-H41	MDA RADIATOR LEAKAGE TEST	COOLANT
D3-M41	STS/MDA RADIATOR MANUFACTURING CHECK	COOLANT
D3-G42	SUBASSEMBLY COOLANT SYSTEMS	COOLANT
D3-G43	COOLANT MODULE FUNCTIONAL TEST	COOLANT
D3-M43	COOLANT MODULE LEAK TEST	COOLANT
D3-G45	ATM C&D PANEL COOLING MODULE	COOLANT, WATER, OXYGEN
D3-G47	COOLANT RESERVOIR MODULE	COOLANT
D3-G48	SUIT/BATT COOLING MODULE FUNCTION	COOLANT, WATER
D3-M48	SUIT/BATT COOLING MODULE MANUFACTURING	COOLANT, WATER

FIGURE 2.4-43 THERMAL CONTROL SUBASSEMBLY TESTS

B. Systems - Final system and integrated acceptance test flow of the TCS at the contractor facility are depicted in Figures 2.4-44 and 2.4-45 for the Coolant System and the ATM C&D/EREP Cooling System.

- Coolant System - Prior to the first systems test, leak tests were performed and servicing was accomplished. During SEDR D3-N46-1, the STS Radiators were leak checked with a helium mass spectrometer and a flow ΔP test was performed to verify that the radiator flow pressure drop was within the specified limits. The radiators were then drained and flushed with solvent, and connected to the vehicle coolant system. A complete system leak check was performed. The vehicle was then serviced per SEDR D3-F90-1 Vol. I. Servicing consisted of filling the primary and secondary loops with MMS 602 coolant, establishing proper reservoir pressure and verification of acceptable entrapped air volume. The pumps were verified for proper operation by monitoring system flow and ΔP data.
- Systems Validation - The first system test was performed during Systems Validation (SEDR D3-N70). The MDA portion of the radiator system was not installed during this test so a thermal/pressure control unit was added to the primary loop. This unit was set up to supply coolant temperatures simulating expected radiator outlet temperatures during flight while controlling the differential pressure of the unit to correspond to expected flight system conditions. During SEDR D3-N70-1 system performance was verified utilizing both single and dual pump modes of operation. All parameters were verified to be within specified limits. Caution and warning system functions were verified by applying heat and cooling to the appropriate sensors. The automatic switch-over capability of the cooling system was demonstrated. Coolant fluid samples were withdrawn and particle and chemical analysis were performed. A thermal stability test was conducted to verify the thermal stability of the vehicle cooling systems during extreme thermal conditions. Various heat loads were applied and system response noted. A coolant pump "start-up" during simulated orbit storage also was demonstrated. Deservicing of the vehicle prior to connecting the MDA radiators to the coolant loops was accomplished by SEDR D3-F90, Vol. II. It was also necessary to replace several damaged flexible hoses and to remove the coolant pump module for transducer replacement and to rework

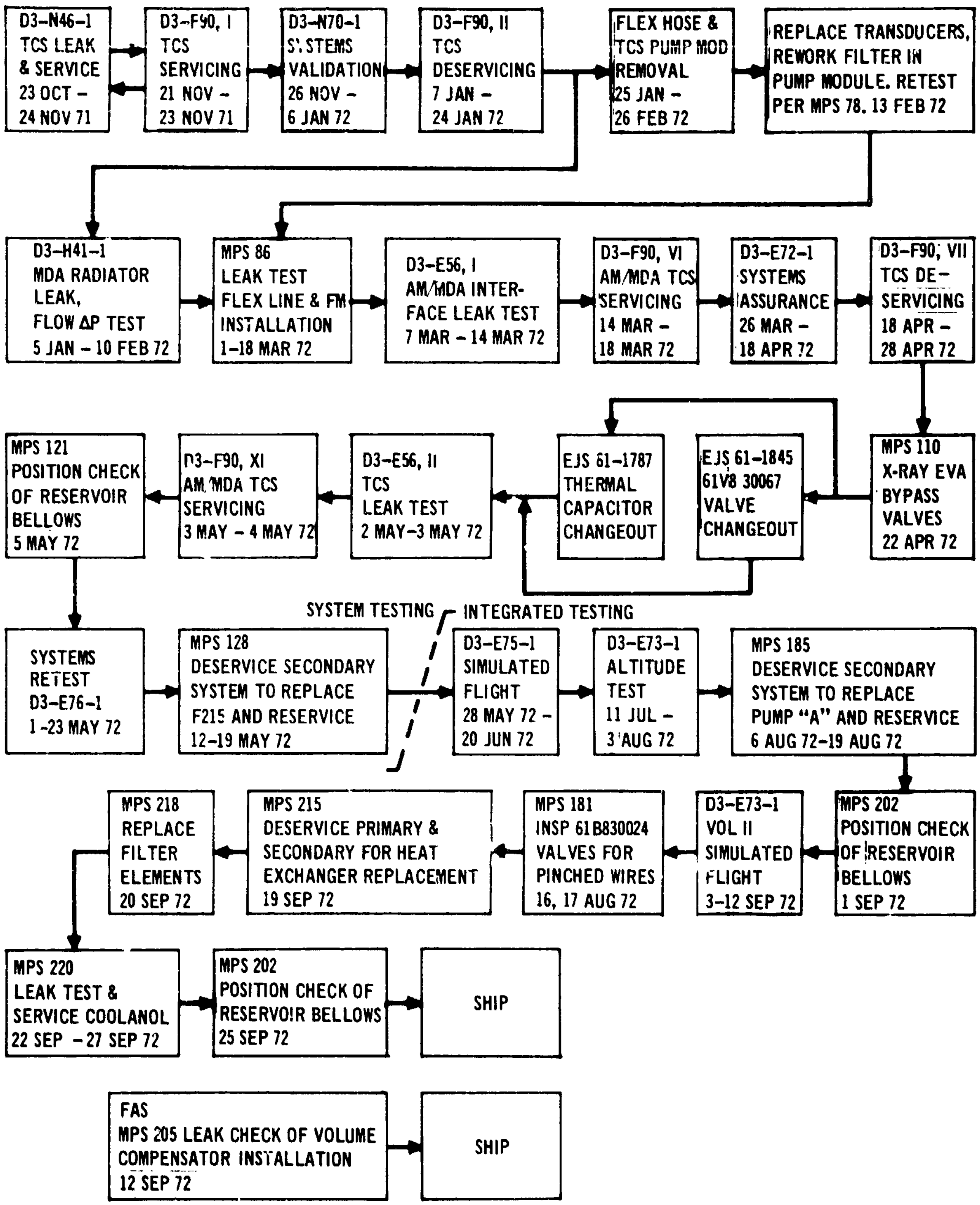


FIGURE 2.4-44 COOLANT SYSTEM TEST HISTORY - MDAC-E

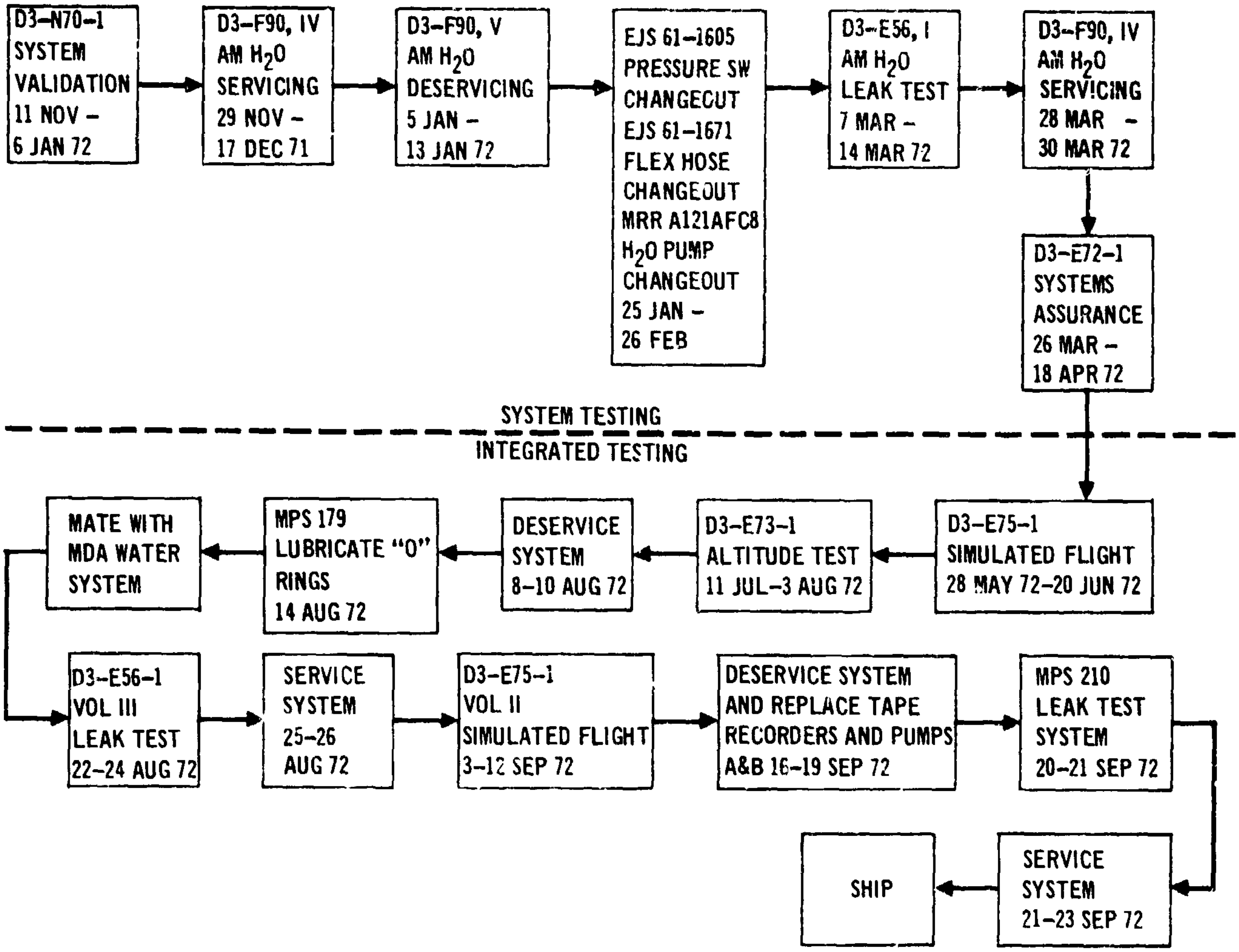


FIGURE 2.4-45 ATM C&D PANEL/EREP COOLING SYSTEM TEST HISTORY - MDAC-E

module filters. After rework of the pump module, a module retest was performed by MPS 78. Following module installation miscellaneous line leak checks were performed by MPS 86. A system leak check per SEDR D3-E56-1, Vol. I was performed using a helium mass spectrometer.

Reservicing of the coolant loops was then performed per SEDR D3-F90-1, Vol. VI. A system air inclusion test was performed, and particle count and chemical analysis of the vehicle coolant fluid was accomplished.

- Systems Assurance - During SEDR D3-E72-1 the AM/MDA coolant systems were activated to provide cooling for subsequent testing. Cooling system pumps and transducer performance were verified in single and dual pump modes. The coolant systems were then placed in command mode and various pump, relay and pump inverter combinations verified. The high and low temp parameters of the C&W system were verified. After completion of SEDR D3-E72-1, a decision was made to redesign the thermal capacitor and to rework the EVA bypass valves per requirements of EJS 61-1845. The valves were x-rayed per MPS 110. System primary and secondary coolant loops were deserviced and the original thermal capacitor and three-way solenoid valves removed. The equipment removed was replaced and the vehicle was leak tested per SEDR D3-E56-1, Vol II with a mass spectrometer using a mixture of nitrogen and helium. The vehicle coolant systems (AM/MDA) were then serviced per SEDR D3-F90-1, Vol. XI. Air entrapment tests were performed on both coolant loops. After vehicle servicing a check of reservoir position was performed per MPS 121.

  A systems retest per SEDR D3-E76-1 was begun after the completion of the reservoir position test verified that all reservoirs were properly serviced. The coolant systems were activated and verification of performance, using both DCS and Manual control, was accomplished. A functional test of the coolant filters located on both the coolant pump module and the reservoir module was performed to verify proper operation of the filters in both "off" and "on" positions. During the performance of SEDR D3-E76-1 the secondary coolant loop flowmeter system failed and the secondary loop was deserviced, a flowmeter system was removed from U-2 vehicle and installed on U-1 per MPS 128. MPS 128 also leak checked and reserviced the secondary coolant loop to prepare for Simulated Flight Test.
- ATM C&D Panel/EREP Cooling System (AM Portion) - When SEDR D3-N70 was performed, the MDA was not mated to the Airlock Module (AM). Therefore, the AM portion of the ATM C&D panel/EREP cooling system was jumpered at the AM/MDA interface to provide a flow-thru capability. During SEDR D3-N70 the AM portion of the loop was leak tested and serviced per D3-F90-1, Vol. IV. After servicing, the system was operated and

functionally tested. After D3-N70, the system was deserviced and vacuum dried per D3-F90-1, Vol. V to change pressure switch configuration per EJS 61-1605 and flex hoses per EJS 61-1671. The water pumps were also replaced per MRR A121AFG8. Since the MDA portion of the loop was still not available, the AM portion of the loop was maintained jumpered at the AM/MDA interface and another leak test performed per SEDR D3-E56-1, Vol. I. The system was serviced per SEDR D3-F90-1, Vol. IV and pumps were operated to support Systems Assurance (D3-E72-1) testing.

- AM Heating System - The electrical heating portion of the Thermal Control System was tested by heating and/or cooling appropriate thermostats and obtaining respective voltage at associated test points. Resistance of heating elements was also verified.

During Systems Validation (SEDR D3-N70), testing consisted of a heater resistance test and a voltage test of the STS, lock, mole sieve and condensate heaters. Voltage tests were accomplished by heating and cooling thermostats and by DCS commands which operated the high and low temperature heaters. A reverification of AM wall heaters and condensate system heaters was performed during Systems Assurance (SEDR D3-E72).

During SEDR D3-N70, mole sieve heater No. 9 resistance measurement indicated an open circuit. A miswired thermostat was corrected and satisfactorily retested. Mole sieve B duct heaters No. 5 and No. 6 were found to have internal shorts to structure. Heater No. 6 was replaced and verified acceptable. Heater No. 5 was replaced and subsequently retested at KSC. The cause of failure was determined to be corrosion caused by moisture and dissimilar metals.

Mole sieve duct heaters No. 3, No. 4, and No. 5 had corrosion damage on the heater rods at KSC. The damaged heater rods were replaced and retested. All heater rods were waterproofed to prevent electrolytic corrosion.

2.4.4.5 Integrated Tests

Integrated tests were conducted to verify the vehicle was ready for flight both before it left the factory for the launch site and after it arrived at the Kennedy Space Center (KSC).

A. Factory Tests - The tests conducted by MDAC-E in St. Louis are summarized below. Integrated test requirements were specified by SEDR.

(1) Coolant System - Integrated testing of the Coolant System at St. Louis is shown in Figure 2.4-44.

- Simulated Flight Test - Thermal tests of the coolant system were conducted during simulated flight checkout of the vehicle per SEDR D3-E75-1, Vol. I. Thermal tests were conducted to verify the AM coolant system performance during hot and cold modes, and during simulated EVA operation and Orbit Storage. The Simulated Flight Test (SEDR D3-E75) occurred after the Coolant System was completed (i.e., all radiator panels and new thermal capacitor module were installed). Since it was not desirable to break into the Coolant System to control radiator outlet temperatures directly, it was decided to control temperatures through the ground cooling heat exchanger.

  As in the previous test in conjunction with Systems Validation Test (SEDR D3-N70), it was planned to operate the system at various temperature profiles to ensure the proper operation of the Coolant System. The initial phases of the test were performed without problems, however, the low temperature portions of the test, i.e., simulated radiator temperatures below -40°F, were not accomplished. Heat loss to ambient from the ground cooling heat exchanger and associated plumbing under the abnormal conditions being applied was such that the desired system temperatures could not be attained. Since all other phases of the test were successful and tests performed in Systems Validation Test (SEDR D3-N70) had demonstrated acceptable system performance, it was not deemed necessary to modify the vehicle to perform the low temperature tests.

- Altitude Test - After the simulated flight test the vehicle was moved into the altitude chamber for the altitude test per SEDR D3-E73-1. During the altitude chamber test the reed switch on coolant pump A of the secondary coolant loop malfunctioned. After completion of the altitude chamber test, the vehicle secondary coolant loop was deserviced and coolant pump A was replaced, the system was leak tested and reserviced per MPS 185.
- Flight Checkout Simulation - After reservice of the vehicle coolant system, MPS 202 was initiated to verify that the bellows in the secondary loop reservoir module were properly positioned, and vehicle service was acceptable. SEDR D3-E75-1, Vol. II was performed to simulate flight checkout of the vehicle. This test manually activated the coolant loops and then the inverter select switches were placed in the command position for the remainder of the test. The coolant loops were maintained in a vehicle support configuration during the remainder of the simulated flight test. Following the test, MPS 181 was accomplished which inspected the 3-way latching solenoid valves, located on the suit and battery module, for pinched electrical conductors. The results were acceptable. As a result of problems in the suit cooling system MPS 215 was initiated which removed the primary and secondary coolant loop EVA H_2O heat exchangers from the suit and battery module. The primary and secondary coolant loops were deserviced and replacement heat exchangers installed. While the coolant loops were deserviced, the coolant filters were replaced by MPS 218. The system was then helium leak tested and reserviced with Coolanol per MPS 220. A final check of reservoir bellows position was performed per MPS 202.
- Leak Check - The ground coolant volume compensator installed in the FAS was leak checked with nitrogen per MPS 205. This portion of the coolant loop was not serviced with coolant fluid at St. Louis.

(2) ATM C&D/EREP Cooling System - The integrated testing of this system at St. Louis is shown in Figure 2.4-45. The ATM C&D/EREP Cooling System was operated to support Simulated Flight (D3-E75-1, Vol. I) and Altitude Chamber Test (D3-E73-1). After the altitude chamber test, the system was deserviced to lubricate "O" rings in the water filter quick disconnects and to mate the MDA portion of the loop to the AM. After final mate the entire system was leak tested per SEDR D3-E56-1, Vol. III and serviced per D3-F90-1, Vol. XIII to operate the water pumps for EMC purposes in support of SEDR D3-E75-1, Vol. II. After Simulated Flight Vol. II, the system was drained to replace EREP tape recorders and two water pumps. After component replacement, the system was leak tested per MPS 210 and final system servicing was performed prior to delivery to the launch site.

B. KSC Tests - Results of testing at KSC together with factory test results are presented in Figures 2.4-46 through 2.4-52. Launch site test requirements were specified in MDAC-E Report MDC E0122, Specification and Criteria at KSC for AM/MDA Test and Checkout Requirements; and KSC Report KS 2001 Test and Checkout Plan.

REQUIREMENTS		VERIFICATION				
		FACTORY		KSC		COMMENTS/REMARKS
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
1.0 GROUND COOLING						
1.1 FAS ground cooling loop loop leak test at 230 psig N_2.	10.1 in H_2O/hr MAX		N/A	KM-3000	Verified	
2.0 EQUIPMENT COOLING						
2.1 Pri and Sec pump operation (manual) - dual pump operation (A&B, B&C, C&A) and single pump operation (A, B, and C) with radiator flow in the normal and bypass modes.	a Pump inlet pressure 39 psia max. b Normal mode. Pump delta pressure 70 psid maximum (single pump) 170 psid maximum (dual pump) Flowrate 230 lb/hr minimum (single pump) 420 lb/hr minimum (dual pump) c. Bypass mode. Pump delta pressure 48 psid maximum (single pump) 115 psid maximum (dual pump) Flowrate 236 lb/hr minimum (single pump) 440 lb/hr minimum (dual pump)	03-E76-1	See Fig. 2.4-47 Verified	KM-0003	See Fig. 2.4-48 Normal Mode (Single Pump) Pmax=58psid (Dual Pump) Pmax=166 psid Normal (Single Pump) Fmin=26[illegible] lb/hr (Dual Pump) Fmin=4[illegible] lb/hr Bypass Mode (Single Pump) Pmax=40psid (Dual Pump) Pmax=115 psid Bypass Mode (Single Pump) Fmin=265 lb/hr Bypass Mode (Dual Pump) Fmin 496 lb/hr	DR AM1-07-0081 Faulty Sec Pump inlet press Xducer (0223) replaced & retested Single Pump OK Single pump - normal rad flow runs complete in KM-0003. Pmax C-2 within spec Dual Min. flow combination (B, C-2) satisfies spec Single Pump Flowrate for min.flow pump inverter comb (B-2) satisfies spec. Dual Min. flow combination (B, C-2) satisfies spec Single Pump/Inverter comb (C-3) within spec reqmts Dual Pump/Inverter comb. (B, C-2) was equal to max requirement 115 psid. Min. flow for Single Pump/Inverter comb (B-1, B-2) satisfies spec. Flow for minimum Invert Pump Combination (2&C) satisfies spec
	d. C&W (pri & sec) COOL FLOW & RES LO lights remain OUT during system operation		Verified		Verified	
	e Actual data exceeds the performance indicated on the system performance characteristic curves		Verified		Verified for all pump/inverter combinations run	
	f. Pump/Inverter TM events occur properly during system operation.		Verified		Verified	
2.2 Pri/Sec pump operation (DCS) - single pump operation (A, B, and C) (bypass mode)	a. Pump inlet pressure 39 psia max.		See Fig. 2.4-47		See Fig. 2.4-48 Max pump inlet pressure = 38 psia pump B, Inv 1, Pri	Pump inlet press was included in 10 out of 12 coolant pump inverter verif. steps run in bypass mode.
	b. Pump delta pressure 48 psid maximum (single pump)				Max pump delta pressure = 40 psid pump C Inv 3, Sec Loop	
	c Pump/Inverter TM events occur properly during system operation.				Verified for all pump/inverter combinations run	

FIGURE 2.4–46 COOLANT SYSTEM REQUIREMENT VERIFICATION (SHEET 1 OF 2)

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
2.3 Radiator bypass valve operations (Pri/Sec) (manual switching)	Proper TM event indication when valve is cycled. Visual indicator on valve indicates valve has cycled.	D3-E76-1	Verified	AM-0003	Verified	
2.4 Bypass valve operation						
2.4.1 Radiator bypass valve operation (Pri and Sec) (DCS switching)	Visual indicator on valve indicates valve has cycled.		Verified		Verified	Pri loop ΔP incr/decr of 14 psid Sec loop ΔP incr/decr of 15 psid
2.4.2 Suit umbilical sys 1 and 2 coolant flow switch operation	EVA CLNT FLOW light - ON when switch in EVA position. Light - OUT when switch in BYPASS position		Verified		Verified	
2.5 Coolant temperature control valve						
2.5.1 Condensing heat exchanger inlet (Pri and Sec)	TM indicates inlet temperature of 47 ±4°F		(P) 50.8°F (S) 50.4°F		Pri C209, C217 51°F max 49°F min Sec C210, C218 51°F max 50°F min	
2.5.2 Suit/Battery Cooling Module 47° valves (Pri and Sec)	Valve outlet temperature 43°F min		(P) 55°F (S) 55°F		Pri 56.1°F C281, C283 Sec 51.8°F C282, C284	
2.5.3 Suit/Battery Cooling Module 40° valves (Pri and Sec)	Valve outlet temperature: 35°F min		(P) 50°F (S) 50°F		Pri 45°F C273, C275 Sec 45°F C274, C276	Primary - verified by C273, C275 failed (DR AM 1-04-0005-0079) Secondary - verif. by C276, C274 failed (DR AM 1-04-0089)
2.5.4 Suit/Battery Cooling Module Outlet 47° valves (Pri and Sec)	C&W lights remain OUT (PRI COOL TEMP HIGH AND LOW, SEC COOL TEMP HIGH AND LOW)	D3-E72-1	Verified		Verified	
2.5.5 Pri and Sec Coolant Condensing H/X A&B valve operation (manual)	Valves move freely from full open to full close. (8 ±1 turns full travel.)	D3-N46-1	Verified		Verified	
2.6 Coolant System Leak Test	a. Average system leak rate shall not exceed 0.034 psi/week corrected to 70°F (system serviced)	MPS AVE 86	Verified	TPS AM1-07-0004	Verified	
	b. No visible leakage allowed from GSE 530 and 531 during 30 minute monitor period	Gauges remained in system during shipment	Verified	To be verified after gauges are removed from system	Verified	
2.7 Automatic pump switchover Test PRI to SEC and SEC to PRI.	Qualitative verification of decrease in pump delta pressure	D3-N70	Verified	AM-0003	Verified	

FIGURE 2.4-46 COOLANT SYSTEM REQUIREMENT VERIFICATION (SHEET 2 OF 2)

SEDR	INVERTER/ PUMP COMBINATION	COOLANT SYSTEM (PRI/SEC)	52-83700-729 RADIATOR BYPASS VALVE POSITION (NORMAL/BYPASS)	PUMP ΔP (PSID)	SYSTEM FLOW (LB/HR)	PUMP * INLET PRESSURE (PSIA)
D3-E76-1	1A	PRI	BYPASS	35.7	267	34.4
	1A	SEC		37.7	262	34.2
	1A	PRI		35.7	272	34.0
	1A	SEC		36.8	264	33.5
D3-E76-1	1A	PRI	NORMAL	56.5	265	33.6
	2B	↓		52.9	263	-
	3C			56.5	265	-
	1A	SEC		57.7	259	33.5
	2B	↓		56.8	256	-
	3C			58.6	265	-
D3-E76-1	1AB	PRI	NORMAL	155.4	481	32.8
	2BC	↓		149.1	477	32.4
	3CA			155.4	492	32.4
	1AB	SEC		158.7	476	32.3
	2BC	↓		159.6	480	32.3
	3CA			158.7	480	32.3

* VEHICLE HORIZONTAL - ST. LOUIS TESTS (41 PSIA MAX)

FIGURE 2.4-47 COOLANT SYSTEM PUMP/INVERTER FLOW TESTS – MDAC-E

PROCEDURE	INV/PUMP COMBINATION	SYSTEM		RAD FLOW VALVE		PUMP INLET TEMP (°F)	PUMP INLET PRESS (PSIA)	PUMP DELTA PRESS (PSID)	FLOWRATE (lb/hr)
		PRI	SEC	NORM	BY-PASS				
KM-0003									
Seq 34-070	1A	X		X		66	N/R	54	265
	3A	X		X					
	1B	X		X					
066	2B	X		X		66	N/R	54	261
060	2C	X		X		66	30	57	263
	3C	X		X					
004	1A	X			X	68	30	37	270
011	3A	X			X	68	30	38	270
030	1B	X			X	67	38	38	265
025	2B	X			X	68	30	38	265
050	2C	X			X	66	30	38	267
045	3C	X			X	66	N/R	38	[illegible]
110	1AB	X		X		66	N/R	158	481
105	2BC	X		X		66	N/R	154	472
100	3CA	X		X		66	29	157	485
038	1AB	X			X	66	29	109	501
092	2BC	X			X	66	29	107	496
098	3CA	X			X	66	29	111	505
083	1A		X	X		66	N/R	55	263
	3A		X	X					
	1B		X	X					
078	2B		X	X		66	N/R	55	265
073	2C		X	X		66	29	58	272
	3C		X	X					
004	1A		X		X	69	30	38	267
017	3A		X		X	68	29	38	267
036	1B		X		X	67	29	38	267
025	2B		X		X	68	29	38	267
056	2C		X		X	66	29	39	276
045	3C		X		X	67	N/R	40	276
141	1AB		X	X		68	N/R	162	488
136	2BC		X	X		68	N/R	166	494
131	3CA		X	X		68	28	158	488
118	1AB		X		X	67	28	112	508
123	2BC		X		X	68	28	115	514
129	3CA		X		X	68	28	114	510

FIGURE 2.4-48 COOLANT SYSTEM PUMP/INVERTER FLOW TESTS – KSC

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
1. System water flush	Fill tank with water PS 13240 Type I. Withdraw 4.0 ±0.2 lb prior to mechanical closeout.		N/A	KS-0016 TPS AM1-08-225	Verified	System reservice after H_2O tank change-out.
2. Pump operation - Operate individual pumps (A, B, C), via manual switching.	LO ΔP light remains OUT during system operation. Pump delta pressure: 23.2 psid maximum	D3-E75-1 Vol. II ↓	Verified a) 9.8 psid b) 6.08 psid c) 9.28 psid	AM-0003 ↓ KS-0045	Verified ΔPA=9.2psid ΔPB=8.5psid ΔPC=7.8psid	Pump ΔP verified
	Pump flow. 220 lb/hr minimum		a) 293 lb/hr b) -- c) --		ΔA=300 lb/hr ΔB=297 lb/hr ΔC=279 lb/hr See Below	Flow reqmts verified.

MEASUREMENT

PUMP	TEMP INLET (°F)	TEMP OUTLET (°F)	FLOW (LB/HR)	ΔP (PSID)	
A	63.9	73.8	UUU	9.6	Initial Test (9 Feb 73)
B	61.8	71.2	298	9.0	
C	61.3	69.8	281	8.2	
A	67	72	UUU	11.2	Retest following flight servicing (15 Mar 73) H_2O tank was replaced and system reserviced after this test
B	65.8	71	297	10.8	
C	65.6	70.7	281	9.8	

UUU - OFF SCALE HIGH

FIGURE 2.4-49 ATM C&D PANEL/EREP H_2O COOLING SYSTEM REQUIREMENT VERIFICATION

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
1.0 RADIATOR SURFACES						
1.1 Cleanliness	Maintain cleanliness such that the average and local solar absorptivity are acceptable per MDC PS 13614.	PWO *	Verified	KS-0016	Verified	
1.2 Absorptivity - Make 12 measurements on each AM/MDA radiator panel with a portable solar reflectometer. The measurements shall consist of 3 sets of 4 taken along the length of each panel	Solar absorptivity of of any one measurement shall not exceed 0.20. The average of the measurement on any panel shall not exceed 0.19.				See Fig. 2.4-51	
2.0 THERMAL/METEROID CURTAIN GOLD COATED SURFACES						
2.1 Cleanliness	Maintain cleanliness such that the average and local surface emissivity was acceptable per MDC PS 14205		See Fig. 2.4-52	KS-0016 KS-0007	Verified Acceptable	
2.2 Inspection - Prior to installation of curtains, visually inspect gold surfaces for contamination.	No contamination allowed. If contamination is apparent, make emissivity measurements to assure a 0.1 maximum average emissivity for the curtain. Average emission shall be determined by using 6 random measurements per square yard.		Verified Acceptable	KS-0016 KS-0007	Verified Acceptable	
3.0 GOLD TAPED SURFACES						
3.1 Surface inspection - Visually inspected exposed surfaces on gold taped parts and equipment for contamination (smudges, dust, wear, etc.).	No contamination allowed. If contamination exists, make emissivity checks per MDC PS 14100 to ensure 0.05 maximum emissivity and/or repair per MDC PS 14100.		Verified Acceptable	KS-0016	Verified Acceptable	
3.2 Emissivity checks on surfaces identified in MDC PS 14100, Paragraph 7.2.3.2.	Acceptable per MDC PS 14100.					* Production Work Order

FIGURE 2.4-50 THERMAL CONTROL COATING REQUIREMENT VERIFICATION

RADIATOR PANEL NUMBER	GIER-DUNKLE SOLAR REFLECTOMETER MEASUREMENTS MEASUREMENT LOCATION NUMBER												ρ_s AVG	ρ_s AVG*	α_s AVG
61A310264	1	2	3	4	5	6	7	8	9	10	11	12			
-251	.88	.88	.87	.88	.88	.88	.86	.88	.88	.88	.88	.86	.88	.86	.14
-249	.89	.87	.89	.86	.87	.88	.88	.88	.88	.87	.88	.86	.88	.86	.14
-1	.89	.89	.90	.88	.90	.89	.90	.89	.89	.89	.89	.90	.89	.87	.13
61A310223															
-3	.89	.87	.88	.86	.88	.86	.87	.88	.87	.86	.88	.88	.87	.85	.15
-5	.90	.89	.89	.89	.90	.88	.89	.89	.90	.90	.89	.90	.89	.87	.13
-7	.89	.88	.86	.86	.87	.88	.85	.86	.86	.86	.88	.87	.87	.85	.15
-155	.88	.88	.88	.90	.89	.89	.89	.90	.89	.89	.90	.89	.89	.87	.13
61A310222															
-1	.88	.87	.83	.87	.84	.85	.87	.88	.88	.87	.87	.82	.86	.84	.16
-3	.88	.88	.87	.87	.88	.88	.88	.87	.87	.88	.88	.85	.87	.85	.15
-5	.89	.88	.88	.89	.86	.84	.86	.86	.88	.87	.88	.89	.87	.85	.15
-7	.89	.88	.87	.86	.88	.84	.86	.89	.88	.87	.88	.87	.87	.85	.15

*CORRECTED FOR GIER-DUNKLE/BECKMAN CORRELATION, Δ_{ρ_I} = 0.02.

FIGURE 2.4-51 AM U-1 RADIATOR SOLAR REFLECTANCE TEST RESULTS – KSC

DESCRIPTION	PROCEDURE	MEASUREMENT	COMMENTS/REMARKS
THERMAL/METEOROID CURTAIN	PRODUCTION WORK ORDER	AVERAGE EMISSIVITY	MAXIMUM AVG EMISSIVITY SHALL BE .1
61A310237-3		.066	
-5		.063	
-13		NOT AVAILABLE	
61A310245-1		.090	
-5		.070	
-27		NOT AVAILABLE	
61A310246-1		.061	
-2		.080	
-3		.080	
-4		.071	
-5		.058	
-6		.062	
-7		.067	
-8		.055	
-11		.051	
-15		.056	
-21		.050	
-31		.055	
-33		.058	
-43		.078	
-45		.050	
61A310247-1		.055	
-2		.054	
-7		.064	
-9		.072	
-10		.061	
-27		.049	
-28		.074	
-39		.041	
-39		.043	
-40		.055	
-40		.061	
-43		.060	
-45		.070	
-47		NOT AVAILABLE	
-48		.060	
-49		NOT AVAILABLE	
-51		NOT AVAILABLE	
-52		.060	
-65		.060	
-79		.070	
61A310263-7		.052	
-9		.055	
-53		NOT REQUIRED (NYLON)	
-57		.059	
-71		.080	
-73		.080	
61A310267-3		.056	
-5		.055	
61A310280-1		.090	
-3		.104	MRR A32AE25

FIGURE 2.4-52 THERMAL/METEOROID CURTAINS GOLD COATED SURFACE EMISSIVITY MEASURED AT MDAC-E

2.4.4.6 Mission Support

Mission support testing of the Skylab Test Unit (STU) and of the U-2 vehicle was conducted under simulated U-1 flight conditions.

A. STU Simulation - A summary of TCS tests performed utilizing the Skylab Test Unit is presented on the following pages. Test details as well as descriptions of STU are presented in the ECS/TCS Skylab Test Unit Report No. TR 061-068.99.

- TITLE — U-1 Cold Coolant Simulation

  BACKGROUND — U-1 coolant loop temperatures were low due to low heat load of 618 Btu/hr.

  OBJECTIVE — Determine coolant system operational characteristics under simulated low heat load conditions experienced in U-1 and verify proper operation of the SUS and ATM water pumps under these conditions, Reference TR 061-015-600.02.

  RESULTS — The coolant system and the water pumps operated satisfactorily, Reference TM 252:664.

- TITLE — ATM Pump Starting Transient Test

  BACKGROUND — U-1 ATM "Lo ΔP" light took longer than expected to go out after an ATM pump was turned on.

  OBJECTIVE — Determine time required to obtain "Lo ΔP" actuation pressures, TR 061-015-600.05.

  RESULT — The times to obtain minimum and maximum specified actuation pressures of 1.5 psid lower and 5.5 psid higher limit were approximately 1 and 3 seconds, Reference TM 252:660.

- TITLE 47° Temperature Control Valve "B" Tests

BACKGROUND During EVA activities from U-1, the 47° Temperature Control Valve "B" stuck with the cold inlet port open so that the outlet temperature decreased below the control temperature limit.

OBJECTIVE Determine if thermal shocks will cause the Temperature Control Valve (TCV) to stick, Reference TR 061-015-600.06.

Evaluate the effect on coolant system operation of turning two coolant pumps on when the radiator bypass valve is in the bypass position, Reference TR 061-015-600.07.

Evaluate operation of the TCV at various hot and cold inlet temperature profiles, Reference TR 061-015-600.12.

Evaluate operation of the TCV when subjected to contamination by introducing various sizes of metal particles, Reference TR 061-015-600.13.

Evaluate operation of the TCV when subjected to temperatures causing slow stroking of valve, Reference TR 061-015-600.14.

RESULTS Thermal shocks did not cause the TCV to stick. Outlet temperature recovery occurred within 5 minutes of the thermal shocks, Reference TM 252:675.

Turning two coolant pumps on when the radiator bypass valve was in the bypass position did not cause abnormal system operation, Reference TM 252:718.

Various hot and cold inlet temperatures did not adversely effect operation of the TCV. The outlet temperature remained within the specification allowable limits of 47° $\pm$2°F, Reference TM 252:696.

Injection of particulate contamination did eventually cause the TCV to jam. Subsequent attempts to free the valve were successful and the valve functioned properly, Reference TM 252:710.

Induced slow stroking of the TCV did not cause abnormal operation, Reference TM 252:682.

- TITLE Coolant Pump Shutdown and Startup Test

  BACKGROUND Coolant pump start-up characteristics were required to analyze the automatic switchover feature.

  OBJECTIVE Determine coolant pump shutdown and start-up characteristics, Reference TR 061-015-600.19.

  RESULTS Coolant pump flowrates and differential pressures were obtained during pump shutdown and start-up, Reference TM 252:657.

- TITLE Altitude Test of 2-Watt and 10-Watt Airlock Transmitters Without Cooling

  BACKGROUND Airlock 2-watt and 10-watt transmitters might not have active cooling available when operated.

  OBJECTIVE Determine the duty cycle required to keep the temperature of Airlock 2-watt and 10-watt transmitters below their maximum allowable operating temperatures when operated without coldplates, Reference TR 061-015-600.10.

  RESULTS The Airlock 2-watt and 10-watt transmitters were operated for 82 minutes and 32 minutes, respectively, without active cooling before attaining maximum temperature limits, Reference TM 252:723.

- TITLE Coolant Loop Simulated Leak Tests

  BACKGROUND U-1 primary loop coolant pressure was steadily decreasing.

  OBJECTIVES Simulate coolant loop leakage conditions and determine pump operating characteristics at low system pressures, Reference TR 061-015-600.16.

  Determine if the combined exposure effects of solar simulation (IR and UV) and Coolanol would produce visible color change of four thermal control materials used on Skylab:

(a) Z93 thermal control coating, (b) Thermal capacitor cover (gold surface), (c) thermal curtain (white fiberglass surface), and (d) Johns-Manville aluminum covered insulation (aluminum surface), Reference TR 061-015-600.17.

Determine visual characteristics of leaks at typical coolant system connectors as a possible aid in locating leaks on the U-1 vehicle, Reference TR 061-015-600.21.

Determine the pressure-temperature relationship of the coolant pump reservoir and a dual reservoir to aid in predicting U-1 coolant loop performance, Reference TR 061-015-600.23.

Determine coolant system reservoir module performance characteristics at simulated temperature conditions to enable refined evaluation of U-1 coolant system leakage effects, Reference TR 061-015-600.26.

Determine if a Coolanol leak into a water loop system could be detected by visual inspection of quick disconnects, Reference TR 061-015-600.27.

Simulate a Coolanol leak under the condensing heat exchanger module cover and determine quantity of Coolanol removed by the condensing heat exchanger-mole sieve installation, Reference TR 061-015-600.51.

Evaluate the effect of a combination of reduced temperatures and fitting torques on coolant leakage characteristics of six typical coolant line connections and a coolant valve, Reference TR 061-015-600.52.

RESULTS Coolant loop leakage was simulated and resulting profiles obtained of pump flowrate, inlet pressure and differential pressures, Reference TM 252:685.

The four thermal control materials exhibited no appreciable visual color changes after being exposed to Coolanol and solar simulation, Reference TM 252:678.

Coolanol leakage from typical coolant system connections showed up as a light stain on fiberglass tape, only, Reference TM 252:695.

The pressure-temperature relationship of the coolant pump reservoir and a dual reservoir was obtained, Reference TM 252:683.

Coolant system reservoir module performance characteristics were determined for 33 test conditions which included simulated thermal curtain temperatures, Airlock wall temperatures and coolant system heat loads, Reference TM 252:725.

Visual inspection of quick disconnects subjected to Coolanol/Type I fluid mixture (10%/90%) showed an oily appearance, Reference TM 252:689.

All of the evaporated Coolanol from a simulated leak under the condensing heat exchanger module was absorbed in the heat exchanger core, Reference TM 252:731.

Coolant leakage from six typical coolant line connections, subjected to a combination of reduced temperatures and subnormal fitting torques, was minimal on five connections while one connection exhibited gross leakage. Coolant leakage from a coolant valve was minimal, Reference TM 252:729.

- TITLE — Saddle Valve Tests

BACKGROUND — U-1 primary coolant loop might be serviced by utilizing a saddle valve applied onto one of the 5/16" diameter lines in the system.

OBJECTIVES Evaluate two saddle valves (WATSCO Inc. Types A1 and AP1) for possible use to reservice the primary coolant loop, Reference TR 061-015-600.20.

Conduct development tests on MDAC saddle valve, 61A830412-1, for possible use to reservice coolant loops, Reference TR 061-015-600.24.

Evaluate leakage from MDAC saddle valve when installed on 304 stainless steel tube of various outside diameters and wall thickness, Reference TR 061-015-600.25.

Evaluate leakage from MDAC saddle valve for ten different seals when installed on various size tubes, Reference TR 061-015-600.31.

Evaluate the RA34670BN curved seal and then the MSFC cylindrical seal while installed in the MDAC saddle valve, 61A830412-31, Reference TR 061-015-600.33.

Determine if the 61A830412-31 saddle valve with a 61A830412-35 MSFC cylindrical mold seal would leak when puncturing a 5/16 O.D. x .015 wall 304L stainless steel tube filled with Coolanol and pressurized at 5 and 25 psig, Reference TR 061-015-600.35.

Evaluate several secondary seal materials and determine the pressure sealing capabilities of each seal material while installed in the 61A830412 saddle valve, Reference TR 061-015-600.37.

Evaluate epoxy and fluorosilicone as secondary seals when installed in the 61A830412 saddle valves at Coolanol pressure of 100 psig, Reference TR 061-015-600.42.

RESULTS

Two WATSCO Inc. saddle valves were tested for puncture, external leakage and flowrate characteristics at various line pressures. There was no evidence of external leakage from either valve, Reference TM 252:691.

The MDAC saddle valve, 61A830412-1, was tested for puncture, external leakage and flowrate characteristics, Reference TM 252:686.

The MDAC saddle valve tested for leakage when installed on various sizes of tubes did not leak under any of the conditions evaluated, Reference TM 252:692.

Ten different seals were installed one at a time in MDAC saddle valve, applied on various size tubes, and tested for leakage, Reference TM 252:700.

Both the RA346708N curved seal and the MSFC cylindrical seal while installed in the MDAC saddle valve, 61A830412-31, passed all external leakage tests satisfactorily, Reference TM 252:719.

The 61A830412-31 saddle valve with a 61A830412-55 MSFC cylindrical molded seal was leak tested when puncturing a tube filled with Coolanol pressurized at 5 and 25 psig. No leaks occurred during either puncture test, Reference TM 252:693.

Several seal materials including Viton, fluorel, epoxy, and fluorosilicone were tested to determine secondary sealing capabilities while installed on the 61A830412 saddle valve. Viton and fluorel seals leaked while the epoxy and fluorosilicone seals did not leak at pressure of 5, 25, and 100 psig, Reference TM 252:694.

Epoxy and fluorosilicone seals were installed in 61A830412 saddle valves and leak tested with Coolanol at 100 psig. The valve with the epoxy seal leaked while the valve with the fluorosilicone seal completed a 7-day period without leaking, Reference TM 252:699.

- TITLE — Coolant System Reservicing Tests

BACKGROUND — Pressure loss in U-1 coolant system loops indicated coolant leakage. The primary coolant loop required reservicing.

OBJECTIVES — Verify the adequacy of the servicing hardware (including the modified OWS portable tank) and procedure developed for reservicing the coolant loop, Reference TR 061-015-600.29.

Verify the adequacy of the servicing hardware (including the CSM fuel tank) and procedure developed for reservicing the coolant loop, Reference TR 061-015-600.30.

Determine the effect of free gas injected into the coolant system from saddle valve installations on coolant pump performance, Reference TR 061-015-600.36.

Evaluate the SS4JBA NUPRO shutoff valve for use on the SL-4 coolant system servicing kit, Reference 061-015-600.38.

Evaluate sealing characteristics of a repair fixture 61A830421, to be used to seal a punctured line made by a saddle valve installation, Reference TR 061-015-600.39.

Evaluate procedures for removing a saddle valve from a coolant line and installing a 61A830421-23 repair seal, Reference TR 061-015-600.47.

Determine if free gas will cause coolant pumps to cavitate since free gas might be inadverently introduced into the coolant system during reservicing, Reference TR 061-015-600.43.

Obtain coolant loop temperature stabilization data to be used for refining thermal model to predict coolant loop performance during reservicing, Reference TR 061-015-600.46.

RESULTS

The servicing hardware, including the modified OWS portable tank and 61A830412-35 saddle valve, was used to reservice the coolant loop. Both hardware and reservicing procedures were satisfactory. There was no evidence of Coolanol leakage, Reference TM 252:724.

The servicing hardware, including the CSM fuel tank, was used in three test trials to reservice the coolant loop. Two attempts using different S/N 61A830412-61 saddle valves were terminated due to saddle valve seal leakage. Reservicing with an MSFC saddle valve, 20M33247, was successful, Reference TM 252:707.

Free gas, 3 SCC, was injected into the coolant system approximately 10 inches upstream of the coolant pump. No change in pump performance was apparent, Reference TM 252:684.

Two SS4JBA NUPRO valves were subjected to qualification tests for use as backup hardware. Both valves passed all tests except one unit failed the internal leakage test after being subjected to vibration, Reference TM 252:720.

The repair fixture, 61A830421, was installed on a tube previously punctured by a saddle valve installation and leak checked at 5, 35, 100 and 200 psig. No leaks were observed, Reference TM 252:687.

Procedures for removing a saddle valve from a coolant line and installing a 61A830421-23 repair seal were verified satisfactory. No Coolanol leakage occurred during the installation process or when one or both coolant pumps were operated. Reference TM 252:715.

Air in quantities of 4 in^3, 8 in^3, 16 in^3, 32 in^3, and 64 in^3 was injected into the coolant system. No change in pump performance was evident, Reference TM 252:702.

- TITLE Coolant Pumps Power Inverter Startup Tests

BACKGROUND The CPPI No. 2 circuit breaker opened when U-1 crew attempted to turn on second pump in secondary coolant loop.

OBJECTIVE Duplicate U-1 coolant loop conditions and determine procedure for second pump start-up, Reference TR 061-015-600.28.

RESULTS Test Inverter No. 1 was forced into current limiting seven times when pump No. 2 was started with pump No. 1 running. Whenever two pumps operation was required in a coolant loop, a simultaneous start of the pumps had to be performed.

- TITLE Voltage Regulator Thermal Vacuum Test

BACKGROUND Electrical power system operation without cooling might be required due to coolant loop problems during the SL-3 orbital storage period.

OBJECTIVE Determine if voltage regulator stabilization temperature is less than 140°F at low load, vacuum conditions with loss of the coolant loop, Reference TR 061-015-600.44.

RESULTS The voltage regulator maximum temperature attained during test was 128°F, Reference TM 257-107.

- TITLE Primary Oxygen Heat Exchanger Cold Gas Test

BACKGROUND Oxygen temperature at the oxygen heat exchanger inlet port might be as low as -125°F during U-1 repressurization period. The resulting negative heat load might have adverse effects on coolant system performance.

OBJECTIVE Evaluate coolant system operation at conditions simulating repressurization of Skylab, Reference TR 061-015-600.04.

RESULTS The coolant system operated satisfactorily at all test conditions, Reference TM 252:668.

- TITLE ATM Cooling Loop Tests

BACKGROUND U-1 ATM loop pump performance data indicated cyclic flowrates and variable noise levels.

OBJECTIVES Evaluate ATM water pump operation of various inlet pressures and pump differential pressures in an attempt to reproduce the gurgle-like sounds reported by the SL-3 mission crew, Reference TR 061-015-600.48.

Evaluate ATM pump start-up characteristics at various inlet pressures and pump differential pressures, Reference TR 061-015-600.50.

Determine if U-1 ATM pump cycle flowrate is caused by a high differential pressure which activates the relief valve, Reference TR 061-015-600.54.

Determine delta P of ATM loop filter cartridge returned from SL-3 mission and compare with PIA data, Reference TR 061-015-600.55.

Determine if the liquid-gas separator, normally used in the suit umbilical system could remove free air from the ATM loop, Reference TR 061-015-600.56.

Evaluate single and dual pump performance during both normal flow and blocked flow conditions in the ATM cooling loop, Reference TR 061-016-600.57.

Determine transient temperature characteristics of the ATM water pump (exposed to lab ambient environment) when 28 VDC is applied while the pump rotor is stalled, Reference TR 061-015-600.58.

Compare present performance data of two ATM water pumps with initial PIA data. Water pumps were installed in STU ATM loop for approximately seven months, Reference TR 061-015-600.59.

Determine water pump performance characteristics when quantities of air are injected into the ATM cooling loop, Reference TR 061-015-600.60.

Determine transient temperature characteristics of the ATM water pump (exposed to vacuum environment) when 28 VDC is applied while the pump rotor is stalled, Reference TR 061-015-600.61.

Determine ATM pump performance characteristics at voltage between 12 VDC and 28 VDC, Reference 061-015-600.62.

RESULTS The ATM water pump was operated at various inlet pressures and pump differential pressures in an attempt to reproduce the gurgle-like sounds reported by the SL-3 mission crew. None of the tests performed produced gurgle-like sounds, Reference TM 252:712.

ATM water pump start-up performance was determined for various pump inlet pressures and differential pressures. Performance was normal at all test conditions, Reference TM 252:711.

ATM pump flowrate excursions were experienced with pump differential pressure settings between 20 and 26 psid. Flowrates fluctuated as much as 40 lb/hr with no apparent cyclic pattern, Reference TM 252:728.

ATM loop filter cartridge returned from SL-3 mission was tested to determine delta P with water flow of 250 lb/hr. Delta P was 11 inches of water - no change from PIA test conducted during May 1972.

A liquid-gas separator was substituted for the ATM loop filter and removed 60 SCC of gas from the ATM loop before clogging, Reference TM 252:740.

Single and dual pump performance data was determined for both normal flow and blocked flow conditions in the ATM loop. Single and dual pump flowrate and delta P for normal flow was 275 lb/hr at 13.5 psid and 390 lb/hr at 25 psid, respectively. Pump delta P was 30.2 psid for the blocked line condition and single or dual pump operation, Reference TM 252:736.

Transient temperature profiles of the ATM pump motor housing and the rotor housing were determined for the condition of 28 VDC power on while pump rotor is stalled. ATM pump was exposed to lab ambient environment, Reference TM 252:737.

Comparison of two ATM water pumps current performance data with initial PIA data showed no degradation. Water pumps were installed in STU system for approximately seven months, Reference TM 252:741.

Various quantities of air were introduced into the ATM cooling loop to determine effects on pump performance. The largest amount was 130 cubic inches. This amount of air caused the flowmeter instrumentation to register zero flow for a duration of 116 seconds. Normal pump noise was greatly reduced during this period, Reference TM 252:744.

Transient temperature profiles of the ATM pump motor housing and the rotor housing were determined for the condition of 28 VDC power on while pump rotor was stalled. ATM pump was exposed to vacuum environment, Reference TM 252:745.

ATM pump performance data was determined for voltages between 12 VDC and 28 VDC. Data included flowrate, delta P, and current, Reference TM 252:746.

• TITLE Stowage Test of the 61A830416-1 Servicing Hose Assembly

BACKGROUND The servicing hose assembly was to be launched and stowed in the serviced condition. Temperature increases might cause excessive pressurization which could damage the gauge.

OBJECTIVE Determine pressure increases in the 61A830416-1 servicing hose assembly due to elevated temperature after hose is serviced with coolant fluid, Reference TR 061-015-600.45.

RESULTS Hose pressure increased from 0 psig to 50 psig when the hose temperature was increased from 86°F to 104°F, Reference TM 252:726.

• TITLE N_2 Flowrate Through 61A830355-13 and 61A830356-3 Servicing Hose Assemblies

BACKGROUND SL-4 coolant loop reservicing procedure included an N_2 leak check of the saddle valve installation prior to coolant line puncture. Equipment included the 61A830355-13 hose assembly which required purging with N_2 to remove any residual water before using the assembly to leak check the saddle valv .

OBJECTIVE Determine flowrate through the 61 30356-3 and 61A830355-13 hose assemblies with N_2 pressure at 40 psia. Also, determine the relief valve cracking pressure, Reference TR 061-015-600.41.

RESULTS The N_2 flowrate was 0.5 lb/hr at 40 psia inlet pressure. Relief valve cracking pressure was approximately 18 psia, Reference TM 252:690.

B. U-2 Testing - A summary of TCS test activity performed utilizing the U-2 vehicle in support of the U-1 mission is presented on the following pages:

- PROBLEM - During early mission unmanned operations, an automatic switchover (Reference AR 31) occurred from the primary coolant loop to the secondary loop. Since the vehicle was oriented such that the secondary loop low temp sensors were below the trip point and there was no indication of a problem with the primary loop, a crossed sensor was a possibility.

  SUPPORT - Researching the various tests performed on U-1 at St. Louis and KSC showed the sensors were checked at the module level and after installation on the vehicle. Location of the primary and secondary sensors were checked versus wire bundle installation and tubing installation which precluded

- PROBLEM - During activation of the ATM C&D panel and EREP cooling loop, the crew noted that it took a long time for the low delta P light to go out (20 seconds).

  SUPPORT - A test was performed on U-2 to determine if this time was normal. Each ATM loop pump was activated and in each case the light went out in 1 to 2 seconds. Data from the U-1 altitude chamber test was also researched and the time to achieve a delta P high enough (5 psid) to open the low delta P switch was 1 to 2 seconds. All information indicated that the 20 seconds experienced on SL 1/2 was not normal but all other aspects of the system were acceptable.

- PROBLEM - Problem occurred in the operation of the temperature control valve (TCV) in the primary and secondary coolant loops.

  SUPPORT - Steps were taken to determine cause of problem and methods to alleviate the situation. Laboratory effort was supported in building a test setup to check out a valve under the conditions experienced on SL 1/2. A 52-83700-729 radiator bypass valve was cycled through 5383 on-off cycles to demonstrate its capability to be cycled during the mission as a means to control coolant loop temperatures if the temperature control valves remained inoperative. A procedure was written and supplied to the laboratories to perform a vibration cleanliness test on a 52-83700-1205 heat exchanger to determine if contamination from this heat exchanger could have lodged in the control valve causing the initial problem.

- PROBLEM - Possibility of low outlet temperature required contingency work-around plan for SL-3 and SL-4 missions.

  SUPPORT - Evaluated another method to add heat to the coolant system, i.e., the build-up and check-out of an electrical water heater. Information was supplied to NASA and procedures were written to check out the heater and associated wire bundles. A test was performed on U-2 with the heater installed in the SUS loop to determine heat loads that could be transmitted to the coolant loop and also to verify the overtemperature switches in the heater.

- PROBLEM - Coolanol System Leakage - Primary loop low level light came on and secondary loop pump inlet pressure decreased.

  SUPPORT - The U-2 vehicle was utilized to perform a series of tests to evaluate indicated leakage condition:

  (a) Determined pressure profiles of Coolanol reservoirs versus temperature as fluid was extracted from the loop in 50 in^3 increments. This verified that cold reservoir temperatures (~55°F) would cause a lower pump inlet pressure than the initial servicing pressure when the reservoirs are approximately 75% full. As fluid was extracted from the system, the temperature effects were reduced. This verified that the primary loop was losing fluid and the reservoirs were nearly empty and that the secondary loop reservoirs were still over 50% full.

  (b) Determined minimum pump inlet pressure that would cause apparent pump cavitation (indicated by pump noise and drop in flowrate) to determine best time to shut off primary pump to prevent damage.

  (c) Supported activity to determine accessibility of external GSE valves as a method to reservice coolant loops.

  (d) Performed coolant reservicing on U-2 using a saddle valve on an internal coolant line and equipment identical to flight equipment to be utilized on the SL-4 mission (SL-4 flight crew).

2.4.5 Mission Performance

The Airlock Module TCS satisfactorily performed all required functions relating to active/passive thermal control of structure, systems, and equipment. The active coolant system provided cooling for interfacing systems (gas system O_2 heat exchanger, atmospheric control system heat exchangers, ATM C&D Panel/EREP cooling system heat exchanger, and suit cooling systems heat exchangers) and temperature control for coldplate mounted electrical/electronic equipment. Radiator/thermal capacitor rejection of heat from the active cooling system was normal throughout all phases of the mission. Flow of temperature controlled water to the MDA via the ATM C&D Panel/EREP cooling system permitted normal temperature control of associated equipment. Active heating of Airlock module walls, mole sieve exhaust ducts, and condensate system overboard vents was provided as required by electrical heaters. The overall vehicle thermal balance resulted in acceptable temperatures on passively controlled structure and components.

2.4.5.1 Payload Shroud

The Payload Shroud provided adequate protection for enclosed modules from aerodynamic heating experienced during ascent. Actual temperatures experienced during ascent are not available since no instrumentation was installed.

2.4.5.2 Coolant System

A. Coolant Pump Subsystem

(1) Pumps and Inverters - Coolant flowrates were normal during prelaunch. Prior to SL-2 the automatic switchover system switched from the primary to secondary loop on two occasions. It was concluded that the two switchovers were due to a faulty primary loop sensor in sensor group 1. The primary loop was successfully operated for the remainder of the mission with only automatic switchover sensor group 2 enabled.

Operation of inverters and pumps was normal except on two occasions. Several hours after activating inverter 1 and pump A in the secondary loop on DOY 149, the inverter 1 circuit breaker opened. Available data indicated normal operation at the time of occurrence. No further attempt was made to operate this pump or inverter until after SL-4 when postmission testing was accomplished. Results indicated that pump A operation was normal when powered by inverter 3; however, neither pump A nor B would operate when connected to inverter 1. The problem on DOY 149 was therefore attributed to a failure of inverter 1 or associated circuitry. On DOY 233, while operating on inverter 2, pump B in the secondary loop, the inverter 2 circuit breaker opened when pump C was commanded on. No further problem was encountered after the inverter/pumps were turned off, the circuit breaker closed, pumps B and C both turned on, and then inverter 2 turned on. (The correct procedure for initiating 2-pump operation.)

Coolant flowrates for the various inverter/pump combinations utilized throughout the mission are summarized in Figure 2.4-53. The flowrates were as expected and did not decrease with time of operation.

(2) Reservoirs - Pump inlet pressure in the primary loop decreased slowly with time as seen in Figure 2.4-54 to a level of 20.5 psia on DOY 217 at which time a reservoir low level indication was obtained. The loss of fluid from the loop is indicated by Figure 2.4-55 and continued until pump shutdown was required on DOY 235. A similar decrease in secondary loop pump inlet pressure is shown in Figure 2.4-54 and loss of fluid from this loop is indicated by Figure 2.4-55. A reservoir low level indication in the secondary loop was obtained on DOY 039 before the SL-4 crew departed. Although the SL-3 crew inspected the interior of the vehicle, no evidence of Coolanol leakage was identified. Also, pump inlet pressure in the primary loop, following depletion of coolant in reservoirs, eventually stabilized at a level of 2.5-3.0 psia, indicating that the leak was outside of the cabin. However, postflight analysis of CO_2 filter cartridges utilized during SL-3 indicated a possible trace of Coolanol (2 ppm) in the cartridge material. Location of Coolanol leaks cannot be stated with any degree of certainty since there were two loops involved and each loop may have had more than one leak.

COOLANT LOOP	INVERTER	PUMP(S)	FLOWRATE (LB/HR)
PRIMARY	1	A	270
		B	△1
		AB	510 △2
	2	B	270
		C	△1
		BC	510 △2
	3	C	△1
		A	△1
		AC	△1
SECONDARY	1	A	270
		B	△1
		AB	520
	2	B	270
		C	△1
		BC	5[illegible]5
	3	C	275
		A	270 △2
		AC	△1

NOTES: △1 Inverter/pump(s) combination not utilized

△2 Estimated from TCV-3 hot inlet flow measurement since TCV-B outlet flowmeter not available at time of designated operation

FIGURE 2.4-53 COOLANT FLOWRATES

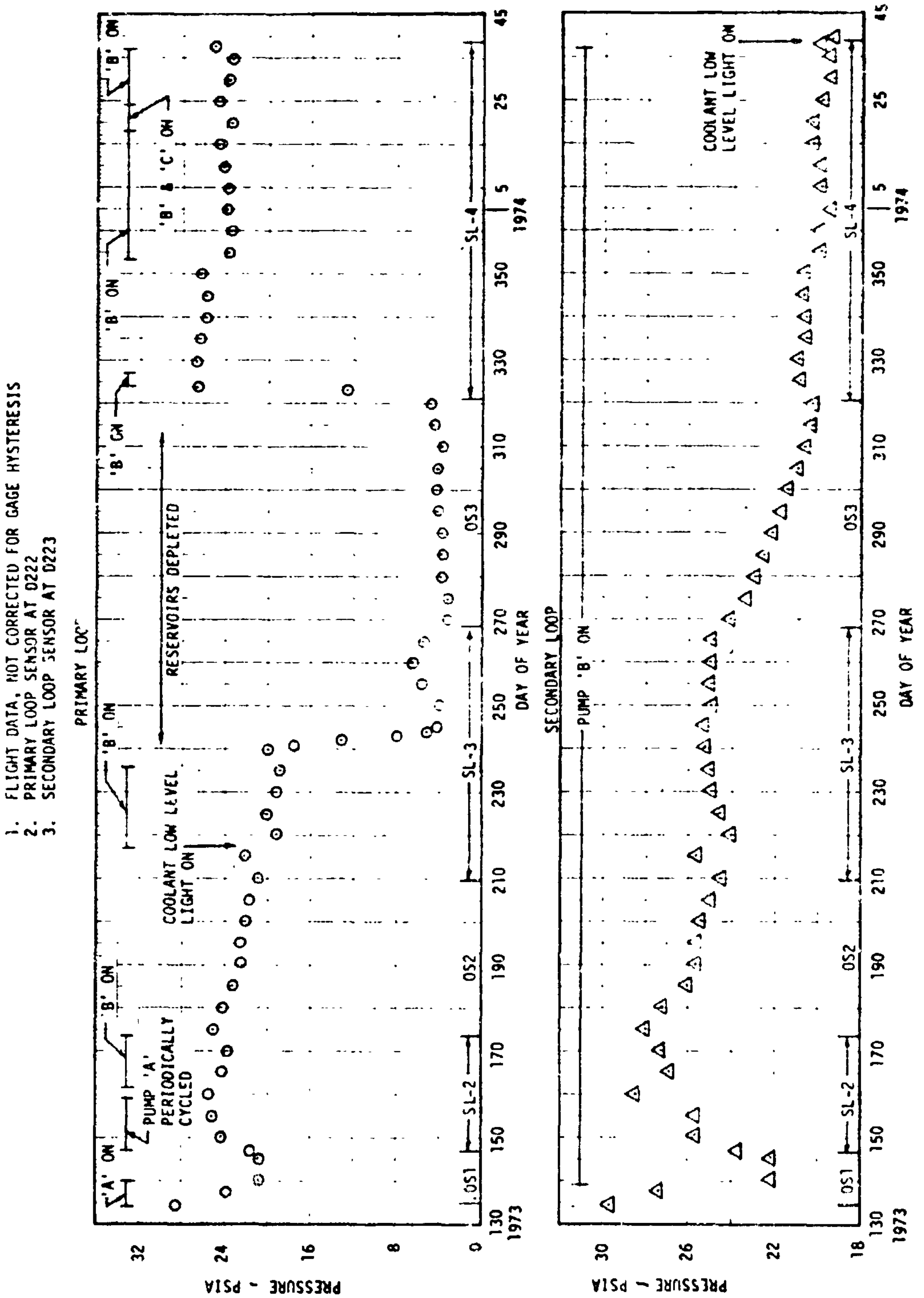


FIGURE 2.4-54 COOLANT SYSTEM PUMP INLET PRESSURES

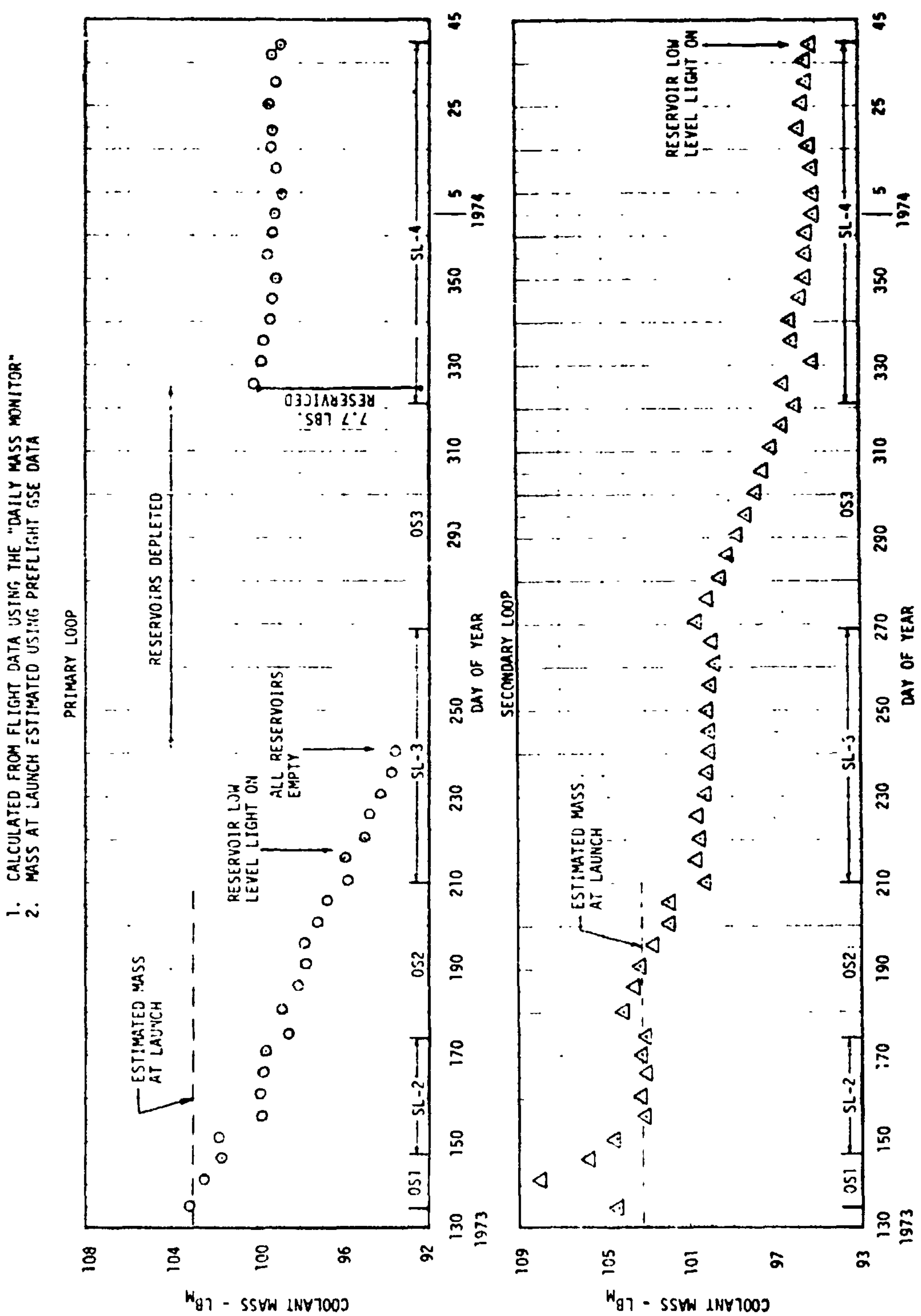


FIGURE 2.4-55 COOLANT SYSTEM COOLANOL MASS

In order to maximize the probability of adequate coolant loop operation, procedures were developed and ground testing performed to provide a means of extracting Coolanol from the backup refrigeration loop in the OWS and introducing the fluid into the AM primary loop. However, such action was not required on SL-3 and hardware was provided on SL-4 along with applicable procedures to perform the reservicing operation. A discussion of the reservicing may be found in Section 2.4.5.2(E).

B. Heat Loads - During the initial ten days of the mission, both the active cooling system and passive temperature control system were exposed to heat loads well below design levels as a result of losing the OWS meteroid shield and one solar array system wing during ascent, the inability to successfully deploy the remaining wing, and an off-nominal vehicle orientation. This situation resulted in delayed activation of AM power conditioning groups with the loss of battery module waste heat, excessive heat leaks to abnormally cold structure obtained with the pitch-up attitude being flown to minimize OWS temperatures, and the obvious need to conserve electrical power. Throughout this period, coolant flow through the radiator was at a minimum and approached zero on several occasions. However, loop operation was maintained by increasing heat loads via DCS commanding of instrumentation system components when absolutely necessary. Also, it was recommended that purging of the cluster be accomplished with N_2 rather than O_2 to avoid unnecessary heat removal from the loop via the O_2 heat exchanger. An N_2 purge was performed. Testing on the ECS System Test Unit (STU) indicated that the cluster could be pressurized with O_2 in preparation for the first manned mission if multiple pumps were operated to add heat to the loop. The cluster was pressurized successfully with two pumps operating in the secondary coolant loop.

Following SL-2 activation and return to solar inertial orientation, coolant loop heat loads increased and normal external loads were restored. Internal heat loads remained somewhat below design load levels as a result of the lower battery module waste heat levels associated with power inputs from only one OWS solar array wing instead of the normal two as discussed in Paragraph 2.4.3.4. Total internal coolant loop heat loads during the various mission phases are presented in Figure 2.4-56. Maximum loads approached 11,000 Btu/hr during the period of high Beta angles when three coolant pumps were operated near the end of SL-4.

C. Heat Sink

(1) Ground Cooling - Operation of the AM primary and secondary coolant loops for prelaunch cooling was normal with freezing of the thermal capacitors complete at approximately 14 hours prior to lift-off. After this time, coolant temperatures at the thermal capacitor outlet remained between -7°F and -11°F until termination of ground cooling approximately ten minutes prior to lift-off. All other loop temperatures were nominal and well within redline limits.

During ascent and prior to radiator cooldown during the initial orbit, coolant temperatures were maintained at desired levels by the thermal capacitors as shown in Figure 2.4-57. All temperature control valves (TCV) remained in control throughout this period. Capacitor skin temperatures during discharge and subsequent refreezing are plotted in Figure 2.4-58.

(2) Radiator/Capacitor - Since heat loads were slightly lower than anticipated, the full capability of the radiator/thermal capacitor system was not required during normal operation with a solar inertial vehicle orientation. Figure 2.4-59 through 2.4-61 present typical inlet and outlet temperatures for a single orbit during each of the manned missions. Data presented is for similar heat loads, Beta angles and operating configuration. No significant degradation is indicated although crew reports and D024 Experiment data have identified contamination and discoloration of painted surfaces exposed to the sun. Although thermal capacitors remained frozen, their heat sink effect on modulation of radiator outlet temperatures can be seen in the above figures.

Maximum radiator outlet temperatures were obtained during those periods when the vehicle was maneuvered out of the solar inertial

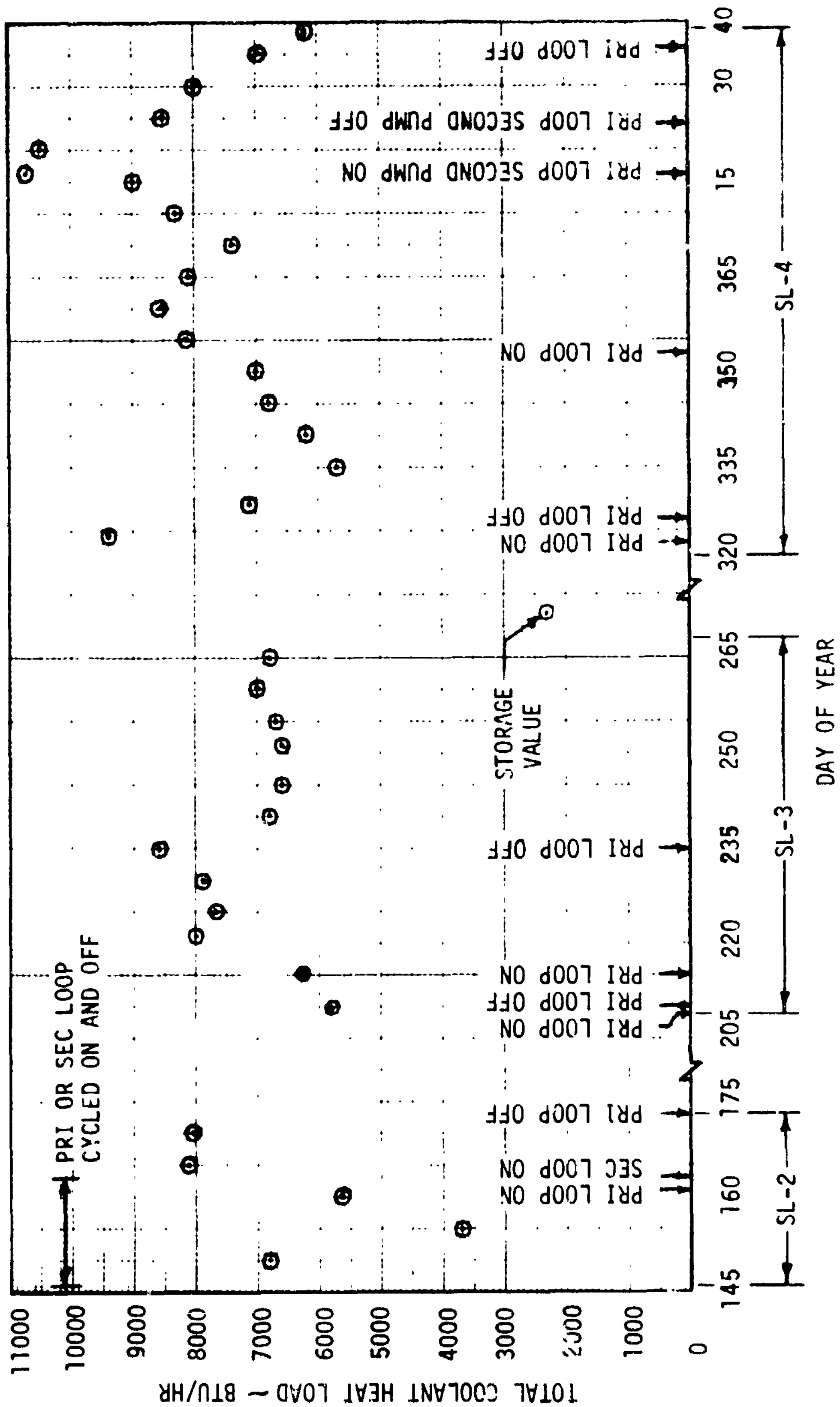


FIGURE 2.4-56 COOLANT LOOP HEAT LOADS

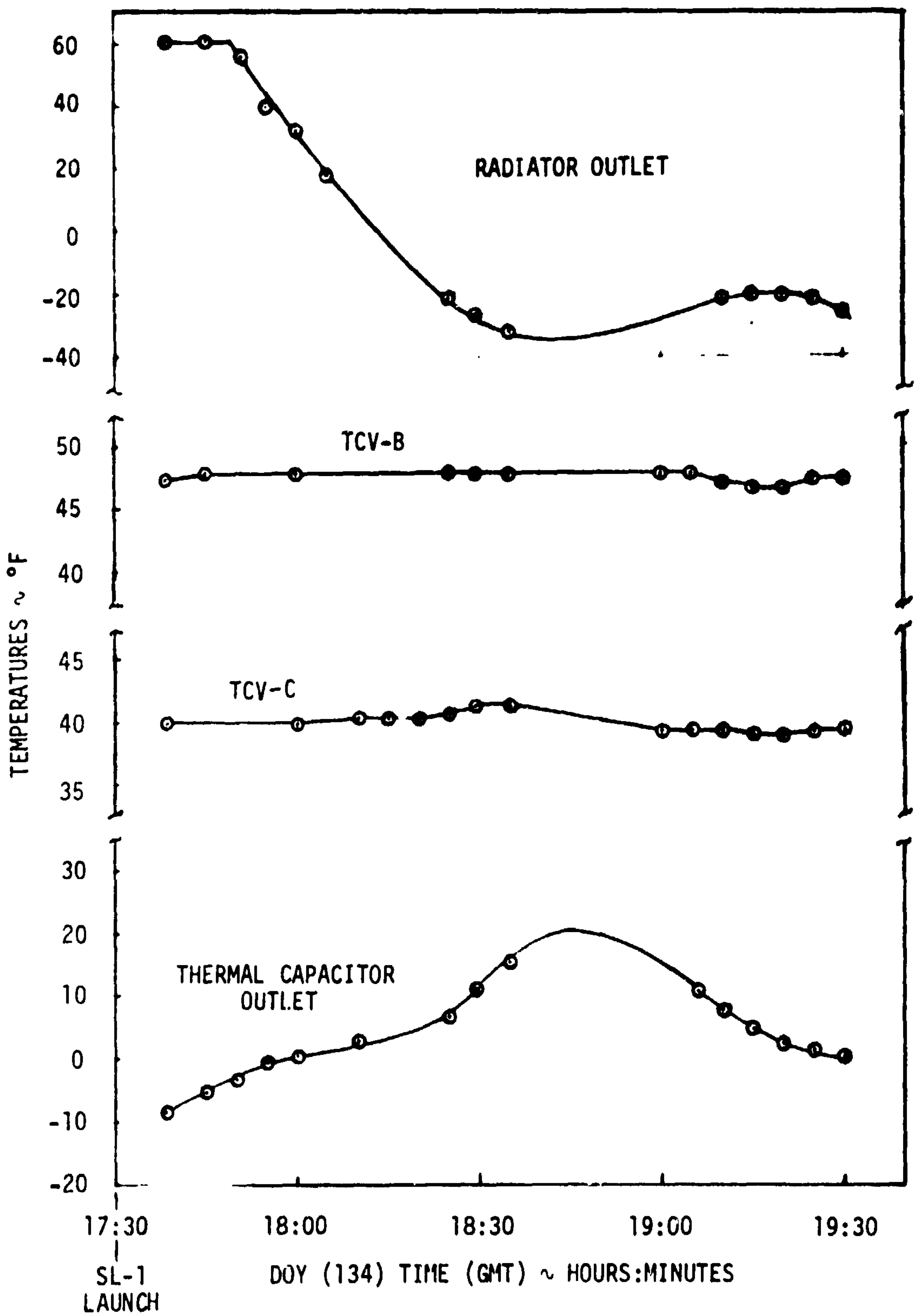


FIGURE 2.4-57 COOLANT TEMPERATURES DURING RADIATOR COOLDOWN

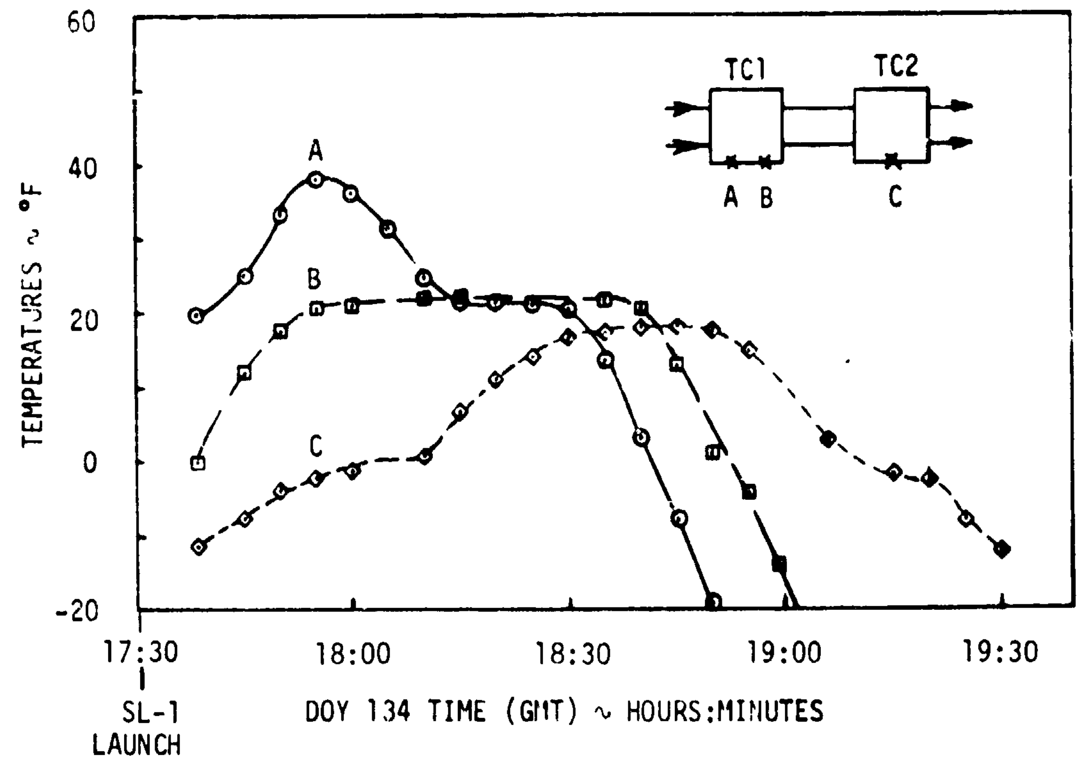


FIGURE 2.4-58 THERMAL CAPACITOR PERFORMANCE

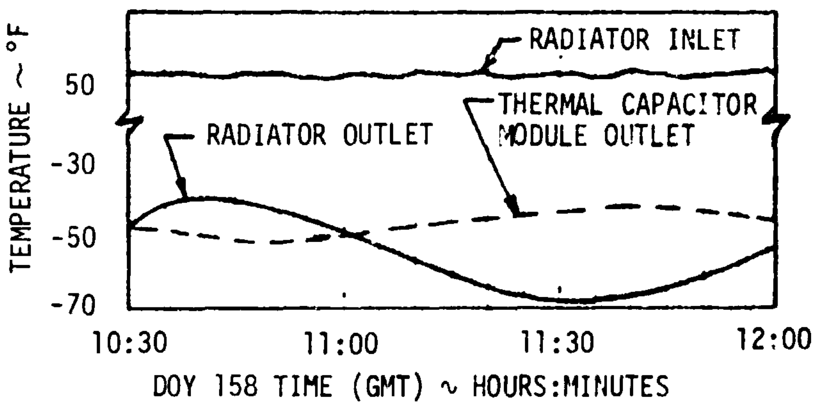


FIGURE 2.4-59 SL-2 RADIATOR/THERMAL CAPACITOR TEMPERATURES

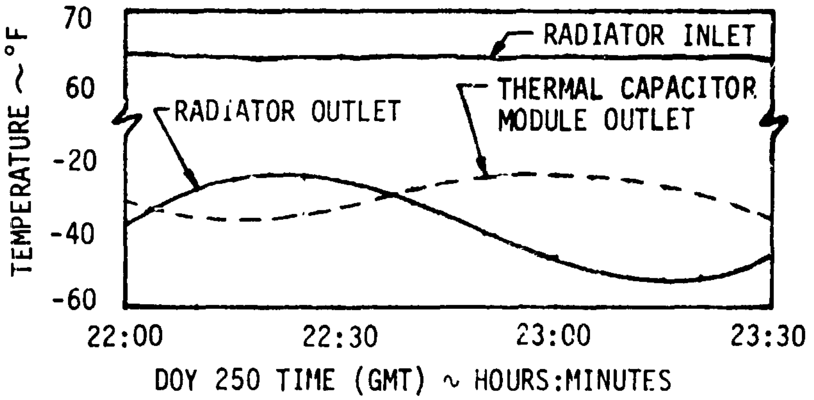


FIGURE 2.4-60 SL-3 RADIATOR/THERMAL CAPACITOR TEMPERATURES

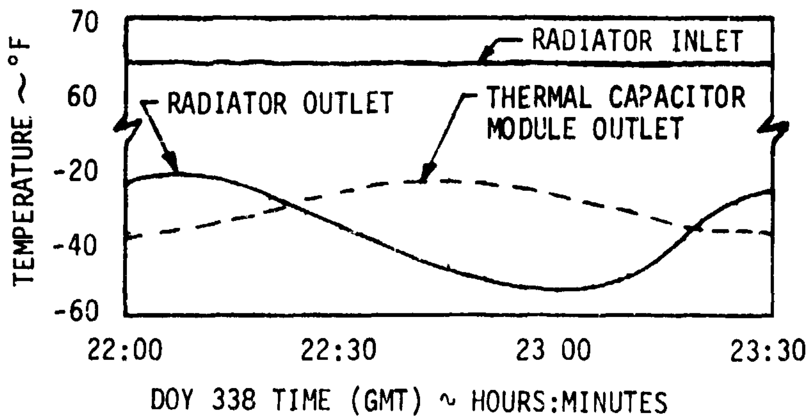


FIGURE 2.4-61 SL-4 RADIATOR/THERMAL CAPACITOR TEMPERATURES

attitude for EREP operations or viewing of the Comet Kohoutek. Figures 2.4-62 and 2.4-63 plot radiator/thermal capacitor inlet and outlet temperatures for the most severe conditions encountered during these maneuvers. Although EREP maneuvers were not originally planned at Beta angles as high as 65 deg, this maneuver was accomplished as the result of real time mission planning. It can be noted that the thermal capacitors were completely melted for this EREP maneuver at high Beta angle; however, all thermal control functions were satisfactorily maintained during and following the maneuver. Minimum radiator and thermal capacitor outlet temperatures recorded during the mission were -97°F and -58°F, respectively.

D. Temperature Control - The active cooling system was able to control coolant loop temperatures within normal ranges at all locations, although heat loads were well below design levels throughout the early portion of the mission. Dew point was controlled within tolerance as discussed in Paragraph 2.5.4 and adequate suit temperature control provided as described in Paragraph 2.6.4.

When the heat exchanger coolant flow valve in the primary loop was placed in the EVA position for EVA on DOY 158, lower than normal temperatures were obtained. Temperature and flowrate data indicate this was due to sticking of the downstream 47°F temperature control valve (TCV-B) in the primary loop. TCV-B in the secondary loop was also stuck in an intermediate position when the EVA was terminated. These problems are believed to have been caused by particles flushed from the EVA heat exchanger into the TCV-B cold inlet. The valve conditions were later corrected by turning off the loop, permitting the valve to warm to temperatures above 50°F thereby fully opening the cold inlet port, and then flushing particles through the valve with coolant flow from two pumps. Subsequently, similar valve conditions and problem corrections were achieved during ground testing which simulated the flight situation.

Following the above situation, active cooling system support of EVA was accomplished leaving the heat exchanger coolant flow valve in the BYPASS position. Due to lack of confidence that TCV-B in the secondary

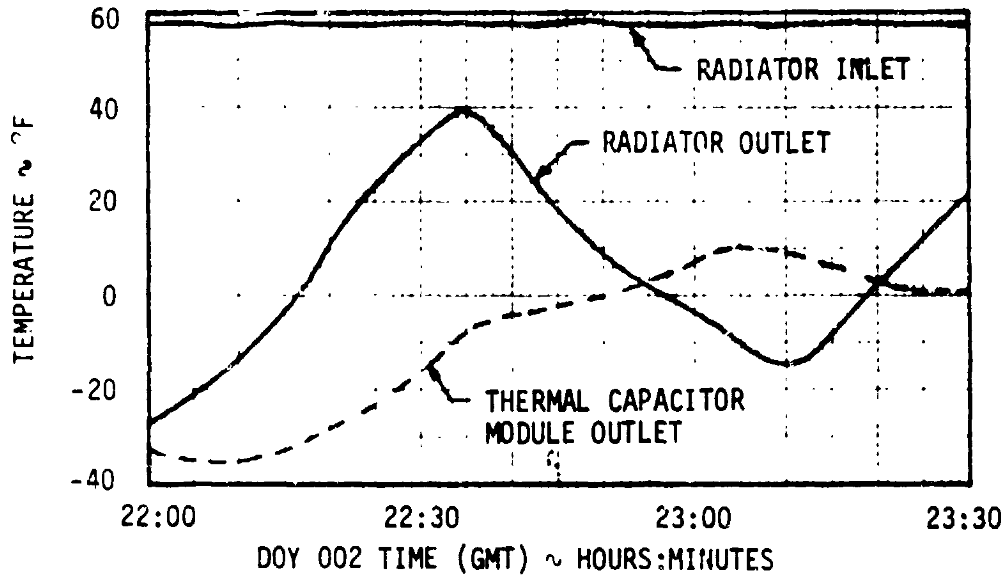


FIGURE 2.4-62 RADIATOR/THERMAL CAPACITOR TEMPERATURES DURING A KOHOUTEK VIEWING MANEUVER

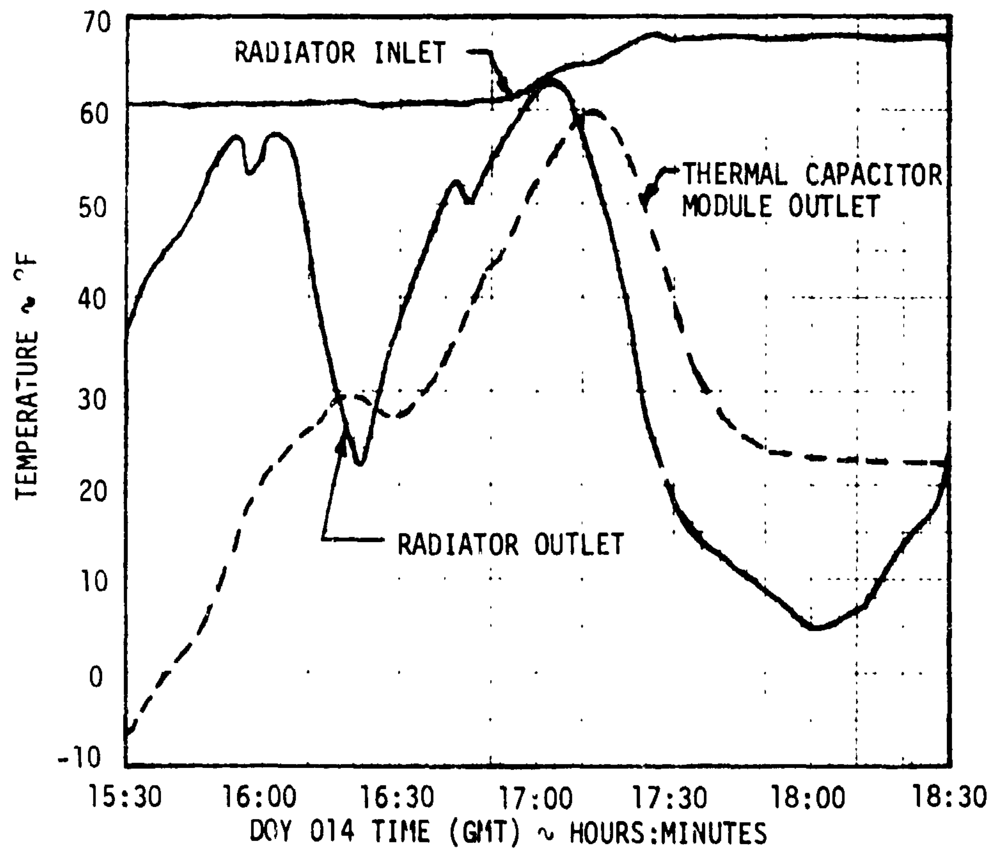


FIGURE 2.4-63 RADIATOR/THERMAL CAPACITOR TEMPERATURES DURING AN EREP Z-LV MANEUVER

loop was free of contamination, action was taken to avoid perturbing the valve until DOY 233 when it appeared to be modulating during a checkout for EVA. However, TCV-B again stuck during EVA on DOY 236 with an outlet temperature of approximately 42°F. Operation with TCV-B in this condition was continued into SL-4 since outlet temperatures were considered acceptable for normal operation. Minimum TCV-B outlet temperature achieved during the colder orbit storage period was 40.1°F.

The TCV-B was observed to modulate properly when subjected to increased temperatures at the cold inlet port during EREP maneuvers at high Beta angles on SL-4. TCV-B operated normally after DOY 019 for the remainder of the mission.

E. In-flight Reservicing - The primary loop was serviced on DOY 323 with approximately 7.7 pounds of Coolanol being added to the reservoirs per the planned procedure in Section 2.4.3.2(E). Although gas leakage was indicated during a leak check of the saddle valve installation, no leakage was observed during the subsequent Coolanol leak check. It was concluded that gas leakage was in the leak test hook-up and servicing proceded normally. Although leakage from the loop continued as shown in Figure 2.4-55, further servicing was not required to complete the SL-4 mission. Operation of the primary coolant loop was completely normal following servicing.

2.4.5.3 <u>ATM C&D Panel/EREP Cooling System</u>

The ATM C&D Panel/EREP cooling system was operated satisfactorily throughout all manned mission phases. Water temperatures of 52°F to 78°F and heat loads between 200 and 1400 Btu/hr were nominal. Water flowrates were normally between 225 and 300 lb/hr; however, periodic, short-duration decreases in flow below this range were observed as described below. The filter in the loop was changed on DOY 149, 165, 266 and 352 and three assemblies returned on SL-2 and SL-3 showed no significant level of contaminants.

Following initial activation of the system on SL-2, water flow was observed to cycle between 240 and 300 lb/hr with a period near one minute for approximately eight minutes. A relatively stable flow of 240 to 245 lb/hr was then achieved. Although this flowrate was well below the 293 lb/hr level obtained during ground testing with pump A operating, no immediate action was initiated since flow exceeded the minimum allowable of 220 lb/hr. Furthermore, the flow remained between 225 and 250 lb/hr with only infrequent variations of short duration below this range. However, near the end of SL-3, the crew reported abnormal noises which were associated with the loop and major flow fluctuations were observed in flight data. Pump A was deactivated on DOY 266 as a result of this condition and pumps B and C were successfully operated as desired until the end of SL-3 on DOY 268.

Following activation of pump B on SL-4, periodic flow decreases from a level of 241 lb/hr were observed. These flow decreases increased in magnitude and frequency as the mission progressed and were also observed during pump C operation. On a number of occasions flow dropped near zero for from one to five minutes. Although ground testing on the ECS System Test Unit (STU) with system flow restrictions or free gas in the loop demonstrated the ability to produce variations in flowrate, a complete duplication of in-flight behavior of the loop was not achieved. Also, since pump differential pressure instrumentation had been disconnected prior to launch, flight data providing an indication of system pressure drop was not available until DOY 347 when the crew observed that the low ΔP light (panel 203) was on at the time of a low flow indication. This observation eliminated further consideration of a system flow restriction as the problem source, and efforts were then limited to definition of a method for removing free gas from the loop. Following ground testing of a liquid gas separator in the STU water loop to ensure compatibility with loop water additives and development of procedures for servicing, installation and operation, a spare liquid gas separator was utilized to remove gas from the flight loop on DOY 352. During this procedure, water flowrates increased to normal preflight levels for operation with either pump B or C. Although flow was stable, the crew noted a definite increase in noise level for this apparently normal condition. The loop was subsequently deactivated per crew request during sleep periods.

Flow remained stable until DOY 360 when flow oscillations similar to those observed at the start of SL-2 were observed. These flow oscillations continued periodically over extended time periods until DOY 001 when the flow dropped to a stable 258 lb/hr following crew placement of the EREP coolant valve to the FLOW position thereby permitting flow through EREP components. Further decreases in flow were observed on the following days and prompted a repeat of the gas removal procedure on DOY 004. Flow rates again increased to normal levels as a result of this operation. However, continued operation of the loop resulted again in a flowrate degradation to 250-260 lb/hr during the last several weeks of the mission, producing the same symptoms caused by gas in the water loop. The flight controllers elected not to attach the liquid/gas separator again for gas removal, since the flowrates were above the minimum allowable of 220 lb/hr except for an occasional downward glitch. Pump A was reactivated on DOY 035 for the first time since shutdown on DOY 266. The pump operated properly, confirming that it had not failed during SL-3; the erratic flowrates instead being due to gas in the loop.

The most probable cause of gas in the loop was generation by electrolysis from stray currents introduced by EREP tape recorders. Gas leakage from the cabin into the water loop was unlikely since the cabin pressure was generally less than the water reservoir pressure except for periods during M509 experiment and cabin oxygen enrichment operations.

2.4.5.4 Battery Cooling

Coolant temperatures and flowrates to battery module coldplates were normal throughout the mission. Temperature control valves upstream of battery modules were in control at all times except during IVA operations with suit cooling system water flow and EREP maneuvers at high Beta angles. During normal operations with the TCV's in control, coolant temperatures ranged between 36°F and 43°F, and battery temperatures ranged from approximately 42°F to 48°F, as predicted. For DOY 014 EREP maneuvers at high Beta angle, coolant inlet temperature reached 68.2°F and battery temperatures approached 60°F. System design changes made to provide supplemental battery cooling permitted batteries to operate at desired temperature levels at all times.

2.4.5.5 Integrated Temperature Control

Integrated temperature control of Airlock Module structure and noncoldplated equipment was provided by normal operation of the active cooling system, atmospheric control system and wall heaters in conjunction with thermal coatings, curtains and insulation. Temperatures of all equipment and structure remained within acceptable limits throughout all phases of the mission.

Since all Airlock Module surfaces with the exception of the FAS were protected by the Payload Shroud during ascent, only minor temperature changes were experienced. Average temperatures on the exposed FAS increased approximately 6°F during this period to a level of 99°F.

With the unplanned vehicle attitude during the initial SL-1 unmanned phase when pitch angle was maintained at 45 to 50 degrees (or higher) to minimize OWS temperatures after loss of the meteoroid shield, Airlock Module temperatures were abnormally cold as seen in Figure 2.4-64. Although indicated temperatures were still acceptable, instrumentation was rather limited and colder areas may have existed. Following deployment of the parasol on DOY 147 and return to the planned solar inertial attitude, temperatures increased to normal levels.

Airlock Module wall heater 42°F and 62°F thermostats were enabled throughout the mission except for short periods when electrical load reductions were required for purposes of power management. Wall temperatures between 53°F and 60°F during storage periods with the vehicle in the normal solar inertial attitude indicate that continuous heater operation was required during unmanned flight phases. Wall and gas temperatures during manned phases of the mission are shown in Figures 2.4-65 and 2.4-66, respectively. Wall temperature levels were such that only infrequent heater operation may have been required during these periods. Indicated temperature levels were reported by crewmen to be comfortable. Sun side STS windows remained free of condensation, while dark side windows fogged occasionally when the covers were open.

Temperatures of the FAS are shown as a function of Beta angle in Figure 2.4-67. The temperature range for O_2 and N_2 tanks is shown in Figures 2.4-68 and 2.4-69, respectively where tank temperatures for hot and cold locations are plotted. It can be noted that the temperature of O_2 tank 6 was off-scale high (above 160°F) at high Beta angles. A maximum temperature of 210°F was calculated based upon increase in tank pressure at constant mass. Since there was no O_2 usage during the period following SL-2, this method should provide adequate results. Although N_2 tanks 1 and 2 reached temperatures somewhat higher than predicted, the maximum level achieved (130-135°F) was well below the 160°F design value.

LOCATION	MINIMUM TEMPERATURE (°F) [1]	
	SL-1 [2]	NORMAL [3]
FAS (-Z AXIS)	-16 to -9	120 to 200
FAS (-Y AXIS)	-48 to -34	15 to 40
FAS (+Z AXIS)	-70 to -56	-50 to -19
FAS (+Y AXIS)	-42 to -21	-21 to 0
O_2 TANK 1	-30.7	-14.4
O_2 TANK 2	-31.6	-12.6
O_2 TANK 3	-51.2	-30.8
O_2 TANK 4	-43.5	-15.2
O_2 TANK 5	5.0	82.8
O_2 TANK 6	3.4	134.6
N_2 TANK 1	32.5	88.9
N_2 TANK 2	24.0	89.2
N_2 TANK 3	17.9	47.9
N_2 TANK 4	10.0	50.7
N_2 TANK 5	1.6	25.0
N_2 TANK 6	4.0	27.4
STS INNER SKIN	38.8	57.0
LOCK COMPT INNER SKIN	41.9	57.1
AFT COMP INNER SKIN	43.1	57.2
SUS 1 WATER LINE	33.7	43.0
SUS 2 WATER LINE	38.6	47.5

NOTES:
[1] ORBITAL RANGE
[2] DOY 140-146, BETA = 22° to 28°, PITCH ANGLE = 45° to 50°
[3] DOY 201, BETA = 25°

FIGURE 2.4-64 EFFECT OF SL-1 ATTITUDE ON AIRLOCK MODULE TEMPERATURE

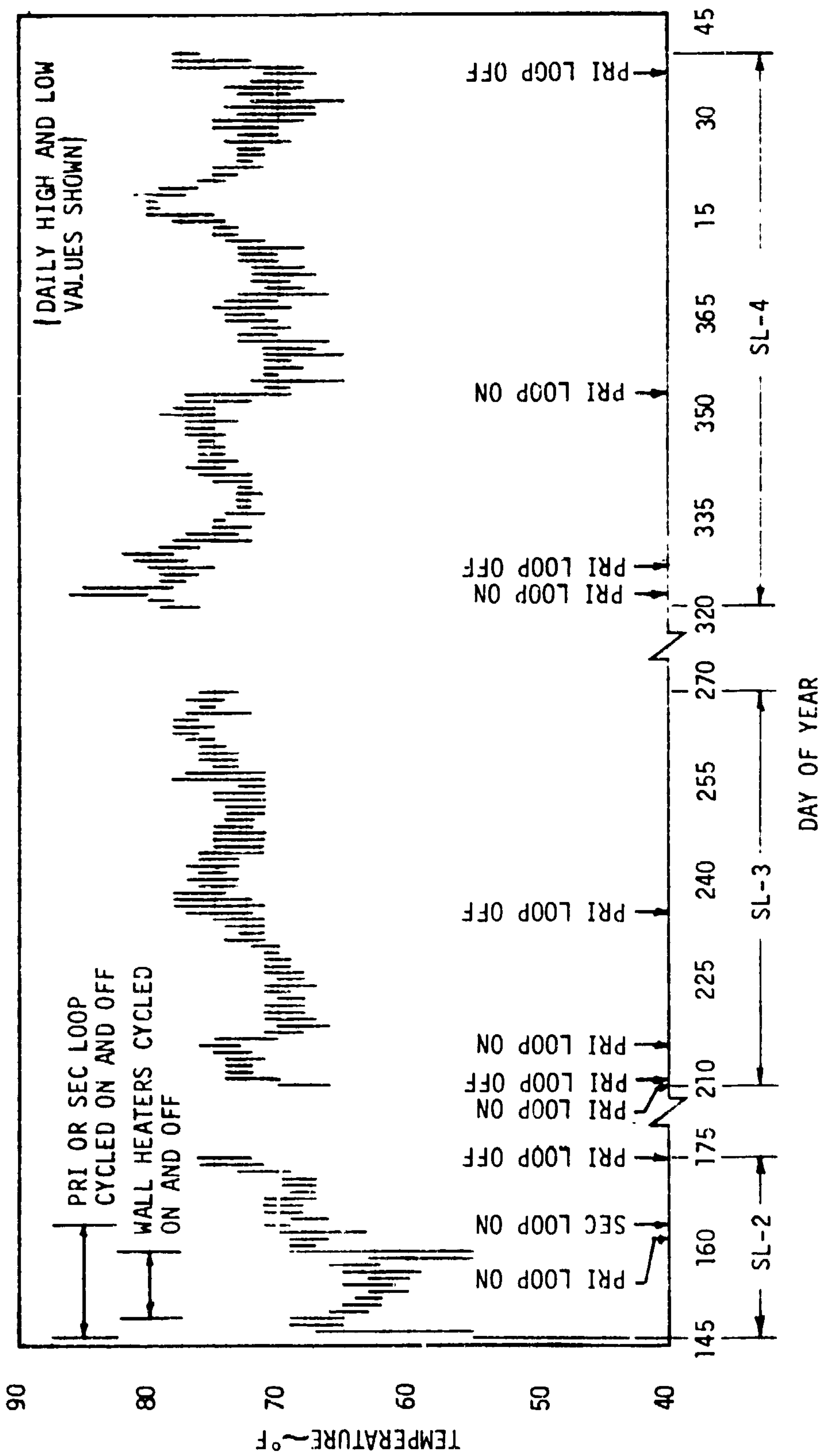


FIGURE 2.4-65 STS WALL TEMPERATURE

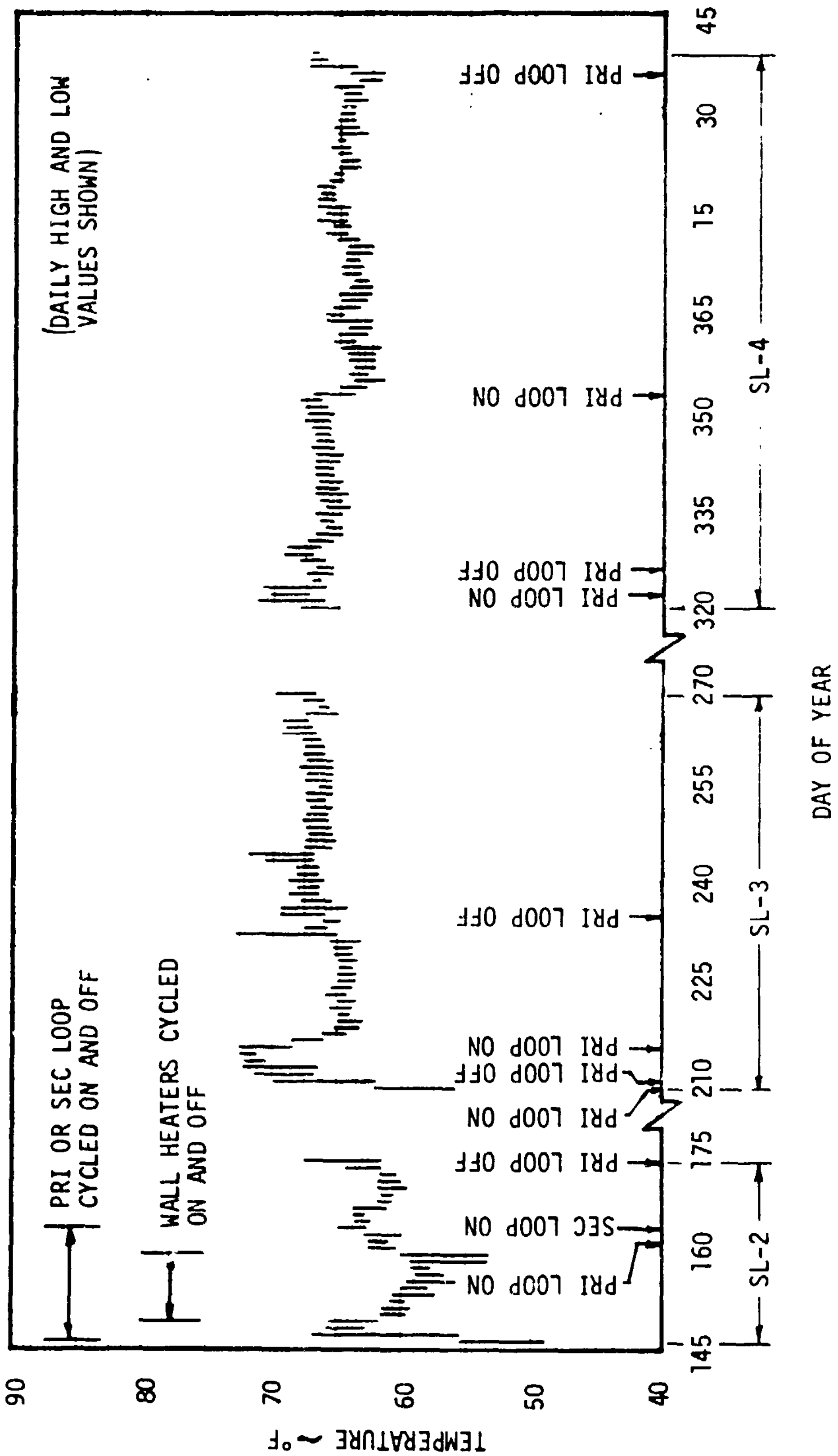


FIGURE 2.4-66 STS GAS TEMPERATURE AT MOLE SIEVE A COMPRESSOR INLET

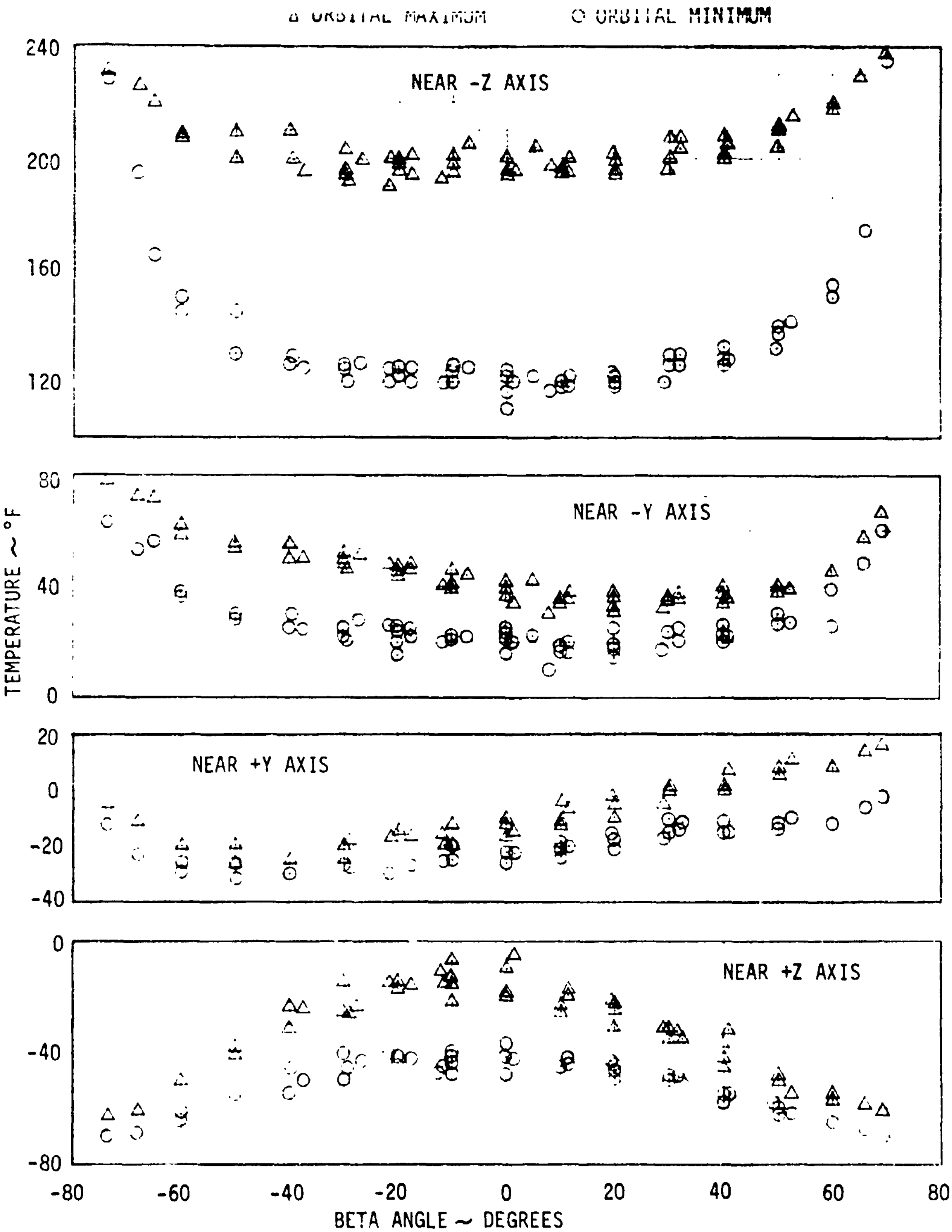


FIGURE 2.4-67 FAS SKIN TEMPERATURE – SOLAR INERTIAL ATTITUDE

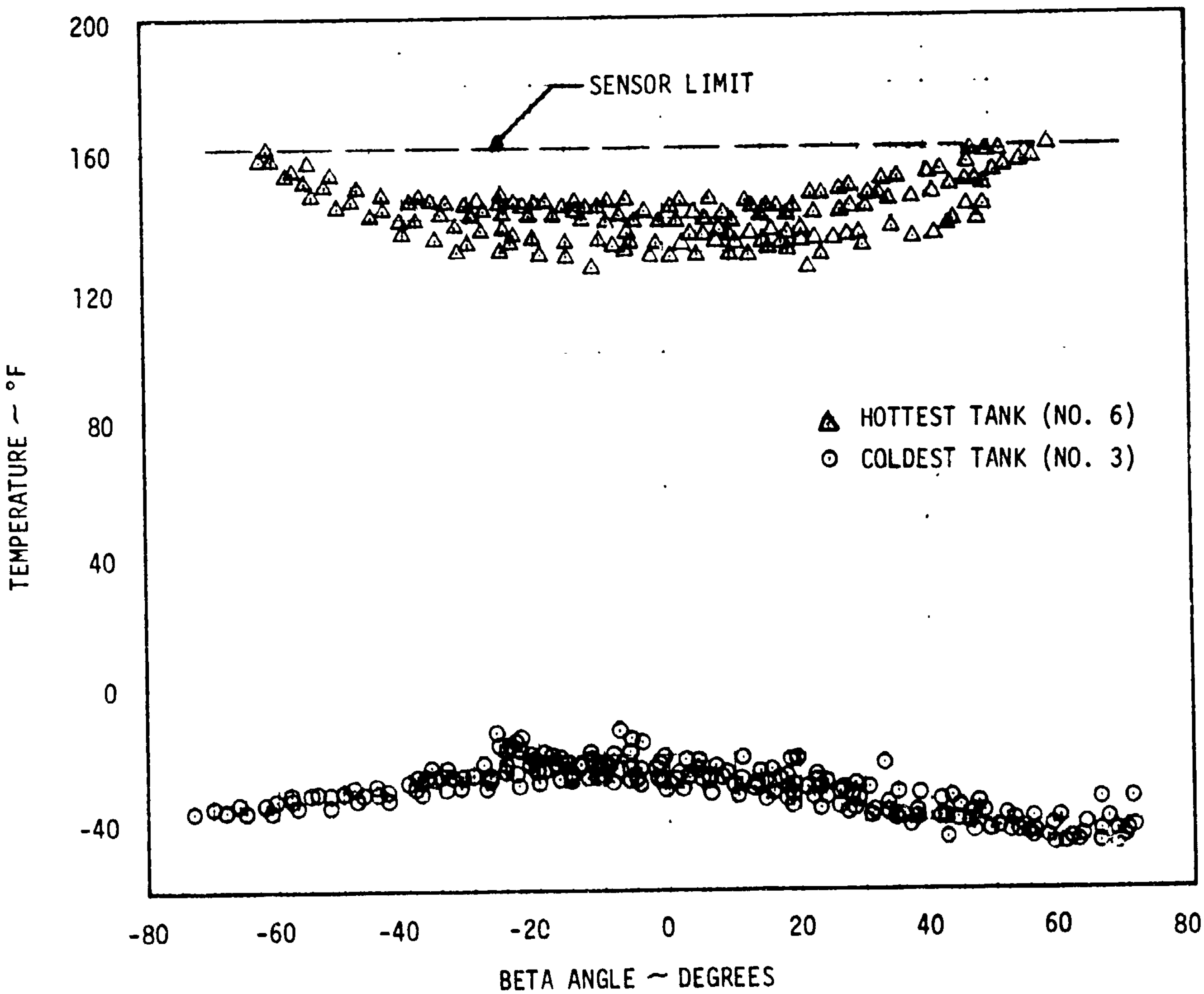


FIGURE 2.4-68 O_2 TANK TEMPERATURE – SOLAR INERTIAL ATTITUDE

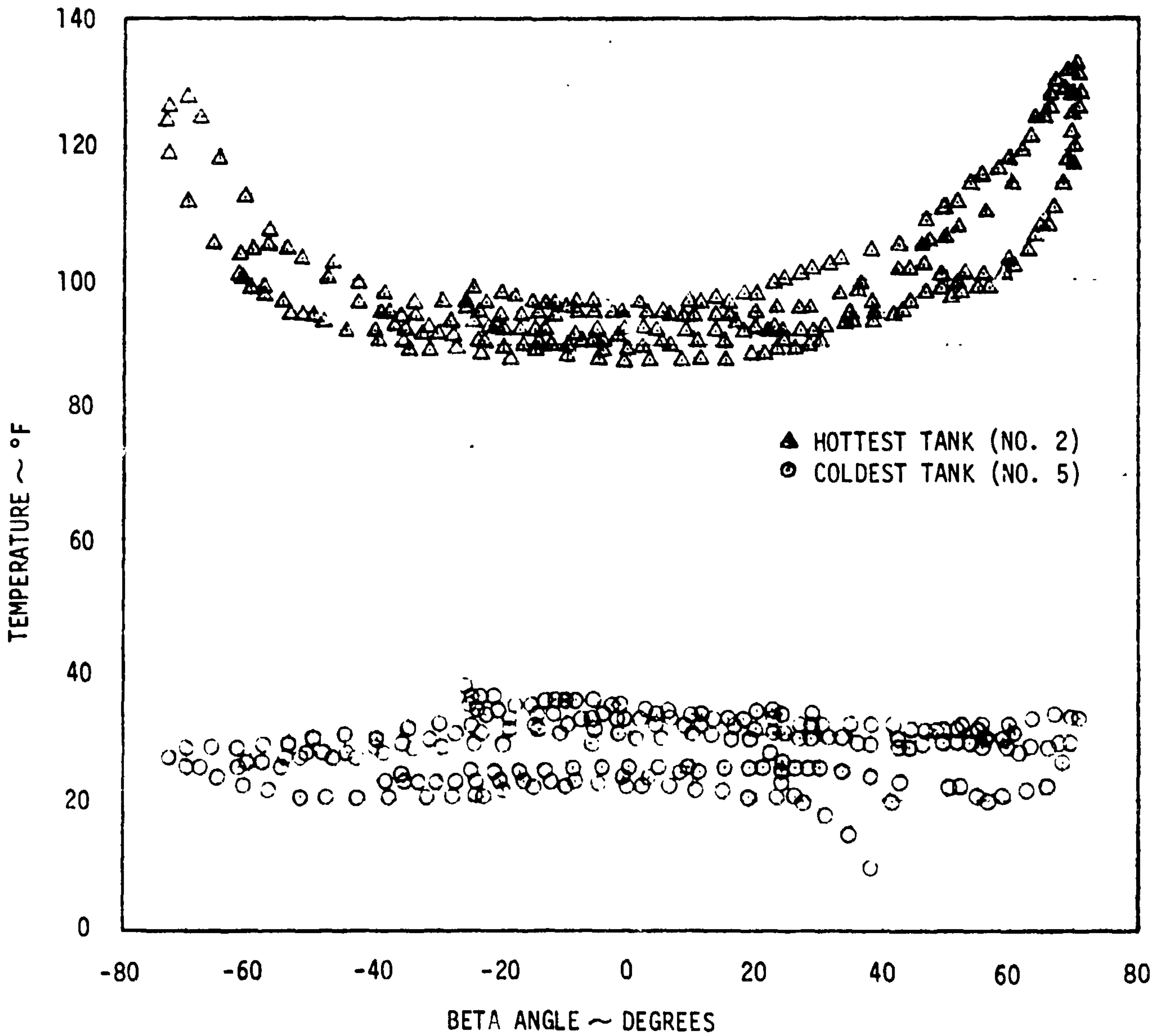


FIGURE 2.4-69 N_2 TANK TEMPERATURE – SOLAR INERTIAL ATTITUDE

2.4.5.6 Molecular Sieve Exhaust Duct and Condensate System Vent Heaters

Based on data available from limited instrumentation, molecular sieve duct heaters maintained temperatures well within desired levels. The only discrepancy associated with Airlock Module heaters was reported on DOY 147 when the crew failed to obtain an indication of temperature rise after condensate system heaters on the secondary vent were activated. Although normal operation was indicated with primary vent heaters, attempts to operate with the secondary vent were later repeated and normal indications of heater operation were obtained. The earlier difficulty was attributed to a display problem.

2.4.6 Development Problems

Three development problems were experienced in the thermal control system. An awarness of these may be of benefit to future programs.

2.4.6.1 Coolant System Temperature Control Stability

When the coolant system configuration was changed to meet the 46°F minimum dew point and supplemental battery cooling requirements, development tests were conducted to determine system stability. These tests showed a stability problem. Because of the short time available to develop a design which would provide control stability, a test approach was taken. The tests led to rearrangement of the tubing interconnecting the suit cooling heat exchangers, the addition of a heat exchanger bypass line with bleed orifice, and the addition of the EVA flow selector valve.

The purpose of the rearrangement of interconnecting tubing was to isolate thermally the hot and cold inlets to the downstream temperature control valve (TCV-B) as much as possible without freezing the water. With the hot and cold inlets coupled through additional heat exchangers, the time lag between TCV-B movement and TCV-B outlet temperature change was larger. In addition, flowrate cycles caused reversal in temperature of streams entering the hot and cold ports of TCV-B.

The purpose of adding the cold bypass around the suit cooling heat exchangers was to reduce the time lag between TCV-B movement and temperature control valve outlet temperature change during suit cooling operation. The small amount of flow bypassed had a negligible effect on suit cooling capacity.

The purpose of adding the flow selector valve was to reduce the time lag between TCV-B movement and TCV-B outlet temperature change when the cold port of TCV-B opened during normal system operation. This was achieved by routing cold fluid to the cold port of TCV-B and preventing warm fluid in the suit cooling heat exchangers from entering the cold port. Before the selector valve was added, most of the time when suit cooling was not utilized there was negligible coolant flowing through the cold port of TCV-B. Consequently there was no coolant or water flow through the main cooling heat exchanger and it approached a temperature equal to the surroundings. Even a small flow of warm fluid into the cold port of TCV-B created some instability. Use of the selector valve in accordance with procedures eliminated this instability.

2.4.6.2 <u>Coolant System Thermal Capacitor Integrity</u>

Because of uncertainties in both the method of fabricating a capacitor housing containing isolated cells and the need for isolated cells, the cells in the original design were interconnected and development tests were run to establish design adequacy. The development and qualification performance simulation tests on the AM capacitor were run for the expected range of temperature profiles with coolant flow through both coldplate passages and no problems were encountered. However, tests conducted on a similar container containing Undecane wax for use in the OWS refrigeration system resulted in bulges of the wax chamber side walls and cracking of the wall. After a thorough investigation it was determined that one-passage coolant flow together with large heat up rates could cause bulging and ultimately cracking of the wax chamber. The capacitor housing was redesigned to provide isolated wax cells.

2.4.6.3 <u>Equipment Insulation Vapor Barrier</u>

During SST prelaunch U-1 tests the Microfoil insulated lines exposed to the cold ground coolant (Supply and Return, FAS to GCHX) and the interfacing GCHX spacecraft lines developed considerable water condensation which apparently penetrated the Microfoil vapor seal and caused a degradation in the insulation effectiveness. The insulation on these lines was then redesigned to include a positive vapor seal.

2.4.7 Conclusions and Recommendations

The TCS performed so effectively that original mission objectives were expanded and all mission objectives were accomplished. Designed-in redundancies and real-time work-around procedures were used to alleviate the effect of system discrepancies that did occur. Consequently, it is recommended that, in general, the Airlock Thermal Control System design and test concept, as well as system hardware, be considered as the starting point for development of a future mission system.

A more detailed discussion of conclusions and recommendations is given below:

- System design requirements were realistic and should be used on other programs.
- The integrated thermal analysis was effective in establishing realistic interface requirements and simplifying the vehicle design. It is recommended that integrated thermal analysis be performed by the designated lead group which will provide detailed requirements to other groups for use as basis for design of their systems.
- The AM temperature control concept which allocated cooling to the various loads on a priority basis was proven and should be considered for future programs.
- Radiator/thermal capacitor performance was outstanding - even during ZLV maneuvers at much higher beta angles than originally planned. No significant degradation was noted. These components are prime candidates for future mission use.
- Thermal Control Valves (TCV) did jam due to loop contamination - several changes are available to minimize this condition:
  - (1) Utilize inline filters and bypass relief valves more effectively.
  - (2) Redesign valves to reduce susceptibility to contamination.
  - (3) Improve loop cleanliness controls.
- Although both coolant systems exhibited leakage, no degradation of mission occurred. The primary system was successfully reserviced by SL-4 crew. Although type and location of leak could not be determined, minimizing use of mechanical fasteners would reduce a leak potential. Capability for reservicing coolant systems in orbit should be incorporated in systems for future programs.

- ATM C&D/EREP cooling system flow became erratic late in SL-3 Mission. Successful deaeration of loop, using liquid gas separator, temporarily corrected flow oscillations. Fluid hydrolysis was the most probable cause of gas in the loop. Better control of stray electrical currents is needed on future programs.
- Battery temperatures were maintained within the normal range of 45° to 50°F. This transient cooling concept should be considered for the next program.
- Thermal system verification concept of detailed thermal analysis plus limited tests was proven. Components were qual tested but no vehicle thermal qualification was performed.
- Acceptable temperatures were maintained on all passively cooled equipment and structure. The use of coating, insulation and single layer thermal curtains should be considered for future space vehicles.

2.5 ENVIRONMENTAL CONTROL SYSTEM

The environmental control system provided a habitable environment for the Skylab crew. It consisted of an integrated array of systems and subsystems. Included were subsystems for O_2 and N_2 gas storage, distribution and pressure control, atmosphere cooling and circulation, CO_2 and odor removal, atmospheric condensate removal and disposal, and in-flight water systems servicing.

2.5.1 Design Requirements

The basic requirements were to provide atmospheric composition, pressure, and temperature control. In addition, interface requirements were provided for the Multiple Docking Adapter, Orbital Workshop, Apollo Telescope Mount, Payload Shroud, IU, Experiments, Thermal Control System, EVA/IVA, and GSE. The interface functions are presented in Figure 2.5-1. Additional requirements associated with the ECS design were to provide instrumentation intelligence and procedures as a basis for system operation.

2.5.1.1 Evolution

The AM ECS evolved with a minimum hardware development concept, utilizing subsystem equipment previously developed, tested, and flown on the Gemini spacecraft. This minimum development approach was maintained where possible throughout the program in the interest of austerity, but subsequent mission changes and new program objectives dictated many design requirement changes. Evolution of the overall Skylab Program is summarized in Section 2.1. The highlights of the resultant effect of these changes on the environmental control system requirements and design are summarized as follows:

A. Gas System - The initial requirements were to store and supply O_2 at sufficient quantities and flowrates for replenishment of atmospheric leakage and metabolic consumption for three crewmen for a 30 day mission and to provide O_2 and H_2 for the CSM fuel cell. Initially the cluster atmosphere was to be 5 psia O_2. To store the required O_2 and H_2, modified Gemini O_2 and H_2 cryogenic tanks were mounted on AM trusses. Thermostatically controlled calrod heaters, installed on the lines downstream of the cryo tanks, warmed the gases supplied to the distribution system. Gemini pressure regulators provided O_2 supply and pressure control. Later, the atmosphere was changed to an O_2/N_2 mixture at 5.0 psia total pressure. N_2 gas tanks, regulators, and manual controls were added to the AM.

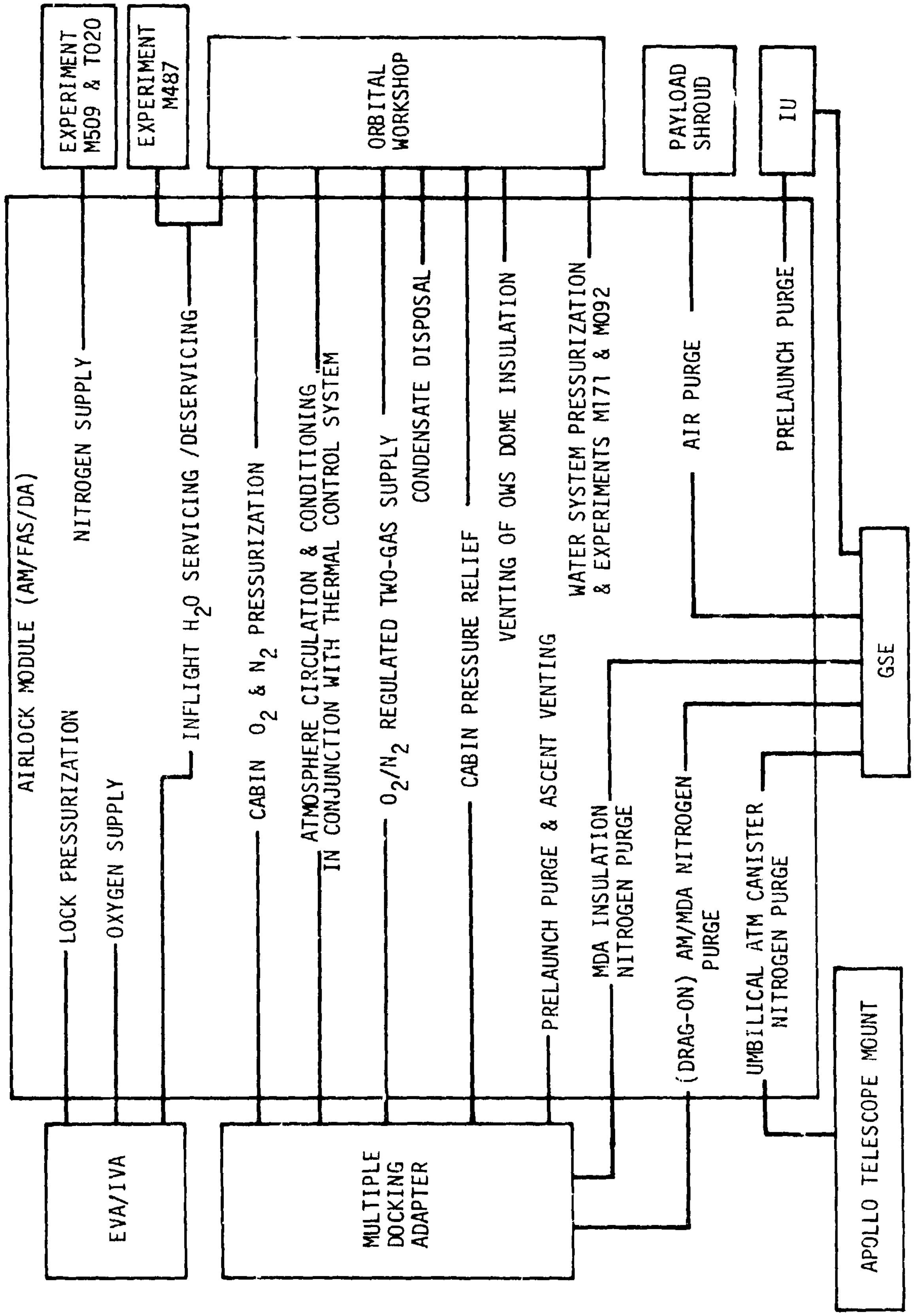


FIGURE 2.5-1 AIRLOCK ENVIRONMENTAL CONTROL INTERFACE

As the Wet Workshop design progressed, both the cryo tanks and gas tanks were removed from AM. O_2 and N_2 was then supplied from the CSM for a two gas atmosphere. O_2 flowrates available from the CSM, however, were insufficient for meeting EVA/IVA and M509/T020 experiments support requirements. To meet the higher flowrate requirements, two high pressure gaseous O_2 tanks (LM descent tanks) as well as 120 psig and 240 psig O_2 pressure regulators were added to the AM. The gaseous O_2 tanks were to be launched pressurized to 2250 psia, and then, after depressurization to below 1000 psi by usage, were to serve as accumulators.

The accumulators were to be charged with O_2 delivered from the CSM to the AM through an umbilical. Gas in the accumulators was used to supplement O_2 flowrates from the CSM. The N_2 and O_2 required for S-IVB purging and initial pressurization was supplied from the CSM through the same umbilical. The N_2 required for maintenance of atmospheric pressure and O_2/N_2 composition control was introduced in the CSM. Both initial pressurization and O_2/N_2 composition were accomplished by inflight manual controls.

Changeover to the Dry Workshop with Saturn V booster permitted a large allowable launch weight. Consequently direction was given to store all O_2 and N_2 supplies required for the Skylab mission onboard the AM. Storage of the O_2 and N_2 as high pressure gases was selected over cryogenic storage because of lower cost, lower development risks, ease of servicing, and more operational flexibility for the multi-mission Skylab program. Additional changes in design requirements which reflected on the system design during this time are listed below:

(1) Requirement for DCS command and onboard control of O_2 and N_2 flowrates for Skylab initial pressurization.
(2) Addition of an automatic two-gas (O_2/N_2) atmosphere control system.
(3) Requirement for supplying N_2 to the OWS for drinking water tank pressurization and M171 experiment.
(4) Requirement to supply N_2 instead of O_2 for mole sieve valve actuation and water system reservoir pressurization.
(5) Addition of an N_2 recharge station for in-flight servicing M509 N_2 supply tanks.

B. Atmospheric Control System

- Humidity Control - The requirement for a minimum atmospheric dew point temperature of 46°F following the flowrate increase through the molecular sieve required to meet the lower CO_2 partial pressure requirement during the Dry Workshop evolution, further complicated overall system design because more atmospheric moisture was absorbed and dumped overboard by the molecular sieve. However, this problem was ultimately solved by increasing the coolant temperature entering the condensing heat exchangers from 40°F to 47°F, thus raising the atmosphere dew point by reducing the amount of moisture condensed in the heat exchangers. The original Gemini 40°F temperature control valves were replaced by off-the-shelf valves of a different design, but modified to control coolant temperature to 47°F. This change was accomplished simultaneously with that required to reduce coolant temperatures delivered to the battery modules (Ref. Section 2.4) and necessitated additional system changes in order to maintain required performance for the EVA/IVA suit cooling system water delivery temperatures (Ref. Section 2.6).
- CO_2 and Odor Control - Cluster O_2 and odor removal was originally supplied by Gemini LiOH canisters, having a 14-day capacity for two men, which were to be replaced in flight. The molecular sieve designed for the Apollo Applications Program was to be carried in the cluster as an experiment. Studies were performed on providing the hot and cold fluids for its operation on Airlock which resulted in the decision to build an adiabatic desorb molecular sieve. This sieve system went from an experiment, to a backup for LiOH, to prime with LiOH backup in the transition to the Wet Workshop configuration. Initially the Gemini LiOH canisters were retained for CO_2 and odor removal during the first 28 day mission, but the mole sieve was to provide this capability on subsequent missions. Atmospheric CO_2 partial pressure was to be maintained below 7.6 mm Hg during normal system operation and below 15 mm Hg during system failure conditions. The configuration was changed later by direction for removal of the LiOH and addition of a second molecular sieve system; with both molecular sieves installed in the AM STS.

The Dry Workshop configuration reduced the allowable atmospheric CO_2 partial pressure requirement from 7.6 mm Hg to lower values, finally settling down to a design limit of 5.5 mm Hg. The molecular sieve was thus modified to provide this new requirement by increasing the gas flowrate through the sorbent canisters from 10 lb/hr to 15.5 lb/hr.

A concern over possible contamination of external optical surfaces by exhaust gases from the molecular sieves during bed desorption resulted in the directive to relocate the molecular sieve overboard exhaust duct. As a result, both molecular sieve overboard ducts were combined and relocated to exhaust from a single outlet from the side opposite the optics.

- Ventilation - The ventilation system originally utilized Gemini cabin fans which were later replaced by GFE Apollo Post Landing Ventilation (PLV) fans. Advantages of the PLV fans were 1) needed no separate AC/DC power inverter, 2) required less power, and 3) standardized fans throughout the cluster since PLV fans were also used in the MDA and OWS. However, the PLV fans had undesirable flow/ΔP characteristics for use in conjunction with the AM cabin heat exchangers. The lack of pressure head from the PLV fan necessitated the use of low pressure drop filters and ducting. In addition, it resulted in considerable systems analyses and tests to define and overcome flow degradation problems due to condensation in the heat exchangers. The inclusion of sound suppression equipment in the fan module designs to satisfy cluster noise level specifications resulted in additional system resistances, which also interacted unfavorably with the marginal fan characteristic.
- Temperature Control - The most significant change made to the atmospheric cooling system occurred with the Wet Workshop phase. It was installation of the aft compartment cabin heat exchanger module to provide more sensible cooling to the OWS. Space previously occupied by the LiOH system was used for this purpose.

C. Condensate System - Concern that dumping condensate overboard may interfere with experiments which involved external sightings, caused condensate system design requirement changes. Some of these changes are listed below:

(1) Relocate the AM condensate overboard dump ports to the opposite side of the spacecraft from the affected optical surfaces.

(2) Provide the capability to dump condensate from the AM storage tank to the OWS waste tank, which precluded release of water or ice particles of sufficient size to contaminate the optics.

(3) Modify the AM dump ports to include restricted outlets designed by Martin Co., which would presumably cause a more predictable exhaust plume profile.

(4) Provide capability to transfer condensate directly from the AM condensing heat exchangers to an evacuated holding tank located in the OWS. The condensate was to be stored in the holding tank and subsequently dumped to the OWS waste tank. The latter change resulted from water freezing at the OWS waste tank dump probe. Freezing was encountered during tests simulating condensate transfer from the AM storage tank to the OWS waste tank. The OWS dump probe was modified to permit dumping from the AM condensate tank to the OWS waste tank. However, transfer to the OWS holding tank was retained as the primary method because the larger volume of the holding tank allowed a longer period of time between dump operations.

D. Inflight Water Servicing - A design requirement change was made relatively late in the program to provide a positive means for in-flight servicing of the condensing heat exchanger water separator plates. This change was prompted by the concern that the plates may dry out during low water generation rate periods of the mission and by uncertainties associated with the previously baselined self wetting method. The new method had the advantage of being a straight forward step-by-step process which assured positive plate wetting. Although, the self-wetting approach had been proven satisfactory during development testing, and required fewer operational steps, its success in-flight would have been strongly dependent on cluster dew point and proper crew attention.

The self-wetting technique was sensitive to both free water carryover to the molecular sieves and gas carryover to the condensate collection system.

Design changes were made to include the capability of servicing the ATM C&D/EREP system and the EVA/IVA Suit Cooling System inflight with water stored in the OWS drinking water tanks. In addition, provisions were made for inflight servicing and deservicing the GFE Life Support Umbilicals (LSU's) and Pressure Control Units (PCU's).

2.5.1.2 <u>Flight Configuration</u>

Design requirements for the flight article are presented in this section.

A. Atmospheric Control - Atmospheric parameters were pressure, humidity, purity, ventilation, and temperature.

- <u>Pressure</u>
  - Allow launch ascent venting and orbital pressurization/ depressurization.
  - Provide remote pressurization of AM/MDA/OWS with oxygen, nitrogen or mixed gas for initial fill.
  - Automatically maintain the minimum O_2/N_2 atmospheric pressure in the AM/MDA/OWS to 5 $\pm$ 0.2 psia with PPO_2 of 3.6 $\pm$ 0.3 psia after activation.
  - Provide on-board pressurization capability for redundancy.
  - Limit the maximum atmospheric pressure to 6.0 psig after activation.
  - Provide the capability to maintain atmospheric pressure above 0.5 psia during orbit storage.
- <u>Humidity</u>
  - Permit prelaunch purge for humidity control.
  - Provide proper humidity levels by removing metabolic moisture for three men from the AM/MDA/OWS to bulk average dew point temperature between 46°F and 60°F after activation. Dispose of condensate by transferring to OWS holding tank or venting to space.
- <u>Purity</u>
  - Permit prelaunch purge of AM/MDA for purity control.
  - Remove carbon dioxide and odors from the AM/MDA/OWS after activation.
- <u>Ventilation</u>
  - Permit prelaunch purge of AM/MDA for ventilation
  - Provide circulation and atmospheric movement after activation.
- <u>Temperature</u>
  - Permit prelaunch purge of AM/MDA for temperature control.
  - Cool atmospheric temperature in AM between 60°F and 90°F after activation.
  - Provide atmospheric cooling for the MDA and the OWS.

B. Store and Supply Gases - Oxygen and nitrogen were stored and distributed for various usages.
   - Store oxygen and nitrogen in gaseous state.
   - Provide oxygen and nitrogen for metabolic consumption and cabin leakage.
   - Provide oxygen for EVA/IVA support.
   - Provide pressurized nitrogen supply for mole sieve operation.
   - Provide nitrogen to the recharge station for Experiments M509 or T020, and to the OWS for water tank pressurization and Experiments M171 and M092.

C. Provide In-flight Servicing/Deservicing - Provide in-flight servicing of the water separator plate assemblies, and water systems used in the ATM C&D/EREP and EVA/IVA systems.

2.5.2 System Description

The environmental control system included several systems such as the gas system, atmospheric control system, condensate system, and the in-flight water servicing/deservicing equipment.

2.5.2.1 Gas System

The gas system permitted prelaunch purge and ascent venting, provided O_2 and N_2 storage, pressure regulation, and gas distribution for in-orbit flight operations. The gas system is shown schematically in Figure 2.5-2.

A. Prelaunch Purge - Prelaunch purging of the AM/MDA internal atmosphere, the Payload Shroud, and the MDA and OWS High Performance Insulation (HPI) was provided to control gas composition, temperature and pressure through launch preparation. Prelaunch purge interfaces and flowrate requirements are shown in Figure 2.5-3. For launch, the AM/MDA internal atmosphere was purged with dry nitrogen gas until the oxygen content within the compartments was less than 4%. The purge gas was introduced through the aft compartment purge fitting, GSE 413 (Figure 2.5-2) at a flowrate of 10 lb/min maximum and exhausted through Government Furnished Equipment (GFE) vent valves located in the MDA. The OWS was pressurized with GN_2 and maintained at a pressure level equal to, or greater than, the AM/MDA pressure. During purging, AM/MDA pressure was sensed from the aft

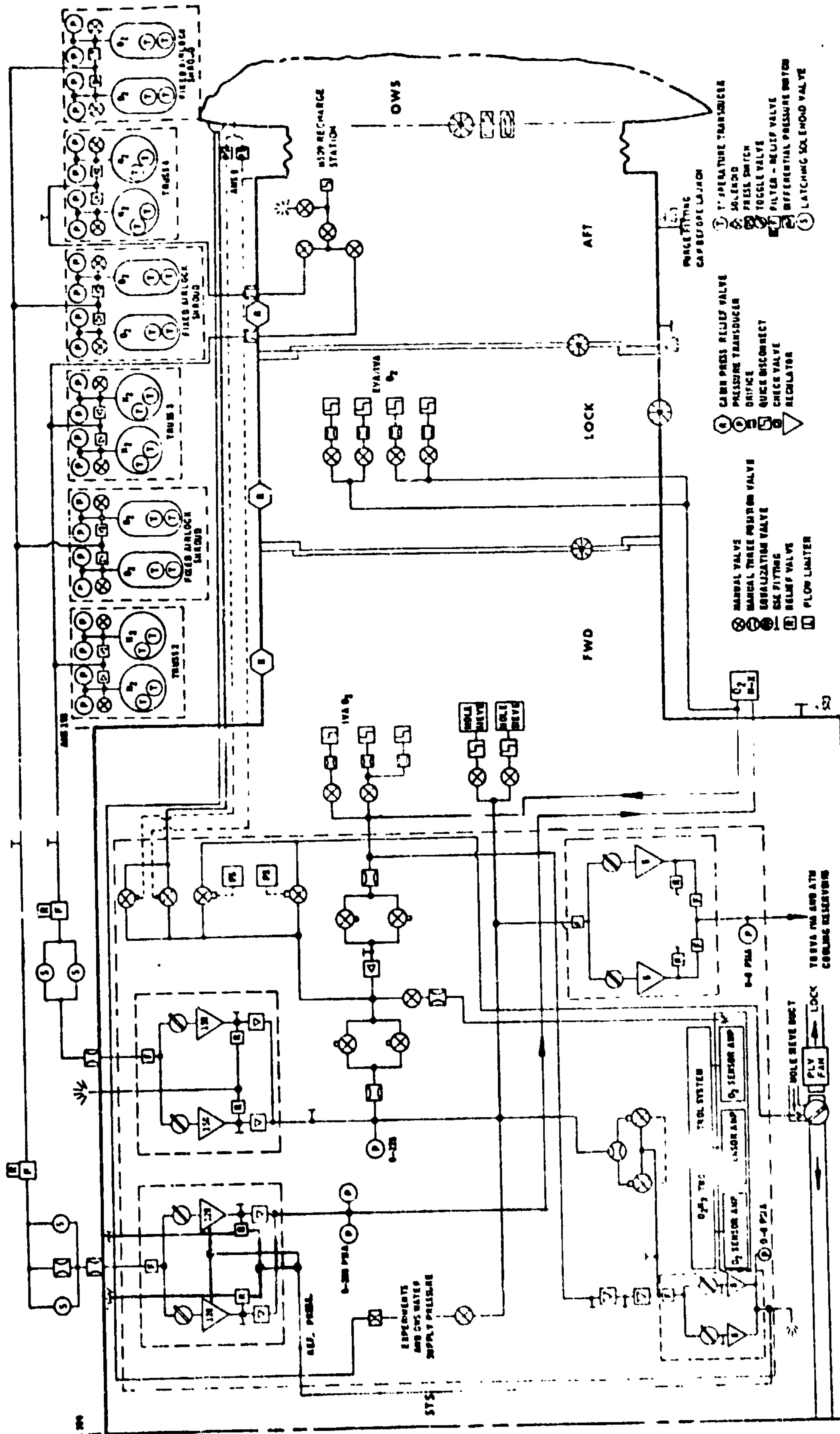


FIGURE 2.5-2 GAS SYSTEM

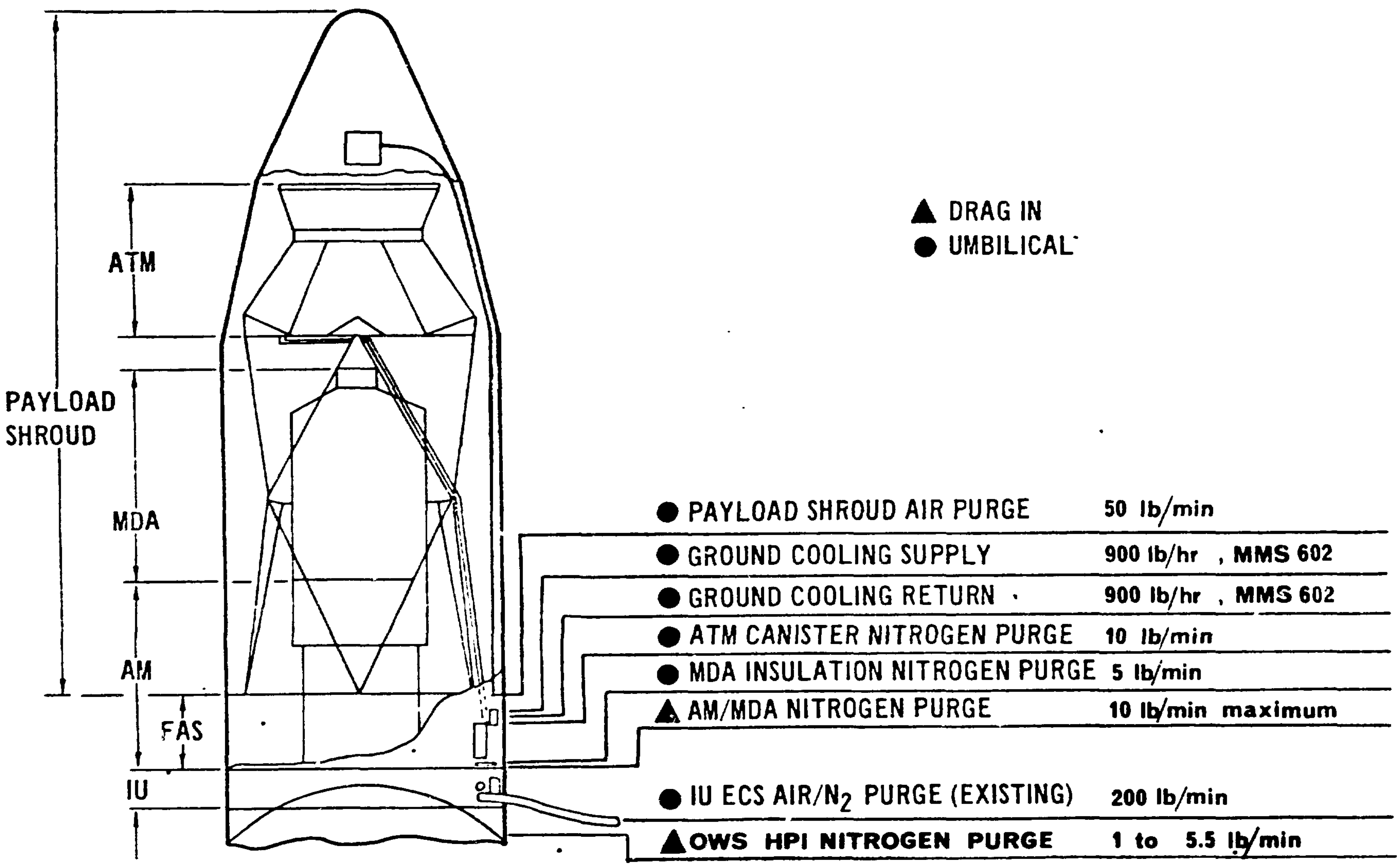


FIGURE 2.5-3 AIRLOCK CLUSTER PURGE AND COOLING REQUIREMENTS

compartment monitor fitting, GSE 412, and was limited to 6.2 psig by a GSE pressure sensing switch which, if actuated, would automatically stop purge flow. Prelaunch purge time to reduce oxygen content to less than 4% was determined during SST by gas composition sampling at MDA vent valves and at the STS monitor fitting, GSE 523. Results of this test showed that 40 minutes was adequate with purge gas flow of 5.0 $\pm$ 0.5 lb/min and was the basis for prelaunch purging criteria at KSC. The maximum allowable depressurization rate was 0.30 psi/sec. Gas from the OWS HPI purge was vented through AM thermal curtain vents.

B. Launch Ascent Venting - Launch ascent venting of the AM/MDA was performed through the MDA vent valves to prevent the cabin pressure differential from exceeding 5.5 psid. The maximum allowable depressurization rate was 0.17 psi/sec. The MDA vent valves were closed automatically at a preselected time during ascent by commands from the IU to maintain the absolute pressure at or above 0.5 psia. After orbit insertion the OWS was vented to a pressure of 0.5 psia minimum through vent valves by IU command. The maximum allowable AM depressurization rate from this venting was 0.10 psi/sec. Prior to launch, the volume between the STS window panes was vented to the cabin; after crew arrival the window pane vents were closed. The STS window controls are shown in Figure 2.5-4.

C. Gas Storage - Oxygen and nitrogen for all missions was stored in the gaseous state and carried during initial launch. Design usable gas quantities were based upon a pressure range of 3000 psig to 300 psig.

(1) Oxygen - Oxygen was stored in six tanks contained in three modules mounted on the Fixed Airlock Shroud. Each tank had a fill valve, a check valve, two temperature transducers, and two pressure transducers. The oxygen tanks shown in Figure 2.5-5 were constructed with a thick fiberglass wrap on a thin welded metallic liner. Each was cylindrical shaped with elliptical ends and was 45 inches in diameter and 90 inches long. A minimum design total of 5,620 lbs of oxygen was required, of which 4,939 lbs was to be usable at normal design flowrates. As described in Section 2.5.4, a total of 6,085 lbs was the calculated quantity actually loaded. The margin above the design value was included to account for instrumentation inaccuracies at the time of servicing.

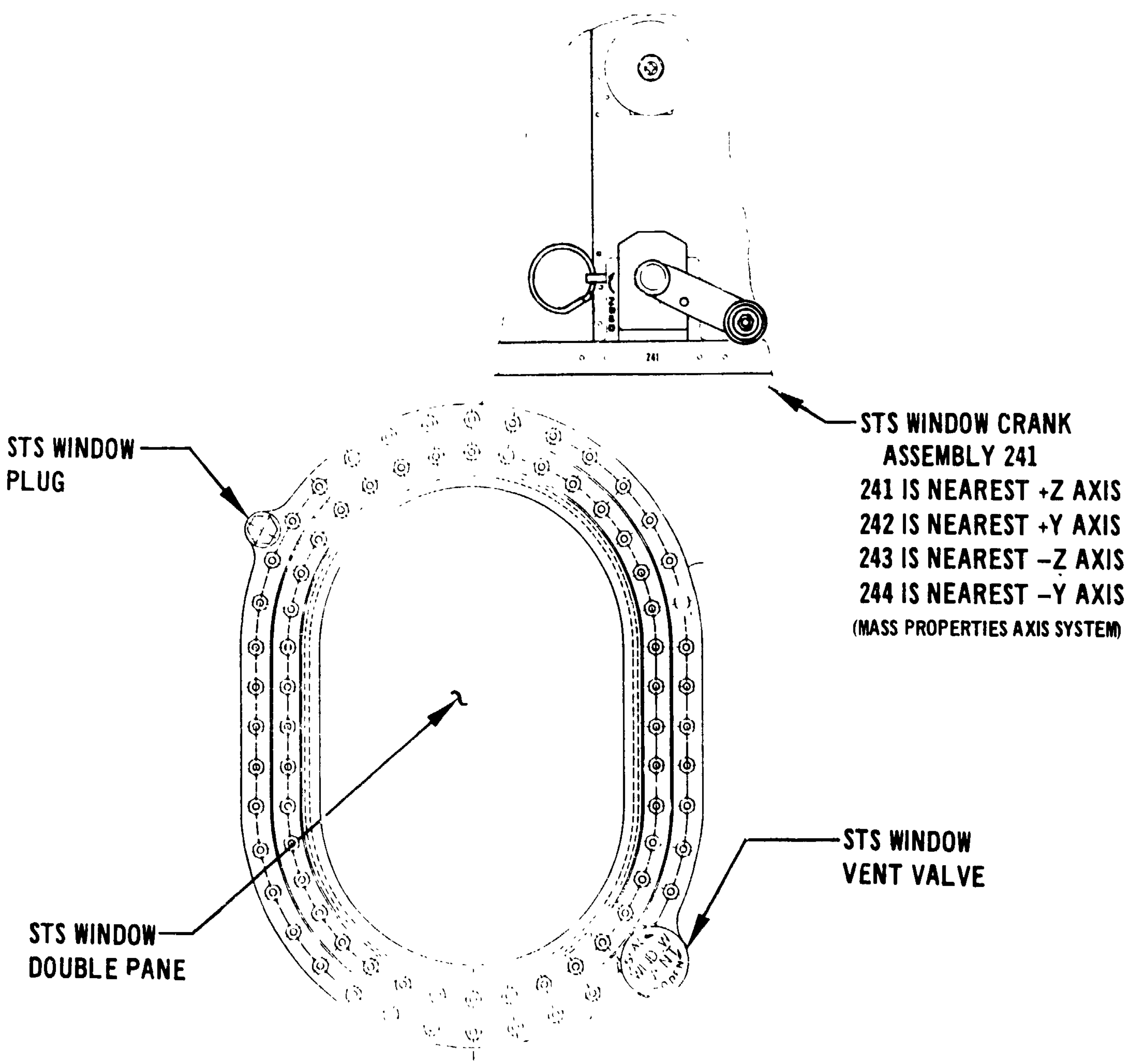
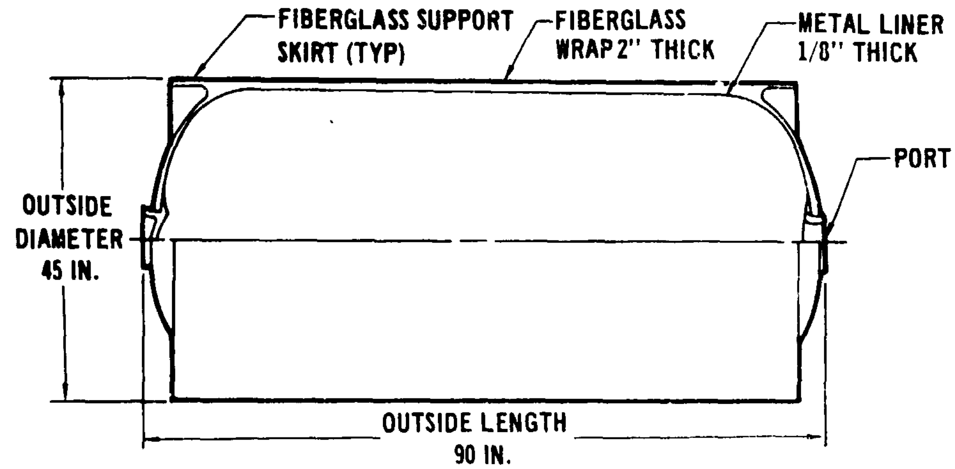
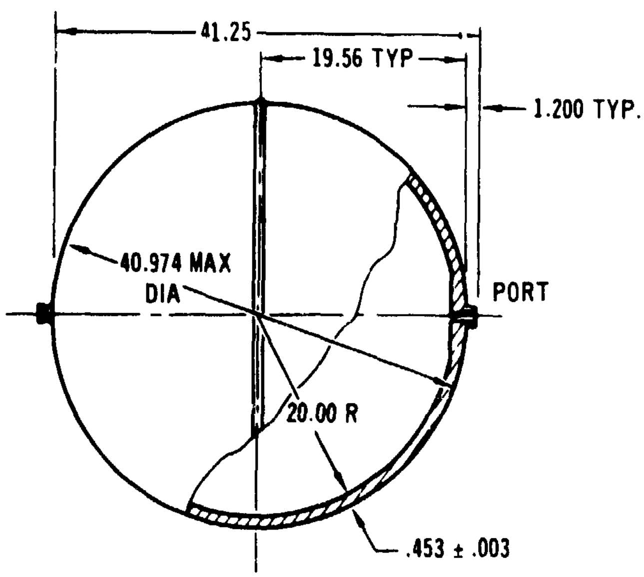


FIGURE 2.5-4 STS WINDOW ASSEMBLY

VOLUME	– 57 CU FT	936 LBS O_2 AT 3000 PSIG, 100°F (TOTAL)
WEIGHT (DRY)	– 2800 LB	822 LBS O_2 (USEABLE)
OPERATING PRESSURE	– 3000 PSIG	
RATED PRESSURE	– 4500 PSIG	
MATERIALS	– 321 ST. STL. LINER/347 ST. STL POLAR CAPS WITH FIBERGLASS WRAP	



OXYGEN TANKS



VOLUME	19.3 CU FT
WEIGHT	393 LB
OPERATING PRESSURE	3000 PSIG
MATERIAL	TITANIUM
251 LB N_2 @ 3000 PSIG, 100°F (TOTAL)	
220 LB N_2	(USABLE)

NITROGEN TANKS

FIGURE 2.5–5 OXYGEN AND NITROGEN TANKS

(2) Nitrogen - Nitrogen was stored in six tanks contained in three modules mounted on the AM trusses. Each tank had a fill valve, a check valve, two temperature transducers, and two pressure transducers. The nitrogen tanks were 40-inch diameter spheres constructed of titanium (Figure 2.5-5). A minimum design total of 1,520 lbs of nitrogen was required of which 1,329 lbs was usable at normal design flowrates. The calculated quantity actually loaded was 1,623 lbs.

Gas quantities were calculated by using tank volumes which were known and gas temperatures and pressures which were measured. On-board displays of temperature and pressure from each of the O_2 and N_2 storage tanks were provided on panel 225 (Figure 2.5-6).

D. Gas Supply - The gas system provided flows of oxygen and nitrogen at regulated pressure. The oxygen flow was used for initial pressurization, the two-gas control system and EVA/IVA support and was regulated to 120 $\pm$ 10 psig. The nitrogen flow was used for initial pressurization, the two-gas control system, molecular sieve valve actuation, water systems reservoir pressurization, OWS water system pressurization, and experiment support and was regulated to 150 $\pm$ 10 psig. Actual flowrates were a function of demand within the limits of regulator maximum flowrate capabilities. The design flow capacity of each of the two regulators within both the oxygen and nitrogen regulator assemblies was 22.8 lb/hr for an inlet pressure range of 3000 to 300 psig.

Both oxygen and nitrogen supply gas pressures were controlled by pressure regulator assemblies. Each assembly included an inlet filter, redundant regulators, and relief valves. Lockup pressures for the supply regulator assemblies were 145 psig for the O_2 assembly and 180 psig for the N_2 assembly. Should a regulator fail open, the maximum downstream pressures would have been limited to 170 psig oxygen and 210 psig nitrogen by the pressure relief valves. The failed regulator could be isolated by a manual upstream shutoff valve. The shutoff valve controls were located on panel 225 (Figure 2.5-6).

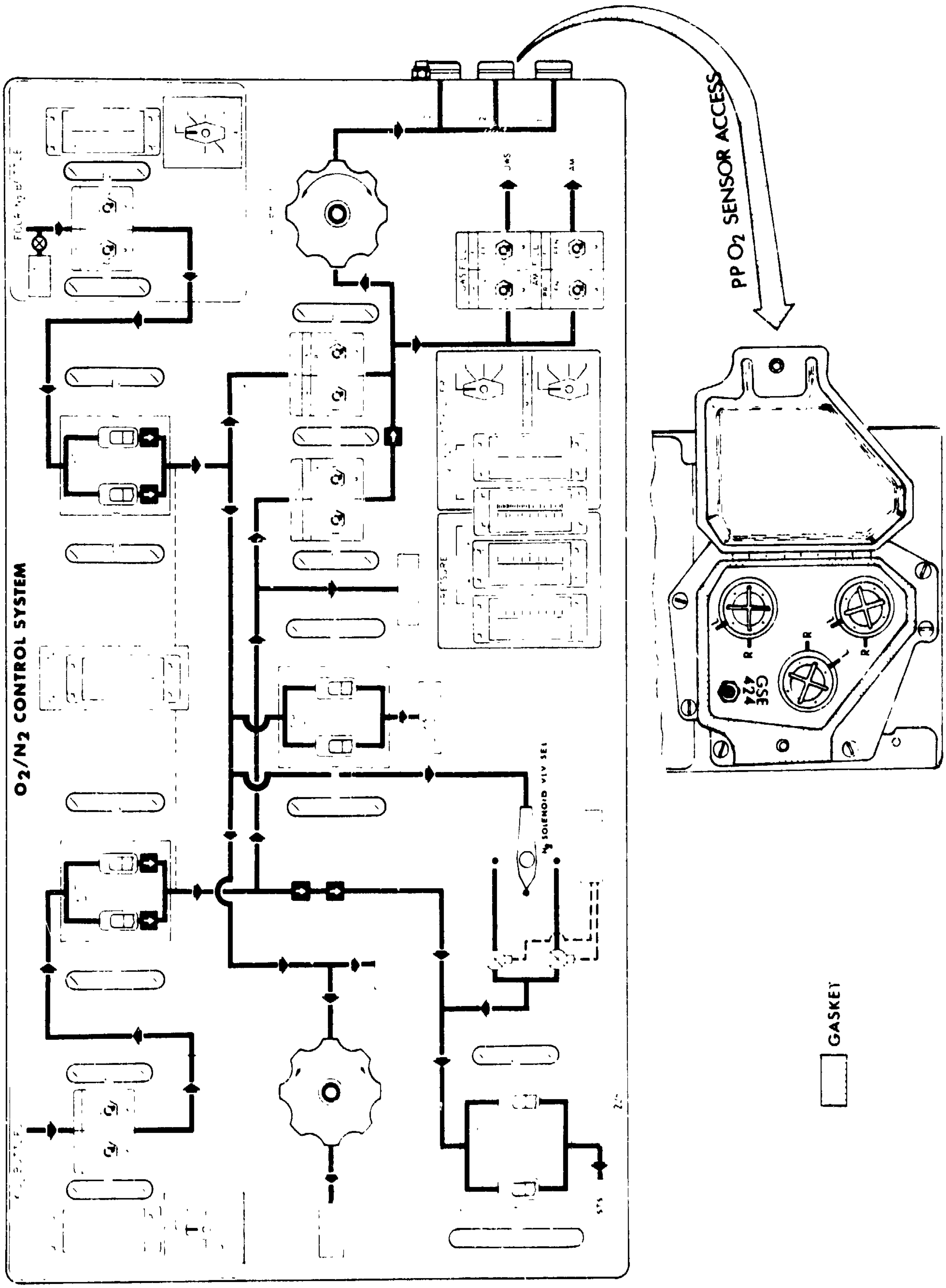


FIGURE 2.5-6 O_2/N_2 CONTROL PANEL 225

To isolate stored gas from the rest of the system when desired, redundant latching solenoid valves were provided in the high pressure O_2 and N_2 lines (Figure 2.5-2). The valves could be actuated by either on-board controls or by ground command. During orbital storage, the valves were normally closed except for cluster repressurization periods. The valves were open during manned missions. Orifices were provided in the high pressure lines downstream of the latching solenoid valves to limit flowrate into the cluster atmosphere to 5 lb/min maximum in the event of an internal line rupture. A bleed orifice was provided around the O_2 latching solenoid valves to reduce the pressure differential across the valves when the valves were closed; thereby, lowering the possibility of a fire hazard from high impact pressures upon opening the oxygen solenoid valves. The orifice size was based on the flowrate required to make up gas loss from allowable leakage of downstream components and still maintain pressure downstream of the latching solenoid valves close to the source pressure level.

(1) Initial Pressurization - Initial pressurization could be initiated by either ground command or on-board controls with automatic shutoff at 5.0 $\pm$ 0.2 psia for the ground command mode. Capability was available by selective operation of gas supply solenoid valves for pressurizing the AM/MDA and OWS separately or together with either oxygen, nitrogen, or the nominal 74%/26% volumetric O_2/N_2 mixture. Nominal design pressurization flowrates were 22.65 lb/hr oxygen and 6.95 lb/hr nitrogen. With these flowrates, the minimum time required for OWS and AM/MDA pressurization from 0.5 psia to 5.0 psia would be 9.3 hours and 1.7 hours, respectively. All solenoid valves (Figure 2.5-2) were normally closed when the Skylab was unmanned except during the pressurization sequence. On-board pressurization controls as well as displays of atmospheric pressure for the AM forward, lock and aft compartments and the OWS were provided on panel 225 (Figure 2.5-6).

(2) Mole Sieve Actuation - The gas subsystem provided pressurized gas to actuate the molecular sieve control valves. The gas used was nitrogen but molecular sieve cycling could be continued by using 120 psi oxygen if 150 psi nitrogen were not available. An operating molecular sieve

system required a flowrate of 2 lb/hr for approximately 8 seconds every 15 minutes. The controls were on mole sieve A and B bed cycle N_2 supply panels 221 and 219, respectively, also mole sieve A and B valve control panels 226 and 227, respectively.

(3) Reservoir Pressurization - The EVA/IVA and ATM C&D/EREP water cooling system reservoirs were at ambient pressure prior to launch and pressurized with nitrogen to 4.8 psia minimum, 6.2 psia maximum in orbit. Pressure was controlled by a regulator assembly containing an inlet filter, two shutoff valves, two regulators, two relief valves, and two outlet filters. Reservoir pressurization controls were located on panel 225, shown on Figure 2.5-6, reservoir pressurization valve panels 223 and 224, and ATM coolant reservoir pressurization panel 235.

(4) EVA/IVA Support - Oxygen was provided to the EVA/IVA system from 120 psig nominal regulators. Control of the regulated O_2 supply was from panel 225 shown on Figure 2.5-6. The system provided atmospheric gas to repressurize the AM lock compartment after EVA. The EVA/IVA system is discussed in Section 2.6 of this report.

(5) Experiment Support - Experiment support included support to the M509/T020 interface with the AM and to the M171, M092, and OWS water bottle pressurization interface with the OWS.

- M509/T020 Interface - Support of the M509/T020 experiments required recharging of portable tanks with nitrogen to a pressure as close to 3000 psia as possible. To accomplish this, provisions were made for keeping two of the six 3000 psi nitrogen tanks isolated as long as possible for experiment tank top-off. The two nitrogen tanks isolated for M509/T020 top-off could be connected to the other four nitrogen tanks, if required. A M509/T020 recharge station was provided in the AM aft compartment, Figure 2.5-2. The recharge station included hand valves for selecting the N_2 pressure source and a quick disconnect fitting for attachment to a detachable recharge umbilical which interfaced with a GFE experiments N_2 tank. The valves were located on aft compartment control panel 390. Orifices in the high pressure N_2 lines limited flowrate to 5.7 lb/min maximum.

- OWS Interface - For displacement of water from the OWS water tanks and support of the M171 and M092 experiments, N_2, regulated to 150 psi nominal pressure, was supplied to the OWS via a hardline. See Figure 2.5-2. The supply of N_2 to the interface was controlled from Panel 225 shown on Figure 2.5-6 Nominal N_2 flowrate to the OWS was 0 to 3.0 lb/hr, with the maximum limited to 13.5 lb/hr by an orifice in the line.

E. Atmospheric Pressure Control - During manned operation, the atmospheric total pressure was maintained between 4.8 and 6.0 psia, and O_2/N_2 composition was controlled automatically by the two-gas control system maintained atmospheric total pressure and oxygen partial pressure during manned operation. During orbital storage, cabin pressure was maintained by the Initial Pressurization System. Overpressure protection was provided by cabin pressure relief valve assemblies located in the AM forward, lock, and aft compartments.

Should cabin pressure drop within the range of 4.5 to 4.7 psia, a C&W system alarm would be actuated. An alarm would also occur should the cabin pressure decay rate equal or exceed 0.1 psi/min. In addition, a low oxygen partial pressure C&W alarm was provided. The sensing for this alarm was integrated with the two-gas control system.

(1) Two-Gas Control System - The two-gas control system automatically controlled atmospheric pressure at 5.0 $\pm$ 0.2 psia and oxygen partial pressure at 3.6 $\pm$ 0.3 psia. It consisted of two check valves, redundant cabin pressure regulators, a selector valve, two solenoid valves, an O_2/N_2 composite controller, three PPO_2 sensors, a sensor calibration housing, an orifice, a manual shutoff valve, and various lines and fittings (Figure 2.5-2). The redundant cabin pressure regulators maintained total pressure at 5.0 $\pm$ 0.2 psia. Flow capacity from each regulator and from the regulator assembly was 1.15 $\pm$ 0.15 lb/hr of gaseous oxygen at an outlet pressure of 4.8 to 5.2 psia. The flowrate tolerance was kept small so that cabin leakage slightly in excess of the maximum allowable would cause a decrease in cabin pressure which would be more readily detectable than a decrease in supply quantity. Flow could be shut off from either one or both cabin pressure regulators by closing manual valves

located upstream of each regulator. The cabin pressure regulator flowrate characteristics are shown in Figure 2.5-7. Cabin pressure control would be approximately 4.9 psia at the design gas makeup supply rate of 23.5 lb/day (consisting of a 6 lb/day metabolic, 14 lb/day cluster leakage, and 3.5 lb/day molecular sieve gas loss).

Oxygen partial pressure was sensed as a basis for supplying either oxygen or nitrogen to the cabin pressure regulator. When the PPO_2 reached the control range upper end, nitrogen was supplied; when it reached the lower end, oxygen was supplied. The cluster PPO_2 was controlled to a nominal 3.6 psia. Considering the controller band maximum width and the $\pm 3\%$ tolerance of the sensor/amplifier, the cluster PPO_2 could vary from 3.3 psia minimum to 3.9 psia maximum as shown in Figure 2.5-8. Three sensor/amplifier, controller systems were provided. One was used for control, another for monitoring, and the third for backup. Three PPO_2 display gages were provided on panel 225, as shown in Figure 2.5-6. Each indicator had a range of 0 to 6 psi, with an accuracy of $\pm 2.0\%$ of full scale. A C&W alarm would be initiated by either the monitoring or controlling sensor at a nominal alarm point of 3.05 psia PPO_2. As shown in Figure 2.5-8, the low PPO_2 alarm band was 2.81 to 3.28 psia because of system tolerances.

Provisions for in-flight calibration could be used to test the PPO_2 sensors. This capability was provided by a hinged sensor calibration shroud and a valve in the AM/MDA/OWS pressurization line to allow flow of either pure O_2 or N_2 gas over the PPO_2 sensors. The indicated PPO_2 should equal cabin total pressure when flowing O_2 and zero for N_2. Tests have demonstrated that check point stabilizations would occur within 2 minutes after gas flow was initiated. The Skylab was launched with the calibration shroud covering the sensors. The crew moved it to the open position during SL-1/2 activation where it remained throughout the mission.

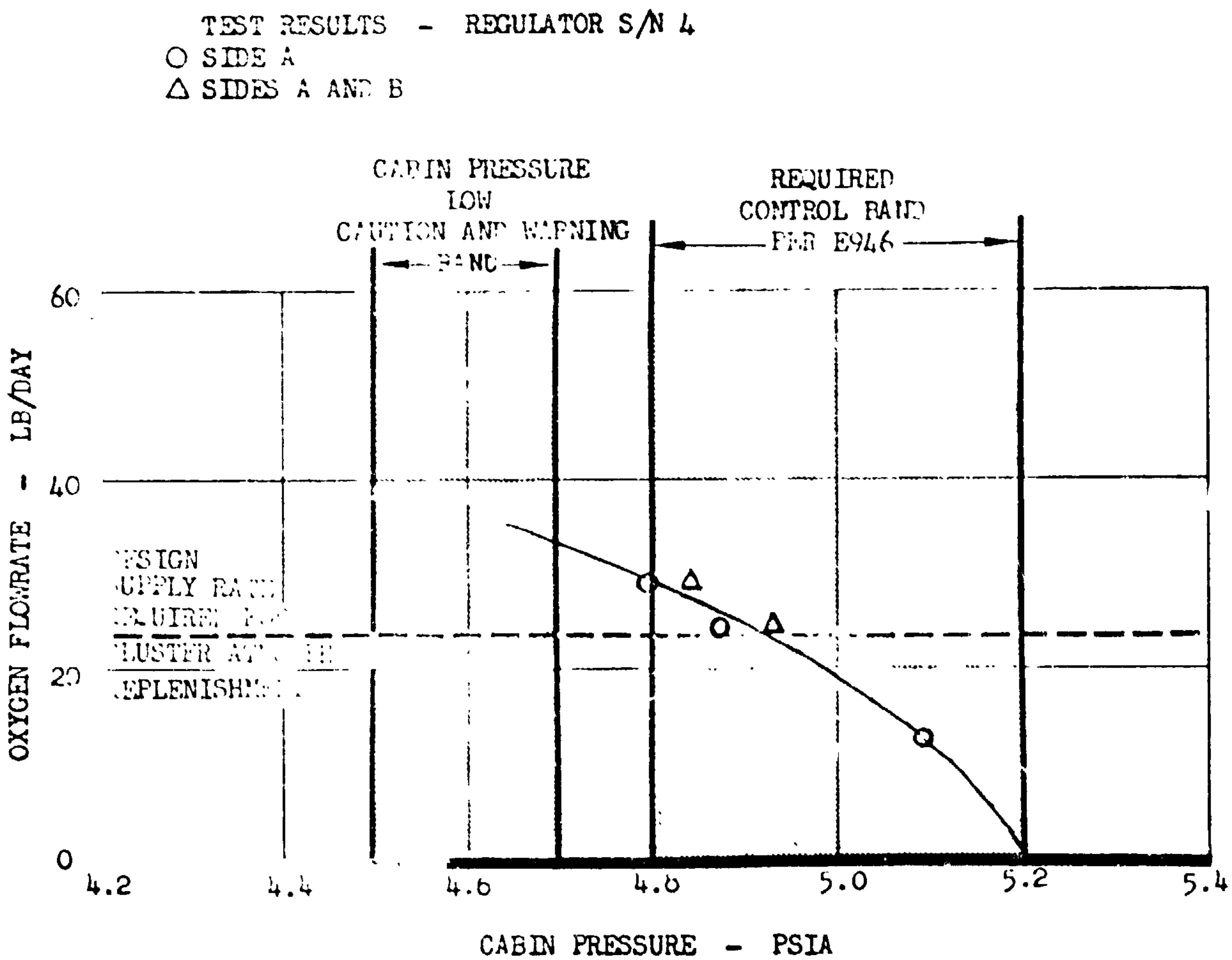
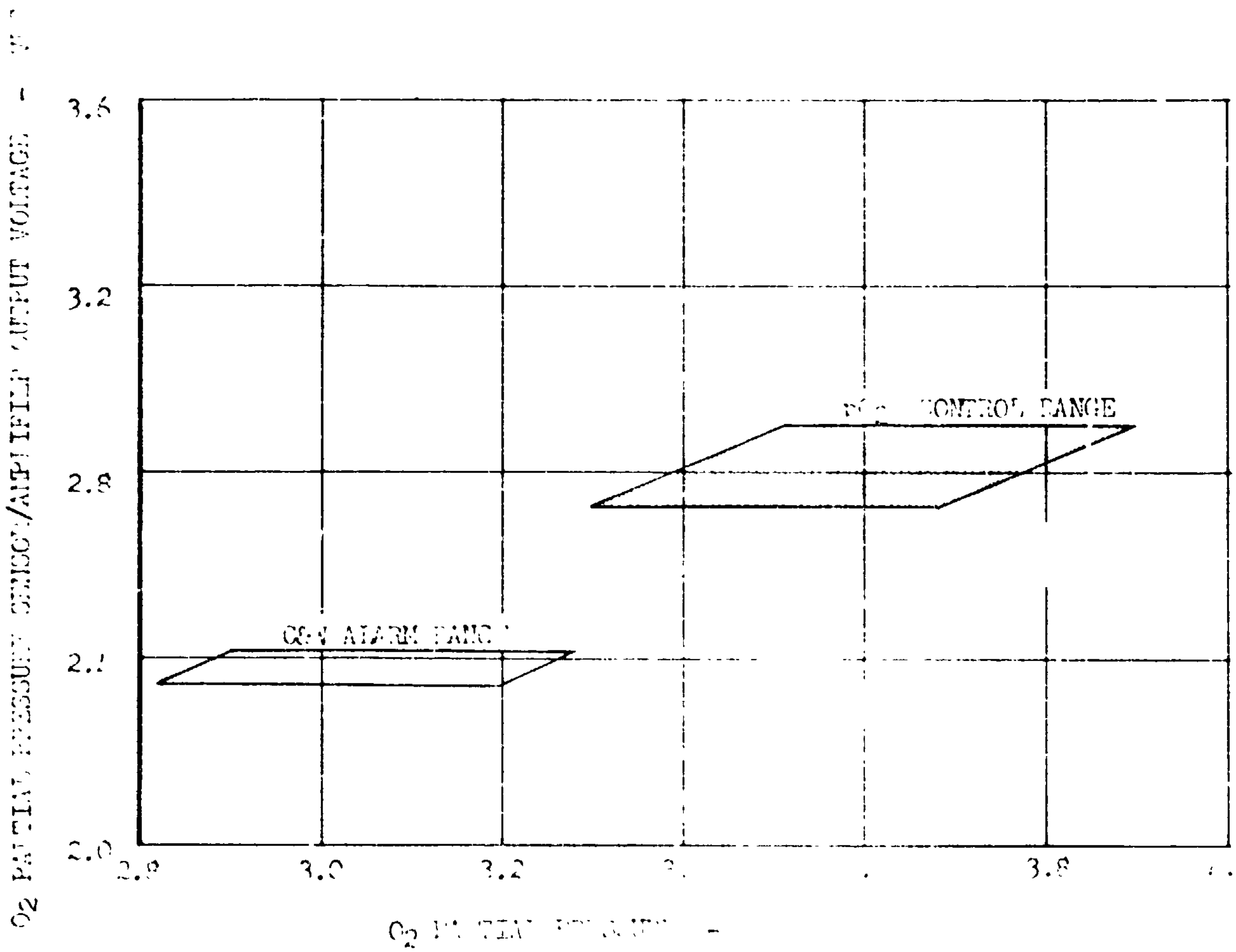


FIGURE 2.5-7 CABIN PRESSURE REGULATOR FLOWRATE CHARACTERISTICS



NOTES: 1. PPO_2 SENSOR/AMPLIFIER ACCURACY OF ± 3%

2. CONTROLLER MAX DEAD BAND OF 170 MV

FIGURE 2.5-8 CONTROL AND ALARM RANGES FOR TWO GAS CONTROL SYSTEM

The Skylab two-gas control system was designed to be "fail-safe". Pure oxygen would be supplied to maintain total pressure in case of electrical power failure, most types of solenoid valve failures, and PPO_2 sensor degradation. In addition, redundant oxygen check valves, redundant nitrogen solenoid valves, and a nitrogen selector valve provided the capability for corrective action in the event of a valve failure.

(2) Cabin Pressure Relief Valve - The maximum total cabin pressure was limited to 6 psig by cabin pressure relief valve assemblies located in the forward, lock and aft compartments. Each relief valve assembly contained two relief valves in parallel which were arranged in series with a manually actuated shutoff valve. The controls for these shutoff valves are located on control panels 300, 313, and 391, Figure 2.5-9.

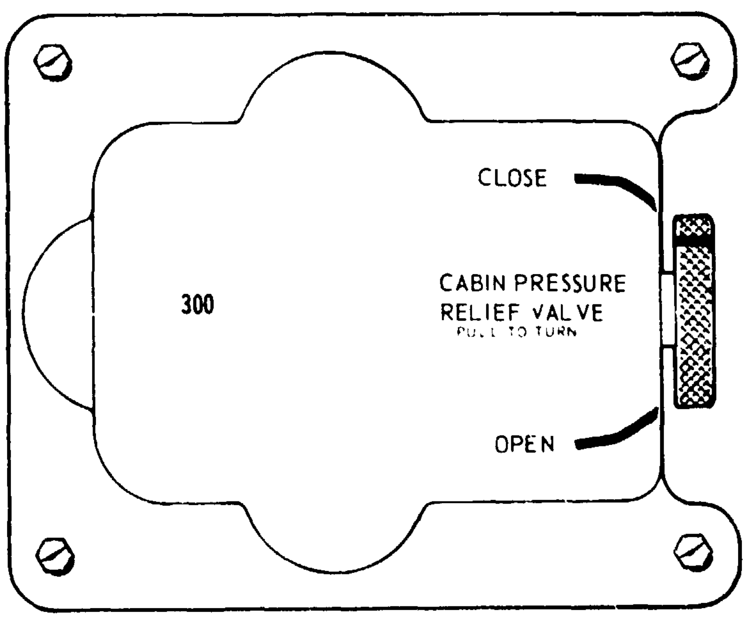


FIGURE 2.5-9 FORWARD COMPARTMENT PRESSURE RELIEF VALVE

2.5.2.2 Atmospheric Control System

The atmospheric control system, shown in Figure 2.5-10, provided humidity control, carbon dioxide and odor removal, ventilation, and cabin gas cooling. Moisture was removed from the atmosphere by condensing heat exchangers and molecular sieve systems located within the STS. Carbon dioxide and odor were removed by the molecular sieve system. Ventilation was provided by PLV fans and molecular sieve gas compressors. Acoustic noise suppression was provided by mufflers. Gas cooling was provided by condensing and cabin heat exchangers. Solids traps upstream of the molecular sieve compressors, as well as six-mesh screens upstream of the PLV fans, provided protection from particulate matter. Replacement solids traps, PLV fans, and a spare molecular sieve fan were stored onboard.

A. Humidity Control - Humidity was controlled by the removal and separation of excess moisture from the atmosphere. Humidity control was performed by the condensing heat exchangers with support from the coolant and condensate systems.

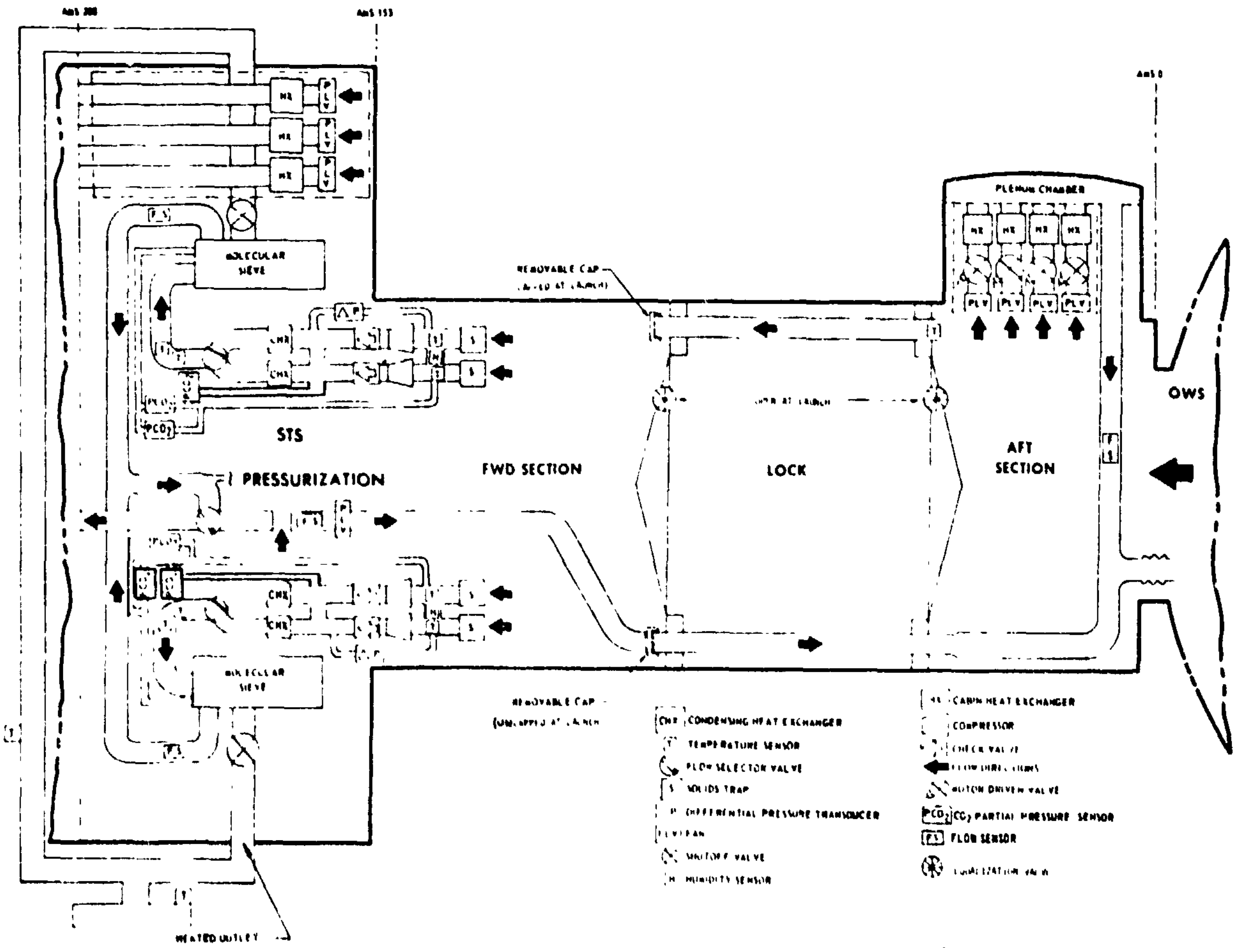


FIGURE 2.5-10 ATMOSPHERIC CONTROL SYSTEM

(1) Moisture Removal - Excess water vapor was removed from the atmosphere circulated through the condensing heat exchangers by condensation. The condensation temperature or dew point was controlled by controlling the temperature of the fins within the heat exchanger core. Core temperature was controlled by the coolant system which controlled the temperature of coolant supplied to the condensing heat exchangers.

Preflight analysis showed that the cluster dew point temperature would be at a very low initial level for each mission, as shown in Figures 2.5-11 and -12, because dry gas was used to pressurize the cabin. After activation, dew point levels would increase due to crew water generation rates. The system was designed to maintain dew point temperatures within the allowable range of 46°F and 60°F after activation.

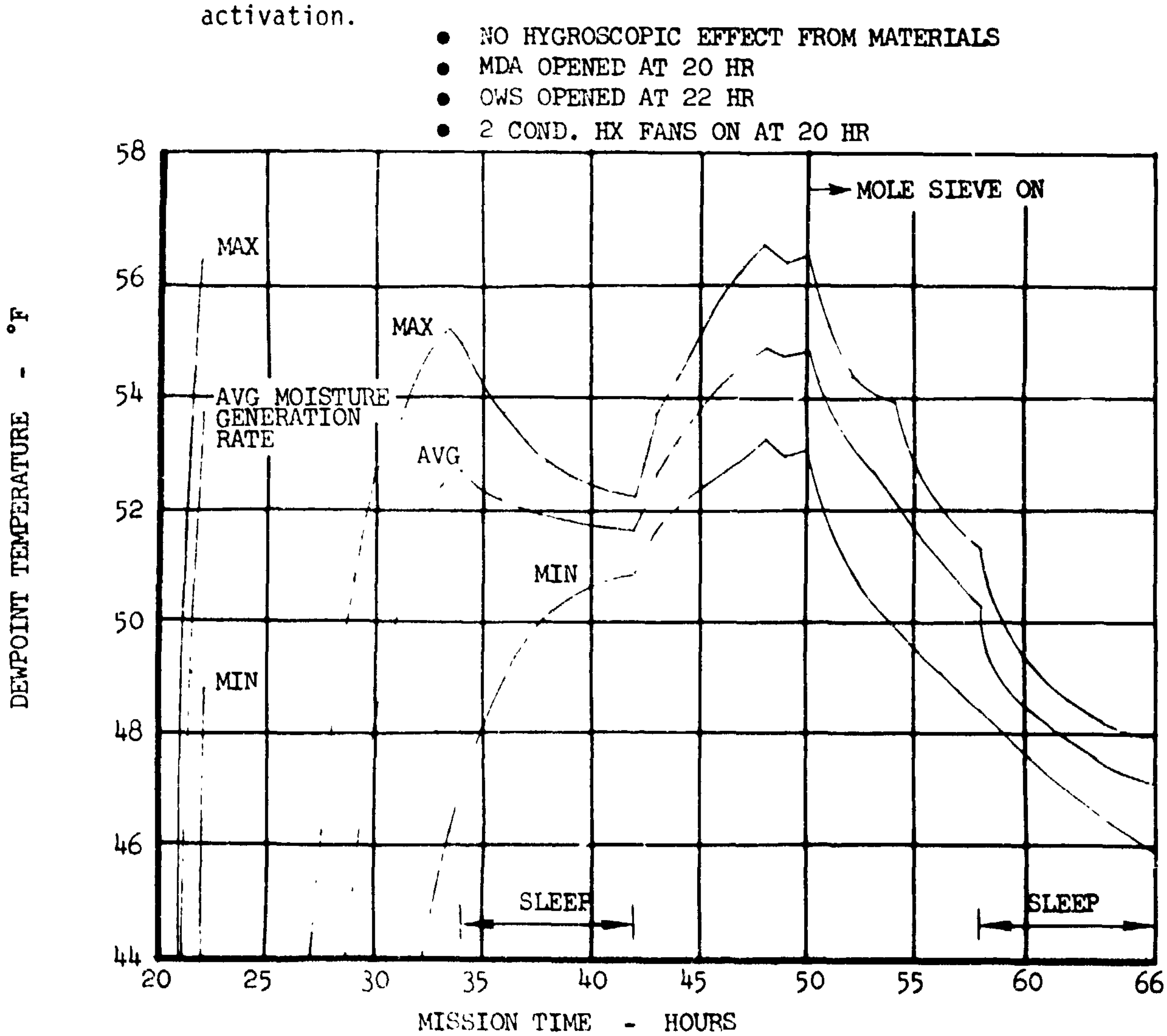


FIGURE 2.5-11 DEWPOINT TEMPERATURE DURING ACTIVATION

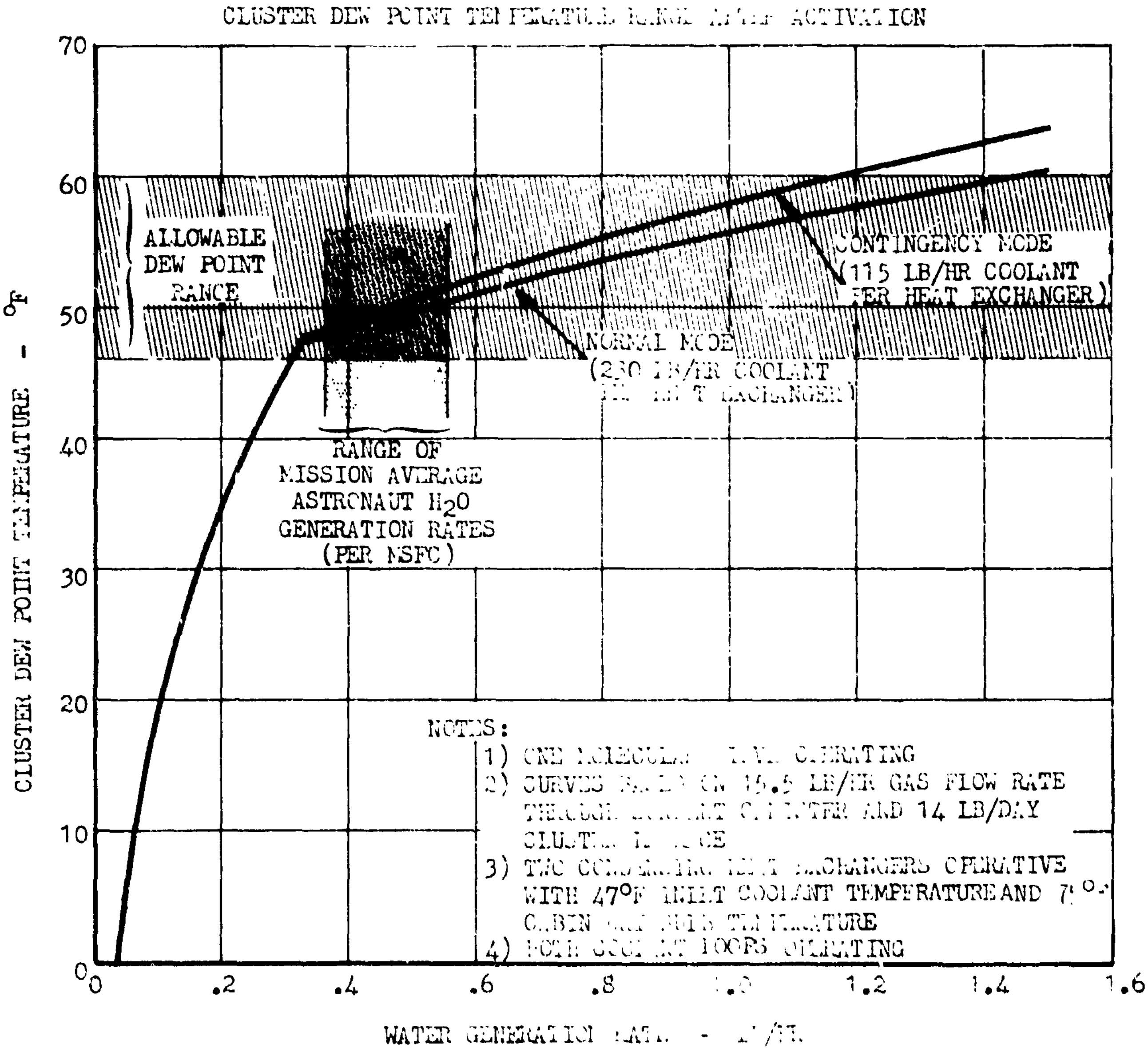


FIGURE 2.5-12 CLUSTER DEWPOINT TEMPERATURE RANGE AFTER ACTIVATION

Four condensing heat exchangers were provided (two in parallel upstream of each molecular sieve). Normal procedure was to operate one condensing heat exchanger upstream of each molecular sieve assembly. The two remaining heat exchangers were redundant. The moisture removal portion of the atmosphere control system was operated from ECS control panel 203 shown in Figure 2.5-13, circuit breaker panel 200, molecular sieve condensing heat exchanger control panels 230 and 232 shown in Figure 2.5-14 and also molecular sieve condensing heat exchanger air flow valves 233 and 239 shown in Figure 2.5-15. The operation of all valves shown in Figures 2.5-14 and 2.5-15 must be integrated in order to assure proper system operation.

(2) Condensate Separation - Within each condensing heat exchanger, condensate was transported from the fins to the wicking by surface tension and from the wicking to the water separator assemblies by capillary action. The water separator plate assemblies served to hold back the atmosphere and allow passage of condensate into the condensate system. The design minimum water removal rate from one condensing heat exchanger was 0.78 lbs/hr for a gas to condensate pressure differential of 8" H_2O. Each water separator plate assembly contained two plates whose faces were covered with fritted glass, see Figure 2.5-16. The pores in the fritted glass were sized to be sealed against 6.3 psi cabin to condensate pressure differential by the surface tension of water.

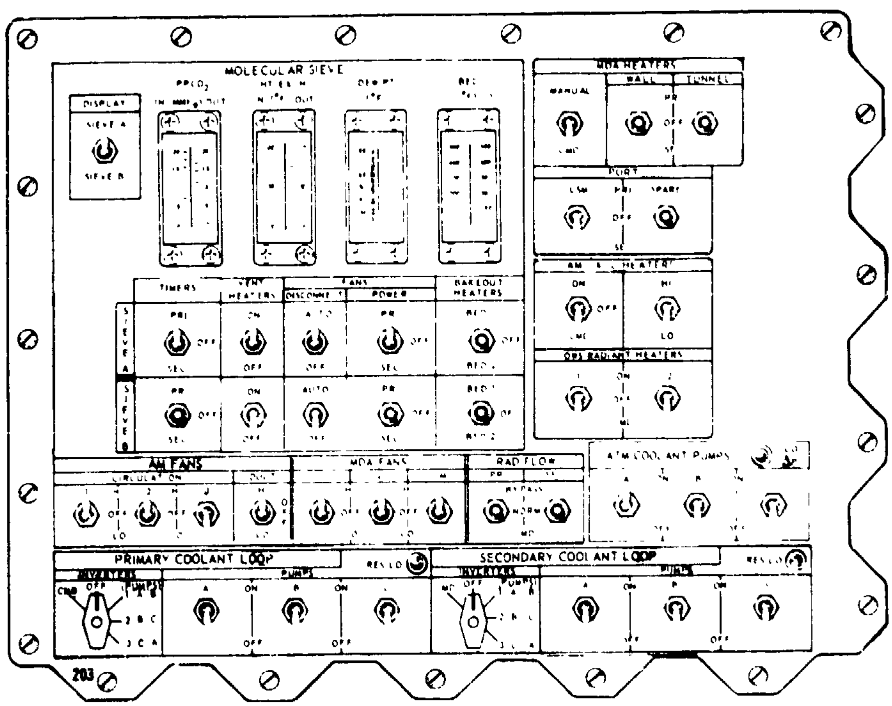


FIGURE 2.5-13 ECS CONTROL PANEL 203

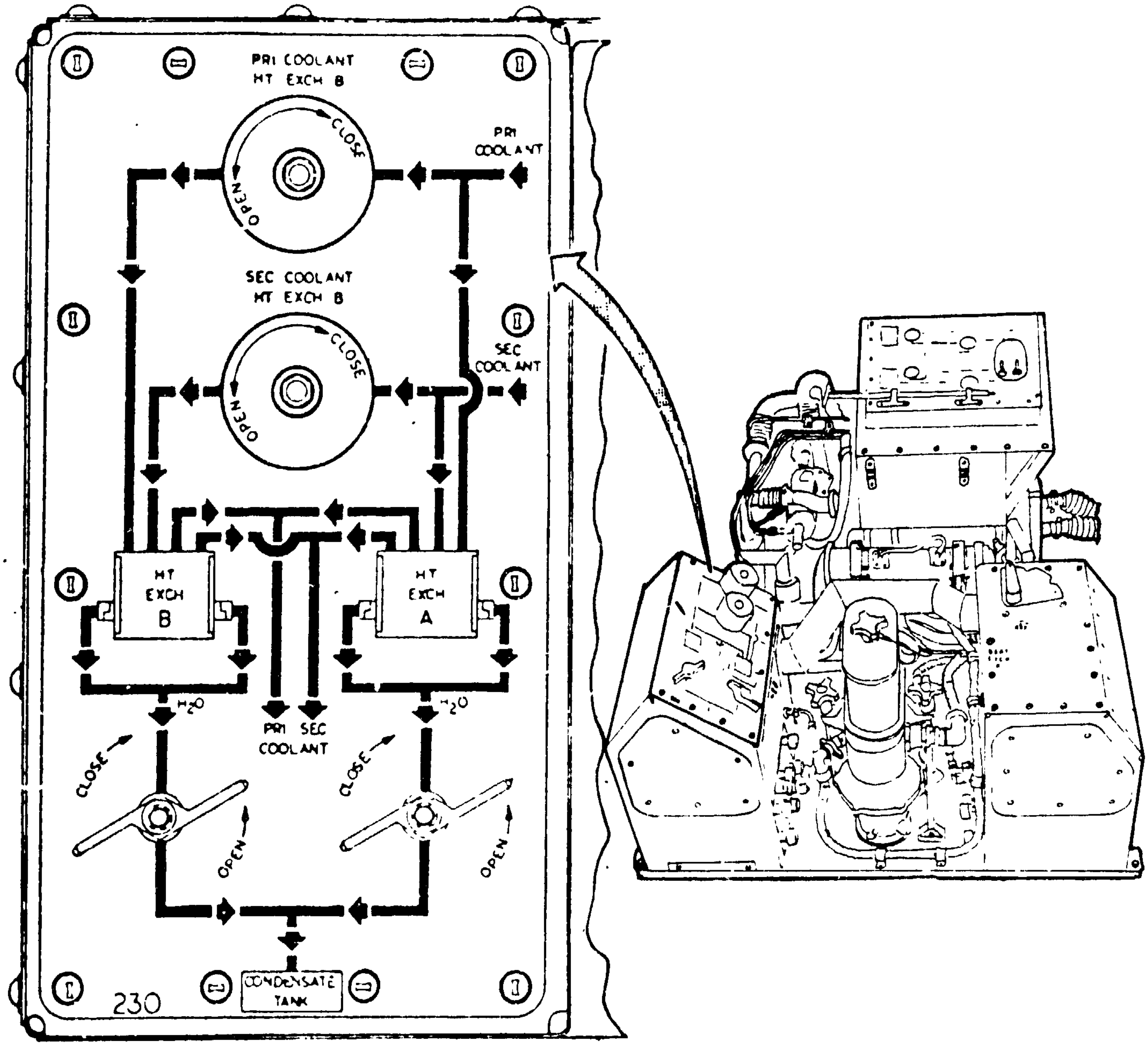


FIGURE 2.5-14 MOLECULAR SIEVE CONDENSING HEAT EXCHANGER CONTROL PANELS (230 FOR SIEVE A AND 232 FOR SIEVE B)

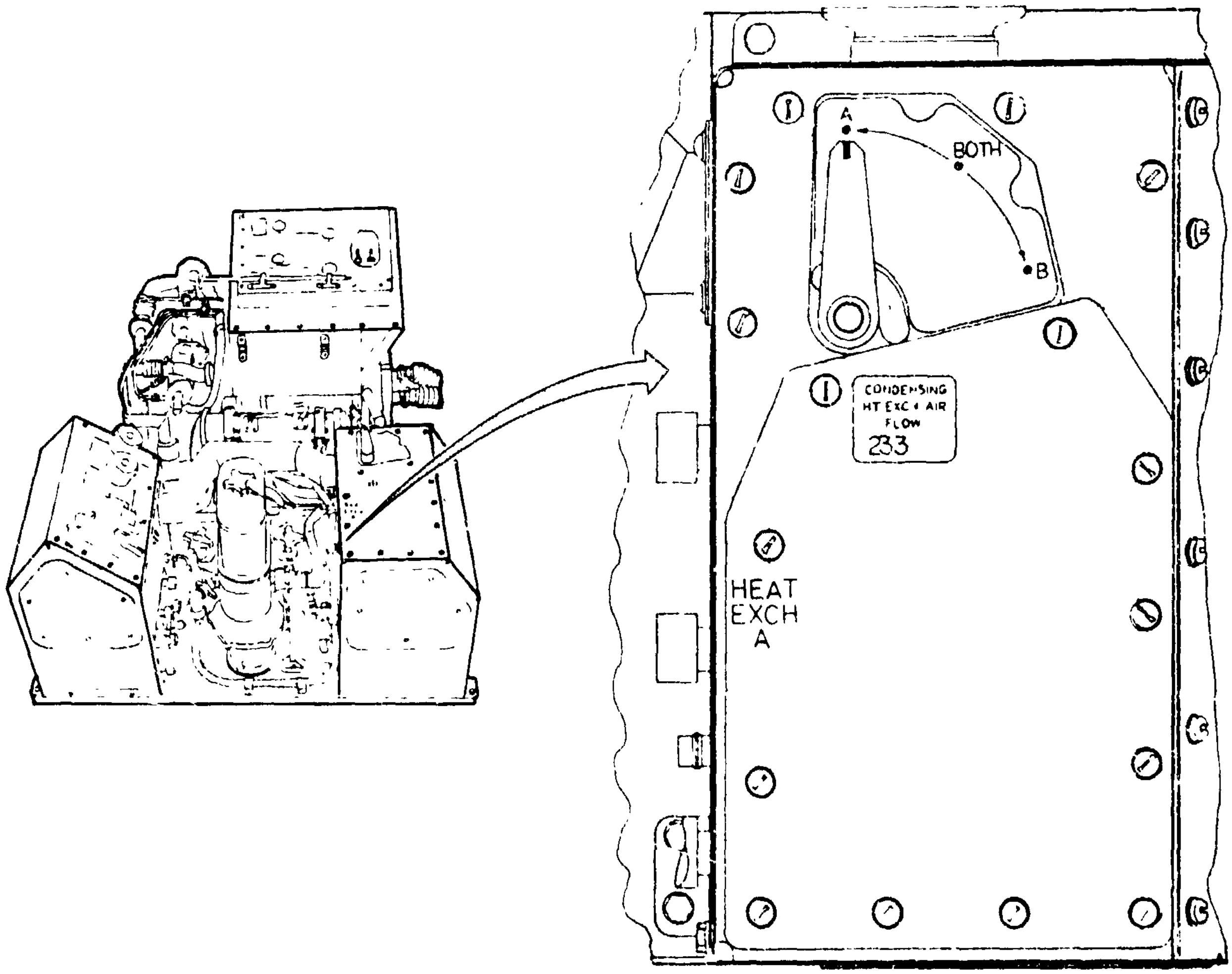


FIGURE 2.5-15 MOLECULAR SIEVE CONDENSING HEAT EXCHANGER AIR FLOW VALVE
(233 FOR SIEVE A AND 239 FOR SIEVE B)

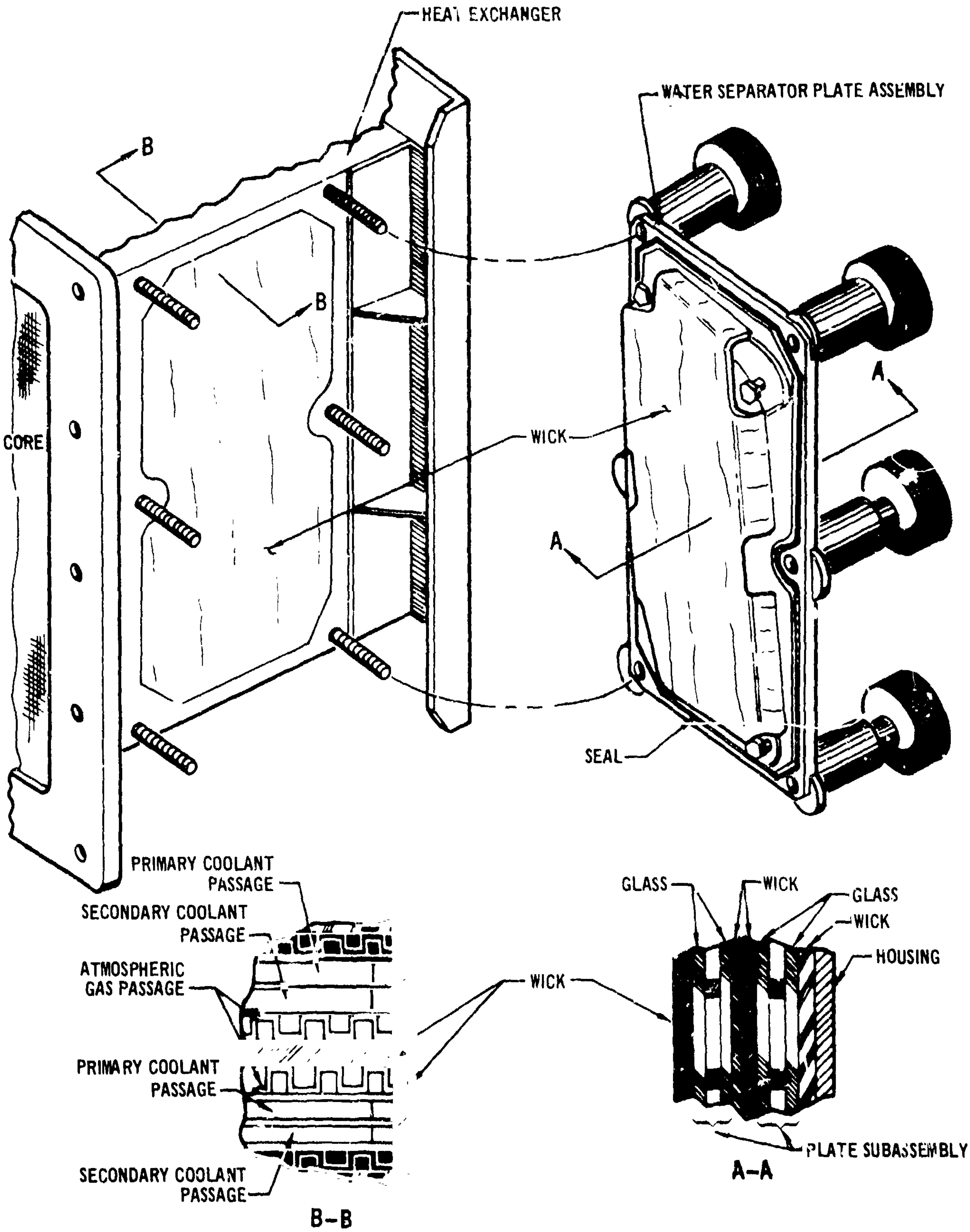


FIGURE 2.5–16 CONDENSING HEAT EXCHANGER

Each condensing heat exchanger contained two in-flight replaceable water separator plate assemblies. An additional four water separator plate assemblies were carried in permanent stowage container No. 202 for use as in-flight replacements should a problem develop. Plates were not replaced on a scheduled basis. All twleve water separator plate assemblies were launched dry with both sides of the plates vented to cabin. Plates installed on operating heat exchangers were serviced in orbit by the servicing procedure of Section 2.5.2.4. After servicing, the pressure within the condensate system was sufficiently low to allow moisture condensed in the heat exchangers to be forced through the heat exchanger water separator plate assemblies and transferred into the condensate system by compartment ambient pressure.

B. Purity Control - The atmosphere was purified by the removal of carbon dioxide, odors and other trace contaminants produced by metabolic generation and off-gassed from materials exposed to the Skylab atmosphere. The design rate for carbon dioxide removal was 6.75 lb/day to maintain the partial pressure of carbon dioxide in the Skylab atmosphere at 5.5 mmHg or less. The capability of removing methane and hydrogen sulfide was necessary to control odors produced by crew members as well as its ability to control the following fifteen trace contaminants:

1. Ammonia	6. Xylene	11. Methyl Isobutyl Ketone
2. Methyl Chloride	7. Toluene	12. Dichloromethane
3. Freon 12	8. Acetone	13. Methyl Chloroform
4. Benzene	9. Isopropyl Alcohol	14. Methyl Ethyl Ketone
5. Freon 113	10. Acetaldehyde	15. Coolanol 15

Two molecular sieve systems (Figure 2.5-10) of the adiabatic desorb type were provided for purity control.

Each molecular sieve system contained two sorbent mole sieve canisters and a charcoal canister, Figure 2.5-17. Each sorbent canister was provided with a pneumatically actuated gas selector valve which could cycle the sorbent beds alternately from an adsorb to a desorb mode automatically at 15-minute intervals. The selector valves were actuated by 150 psig nitorgen flow through solenoid valves which were opened electrically by signals from redundant cycle timers. A C&W caution alarm would be actuated should a cycle timer power interrupt greater than 30 milliseconds occur.

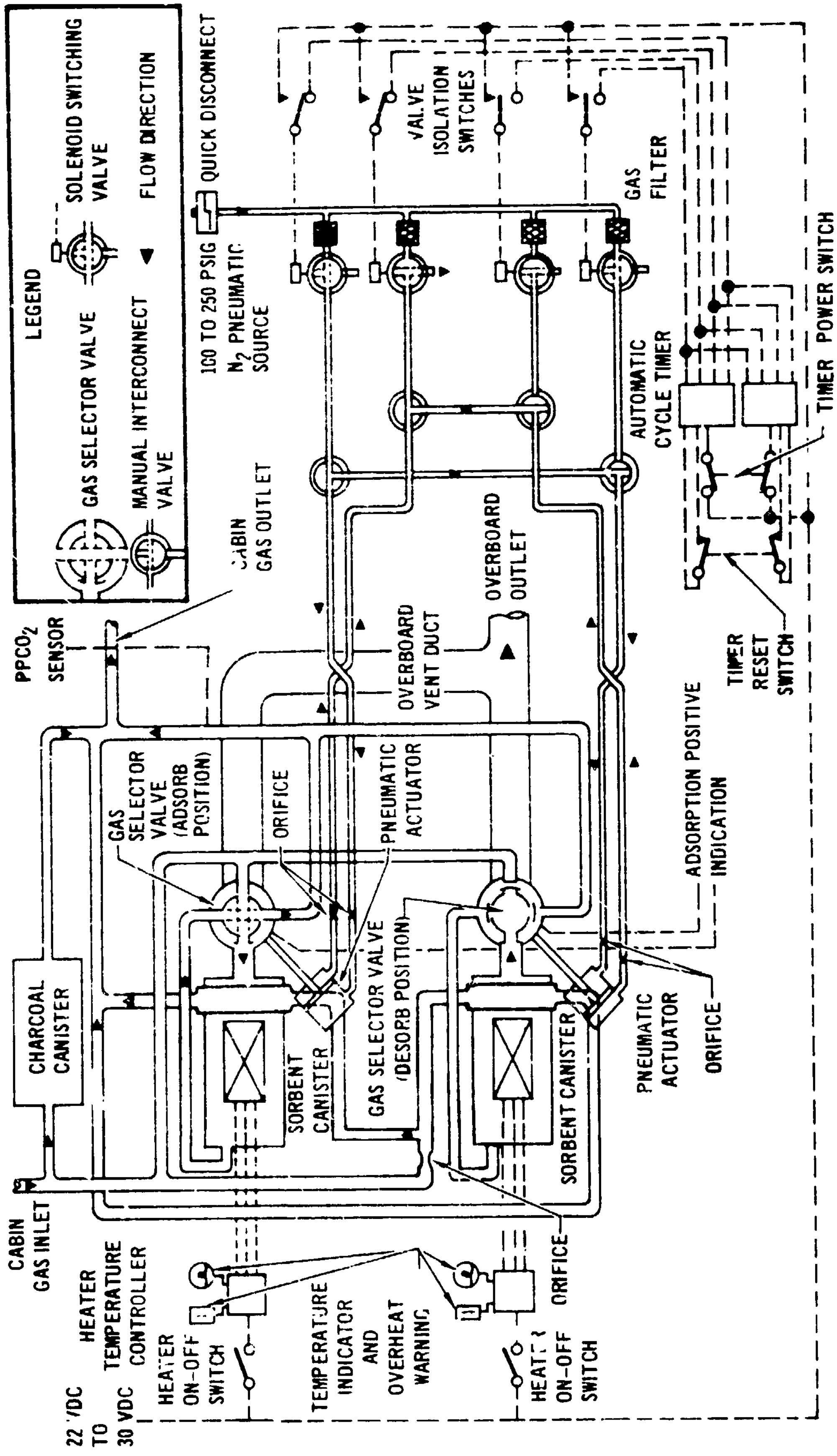


FIGURE 2.5–17 SINGLE MOLECULAR SIEVE SYSTEM

Selector valve cycling required a flowrate of 2 lb/hr for approximately 8 seconds every 15 minutes. The sorbent canisters contained Linde Type 13X molecular sieve material for water vapor adsorption and Linde Type 5A molecular sieve material for CO_2 adsorption. Carbon dioxide water vapor, and some trace contaminants were removed from cabin gas flowing through the sorbent canister in the adsorb mode while the canister in the desorb mode was regenerated by exposure to vacuum.

The charcoal canisters were provided for odor and trace contaminant control. A list of materials removed by the canister is presented in Appendix I. Each charcoal canister was designed for in-flight replacement and contained 9 lbs of activated charcoal. It was scheduled for replacement at 28-day intervals.

Sorbent canister operation of only one system was required for CO_2 removal. The two inoperative sorbent canisters were provided for redundancy. Cabin gas flowing into the operative system (5..5 lb/hr) was routed into three parallel paths: (1) sorbent canister flow for CO_2 removal (15.5 lb/hr), (2) charcoal canister flow for odor removal (18.2 lb/hr), and (3) bypass flow for flow balancing and cooling the sorbent canister housings during bakeout periods (19.8 lb/hr) .

The sorbent canisters of the inoperative molecular sieve were isolated from both cabin atmosphere and vacuum by an intermediate position of the gas selector valve. Cabin gas flow entering the inoperative mole sieve (45.9 lb/hr) went through only the charcoal canister (21.8 lb/hr) and the bypass line (24.1 lb/hr).

The sorbent beds were to be baked out to increase CO_2 removal performance by elimination of moisture accumulated in the downstream portion of the sorbent canisters. Moisture was removed by heating the beds sequentially to between 360°F and 410°F by internal electrical bakeout heaters while the beds were exposed to vacuum. Should bed temperatures increase beyond this range, a C&W alarm would be actuated between 425°F to 450°F to warn the crew of the over-temperature condition. Scheduled bakeouts were 24 hours per canister during prelaunch, 5 hours during the first mission activation, and 6 hours during the second and third activations. Bakeout during the manned mission was recommended if the cabin $PPCO_2$ corrected reading was 6.0 mmHg after a verification check. The time between bakeouts was to be 28 days minimum.

Each molecular sieve system had redundant cycle timers. It had manual interconnect valves which provided isolation and permitted use of redundant solenoid switching valves. In addition, each sorbent canister had a separate bakeout heater temperature controller. In the unlikely event that more than one sorbent canister could not be operated, the inlet $PPCO_2$ could have been limited to 12 mmHg maximum with one canister operation for the design CO_2 removal rate of 6.75 lb/day and an inlet dew point temperature of 52°F.

The carbon dioxide and odor removal portion of the atmospheric control system was operated from panel 203 shown in Figure 2.5-13, circuit breaker panel 200, molecular sieve B and A vent valves panels 218 and 220, respectively, shown in Figure 2.5-18, and molecular sieve A and B valve control panels 226, 227, 228, and 229 shown in Figure 2.5-19. Molecular sieve operating modes are shown in Figure 2.5-20.

Cabin CO_2 partial pressure was measured at the inlet to both operating condensing heat exchangers. Outlet CO_2 partial pressure was measured by the two sensors at the outlet of sorbent canisters of the operating molecular sieve system. Normally no outlet $PPCO_2$ readings were available for the molecular sieve which was in the "isolate" mode. Dummy elements which prevented flow through the transducer replaced the normal cartridge elements in these units. On-board display and TM were provided at both the inlet and outlet. The on-board display appeared on ECS control panel 203 shown in Figure 2.5-13. A C&W signal was provided when the outlet $PPCO_2$ reached 4.4 ± 1.4 mm Hg.

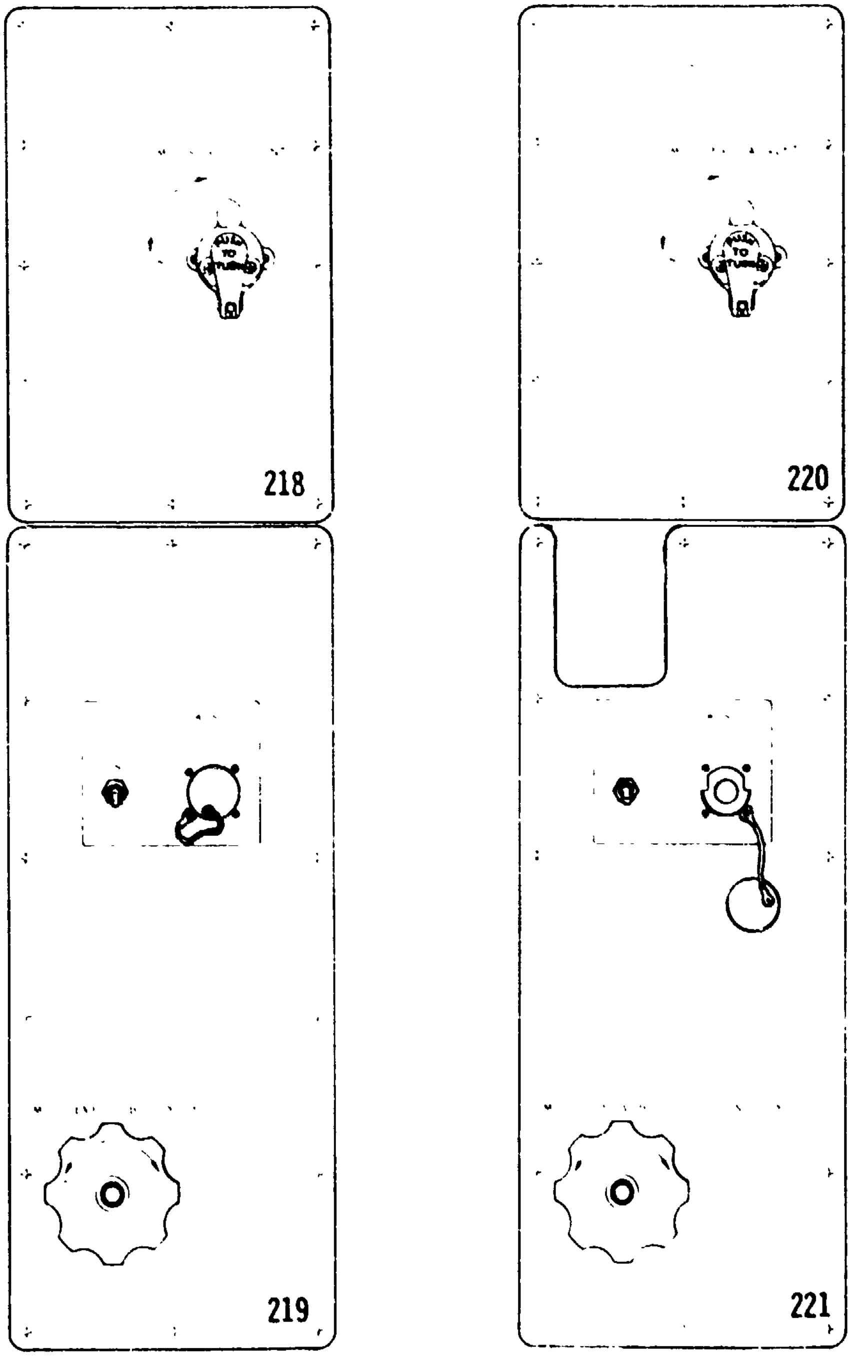


FIGURE 2.5-18 MOLECULAR SIEVE VENT VALVES AND BED CYCLE N_2 SUPPLY VALVES (220 AND 221 FOR SIEVE A, AND 218 AND 219 FOR SIEVE B)

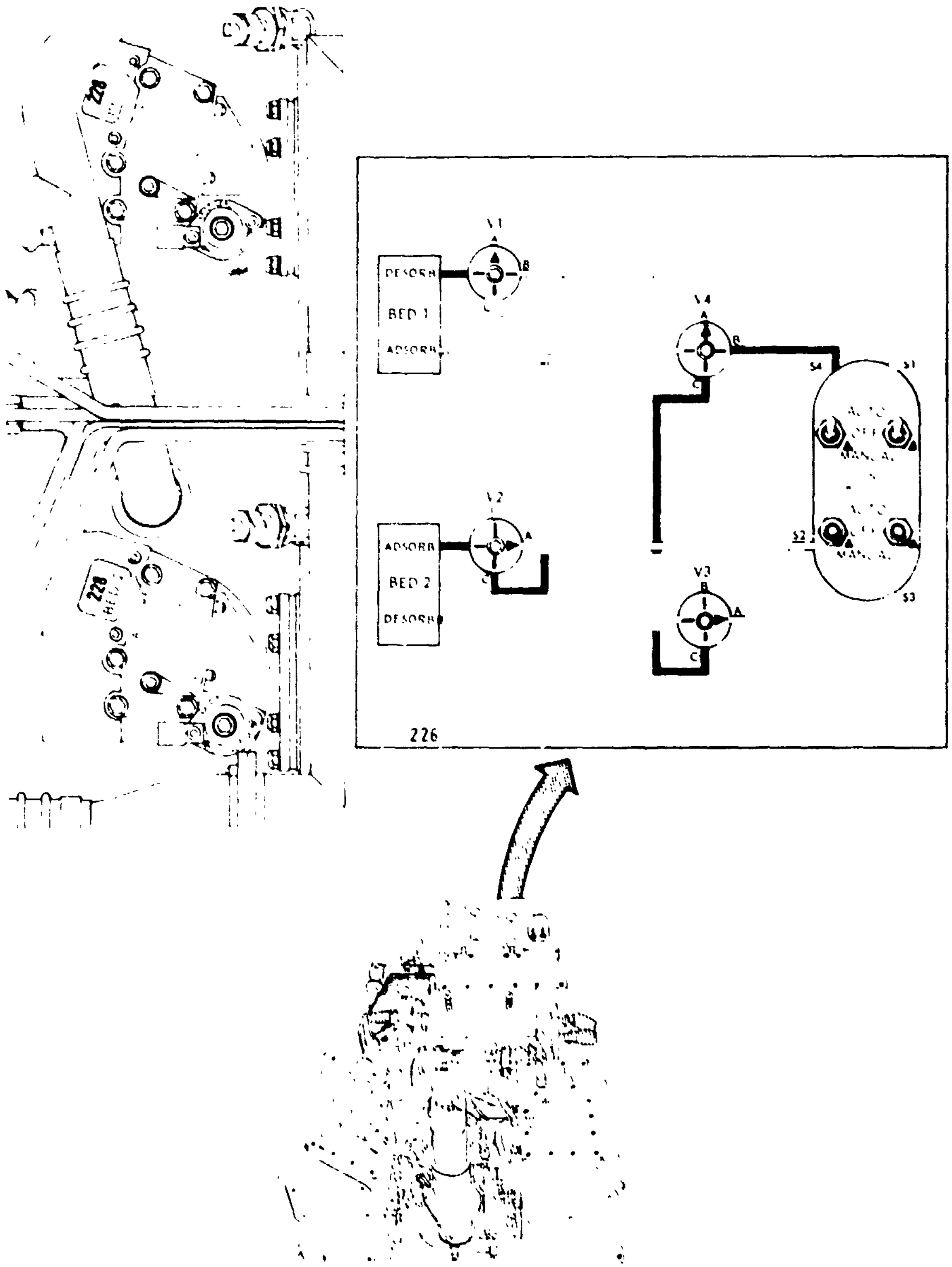


FIGURE 2.5–19 MOLECULAR SIEVE A VALVE CONTROL PANELS 226 AND 228 (227 AND 229 FOR SIEVE B)

MOL SIEVE [illegible]

OPERATIONAL MODE *	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
NORMAL	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	OFF	AUTO
ALT 1	B	B	[illegible]	[illegible]	[illegible]	AUTO	AUTO	OFF
ALT 2	A	A	B	[illegible]	[illegible]	[illegible]	AUTO	OFF
ALT 3	[illegible]	B	A	[illegible]	OFF	[illegible]	OFF	AUTO
NORMAL DESORB (BOTH BEDS)	MANUALLY POSITION [illegible] GAS SELECTOR VALVE [illegible] WITH [illegible] HANDLE				[illegible]	OFF	OFF	OFF
ALT. DESORB (BOTH BEDS)	C	C		[illegible]	[illegible]	[illegible]	MANUAL	OFF
BAKEOUT - BED 1	(MANUALLY POSITION [illegible] GAS SELECTOR VALVE TO "DESORB" WITH [illegible] HANDLE.)							
NORMAL - BED 2	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	AUTO	AUTO	OFF
BAKEOUT - BED 2	(MANUALLY POSITION [illegible] WITH [illegible] HANDLE.)							
NORMAL - BED 1	[illegible]	[illegible]	[illegible]		[illegible]	[illegible]	OFF	AUTO

*PUT ALL SWITCHES IN "OFF" BEFORE SETTING [illegible]

FIGURE 2.5-20 MOLECULAR SIEVE OPERATING INSTRUCTIONS (ON BACKSIDE OF DOOR – TYPICAL FOR SIEVES A AND B)

The $PPCO_2$ transducer divided the sample flow into two streams, filtered and ionized each stream, and compared the ion currents in a bridge circuit to obtain the measurement. Maximum specification inaccuracy of the transducers was ±1.4 mmHg. A diagram of the sensor and filter cartridges is shown in Figure 2.5-21. The filter cartridges were installed in the sensor shown in Figure 2.5-22. The CO_2 cartridges in the transducers were to be replaced at regular intervals as defined in Figure 2.5-23. CO_2 cartridge minimum lifetimes were 14 days and 28 days, respectively for the molecular sieve inlet and outlet locations for the range of dew point temperatures and CO_2 partial pressures expected at those locations. The stowage provisions are shown in Figure 2.5-24.

A verification check for each inlet $PPCO_2$ detector assembly could be obtained by comparing the detector assembly telemetry output with the values obtained from the mass spectrometer included in the M171 experiment. Also, a calibration cartridge was available which permitted a zero check of the $PPCO_2$ detector. Two spare $PPCO_2$ detector end plates were stored in stowage container M202.

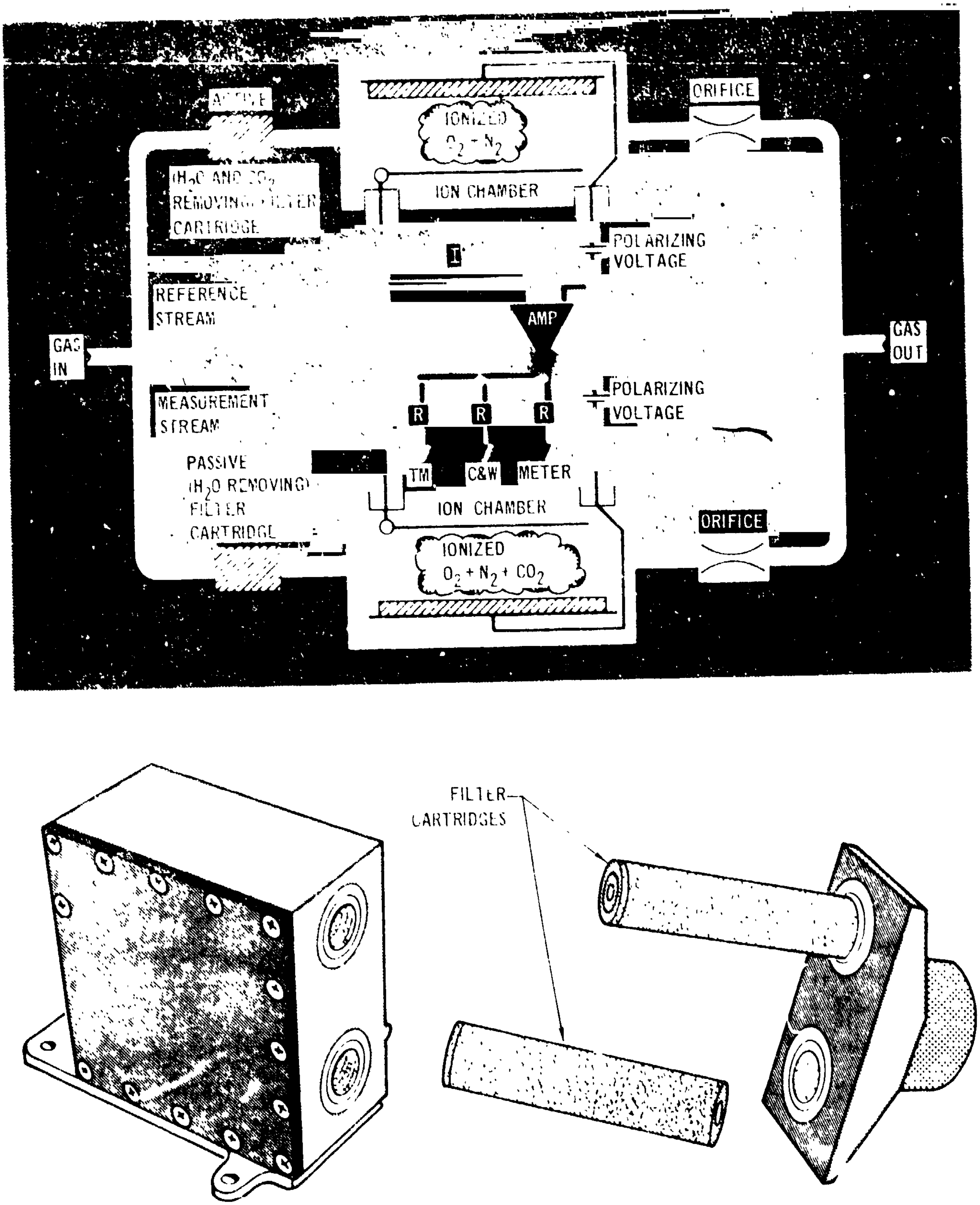
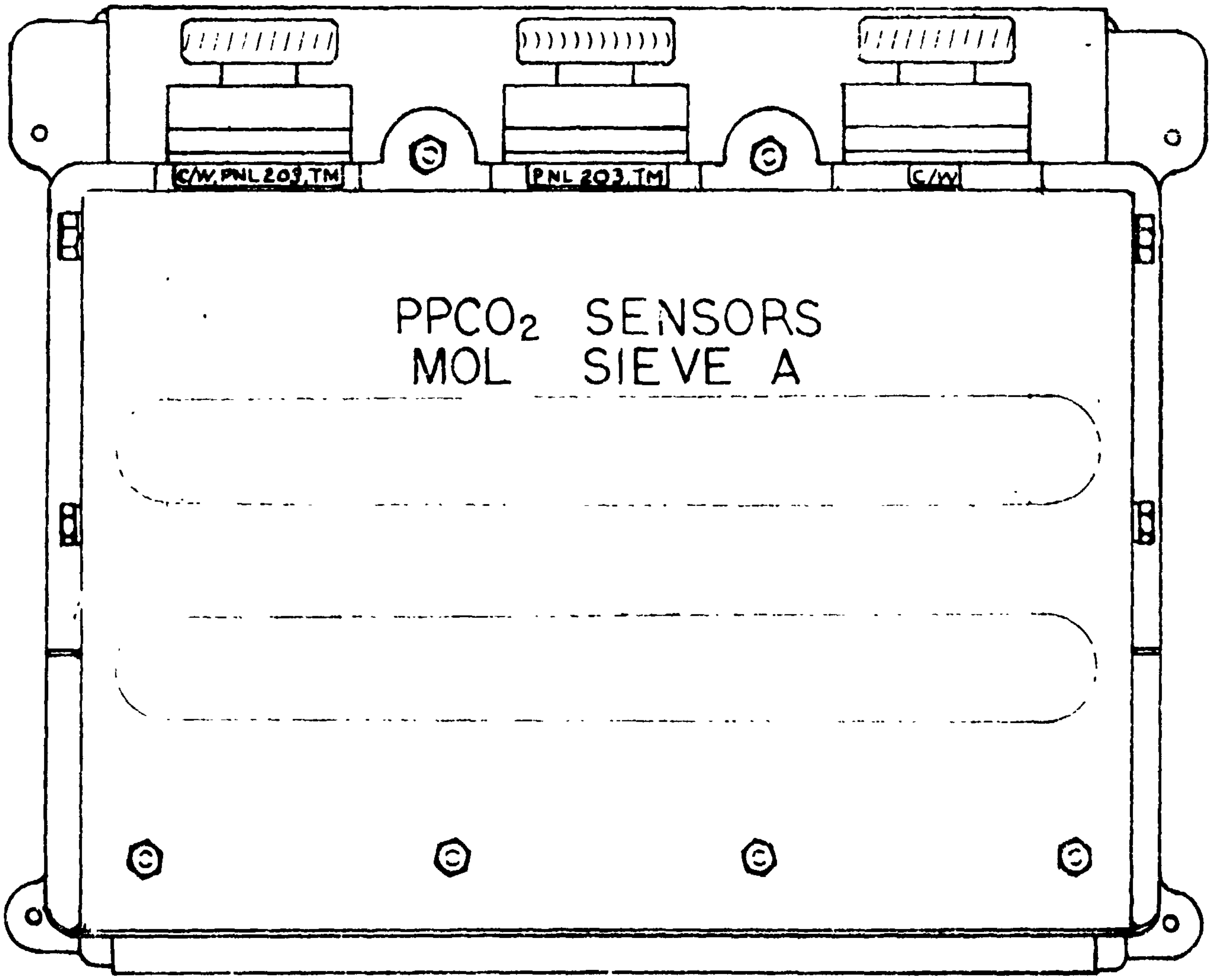


FIGURE 2.5–21 $PPCO_2$ SENSOR



(NOMENCLATURE SAME FOR MOL SIEVE B SENSOR)

////// (RED)

)))))) (BLUE)

FIGURE 2.5-22 MOLECULAR SIEVE A $PPCO_2$ SENSORS

PPCO2 SENSOR AND LOCATION	INSTALLED AT LAUNCH (2)	REQUIRED FILTER CARTRIDGE RECHARGES				SPARE RECHARGES (1)	TOTAL RECHARGES (1)
		SL-2	SL-3	SL-4	TOTAL		
MOLE SIEVE A:							
-117 INLET	-113 & -71 CARTRIDGES	1	4	4	0	1	10
-123 OUTLET	-119 & -77 CARTRIDGES	0	2	2	4	1	5
-123 OUTLET	-119 & -77 CARTRIDGES	0	2	2	4	1	5
MOLE SIEVE B:							
-117 INLET	-113 & -71 CARTRIDGES	1	4	4	9	1	10
-123 OUTLET	-101 & -103 PLUGS	0	0	0	0	1	1
-123 OUTLET	-101 & -103 PLUGS	0	0	0	0	1	1
TOTAL RECHARGES							
-117 INLET		22	8	8	18	2	20
-123 OUTLET		0	4	4	8	4	12

NOTES:

(1) A RECHARGE CONSISTS OF ONE ACTIVE AND ONE PASSIVE FILTER CARTRIDGE:

- -117 SENSOR NORMALLY USES -113 ACTIVE AND -71 PASSIVE CARTRIDGES
- -123 SENSOR NORMALLY USES -119 ACTIVE AND -77 PASSIVE CARTRIDGES

(2) FILTER CARTRIDGES FOR THE -117 AND -123 SENSORS ARE FUNCTIONALLY INTERCHANGEABLE.

FIGURE 2.5-23 $PPCO_2$ SENSOR RECHARGE REQUIREMENTS

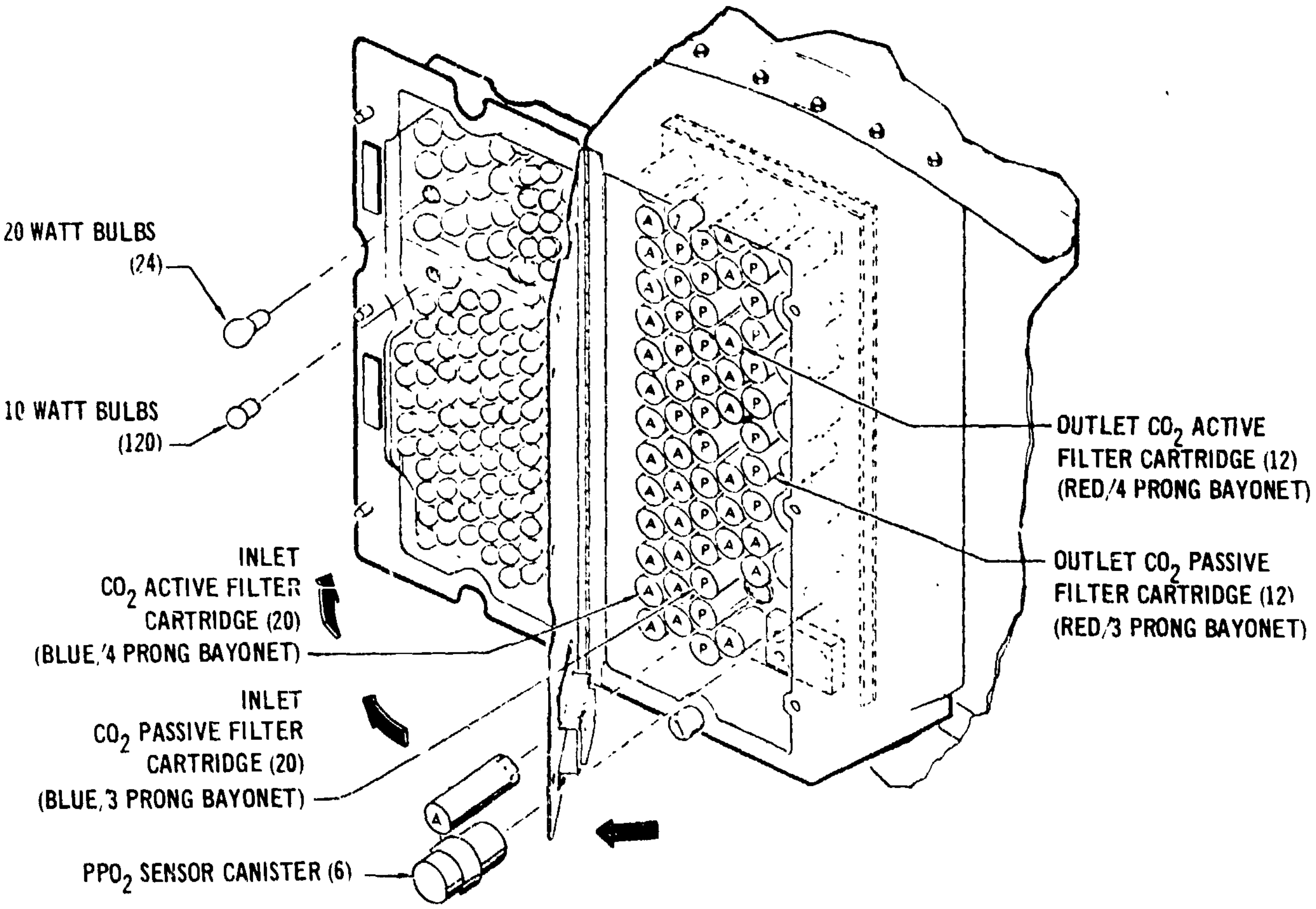


FIGURE 2.5-24 TUNNEL STOWAGE CONTAINER 301

Performance of the CO_2 and odor control system during the mission was completely satisfactory as described in Para. 2.5.4.2. Operation of Mole Sieve B was not required, since CO_2 levels were controlled within allowable limits by Mole Sieve A throughout the entire mission. No hardware failures of any type occurred.

C. Ventilation Control - Ventilation was provided to circulate and mix the atmosphere and to improve heat transfer between the gas and surfaces in the cabin (e.g., crew, lights, equipment, walls, etc.). Ventilation was provided by molecular sieve compressors, heat exchanger PLV fans, and the interchange duct PLV fan. Ventilation control was achieved by selective operation of the various fans from ECS control panel 203 and aft compartment control panel 390, adjustment of the MDA/OWS gas flow selector valve 234, and adjustment of diffusers on the MDA area fans. Preflight estimates of gas flowrates delivered to the OWS and to the MDA vs ICD requirements are summarized by Figures 2.5-25 and 2.5-26, respectively. Gas flow sensors were provided to monitor flowrates through Mole Sieve A, Mole Sieve B, the MDA/OWS interchange duct, and the OWS heat exchanger module. The sensors in the molecular sieve duct would close switches to provide a caution signal should the flowrate be less than 21.1 ± 3.8 cfm. The sensor in the interchange duct would cause a C&W alarm switch to close should the flowrate be less than 45 ± 10 cfm. Performance of the atmospheric ventilation system during flight is described in Para. 2.5.4.2.

NUMBER OF AFT COMPT FANS OPERATING	EXPECTED PERFORMANCE BASED ON U-1 PLV FAN DATA (1)			AM/OWS ICD VALVES - TOTAL FLOW TO OWS (2)
	OWS MODULE FLOW	INTERCHANGE DUCT FLOW (3)	TOTAL FLOW TO OWS	
0	0 CFM	130 CFM	130 CFM	189 LB/HR (121 CFM)
1	58 CFM	127 CFM	185 CFM	-
2	106 CFM	123 CFM	229 CFM	-
3	142 CFM	118 CFM	260 CFM	-
4	170 CFM	115 CFM	285 CFM	423 LB/HR (270 CFM)

NOTES:

(1) BASED ON PLV FAN DATA FROM MSFC (AIRESEARCH ATP DATA FOR U-1 FANS PRESENTLY INSTALLED AND CORRECTED TO ρ = .0261 LB/FT^3 & 26 VDC FAN INPUT) AND INTERFACE DUCT LOSS PER AM/OWS ICD 13M02519.

(2) BASED ON DATA ON MSFC DWG 20M42371 CORRECTED TO ρ = .0261 LB/FT^3 & 26 VDC FAN INPUT.

(3) CONTINUOUS OPERATION IN HIGH SPEED MODE.

FIGURE 2.5-25 VENTILATION FLOW RATES DELIVERED TO OWS

SOURCE	FLIGHT OPERATION - CFM	
	EXPECTED (1)	AM/MDA ICD (2)
STS MODULE		
DUCT #1	58.8	55.7
#2	59.8	55.7
#3	60.4	55.7
TOTAL	179.0	167.1
MOLE SIEVE TOTAL (3)	63.5	62 ± 10

NOTES:

(1) BASED ON PLV FAN DATA FROM MSFC (AIRESEARCH ATP DATA FOR U-1 FANS PRESENTLY INSTALLED AND CORRECTED TO ρ = .0261 LB/FT^3 AND 26 VDC FAN INPUT.) AND INTERFACE DUCT LOSS PER AM/MDA ICD 13M02521.

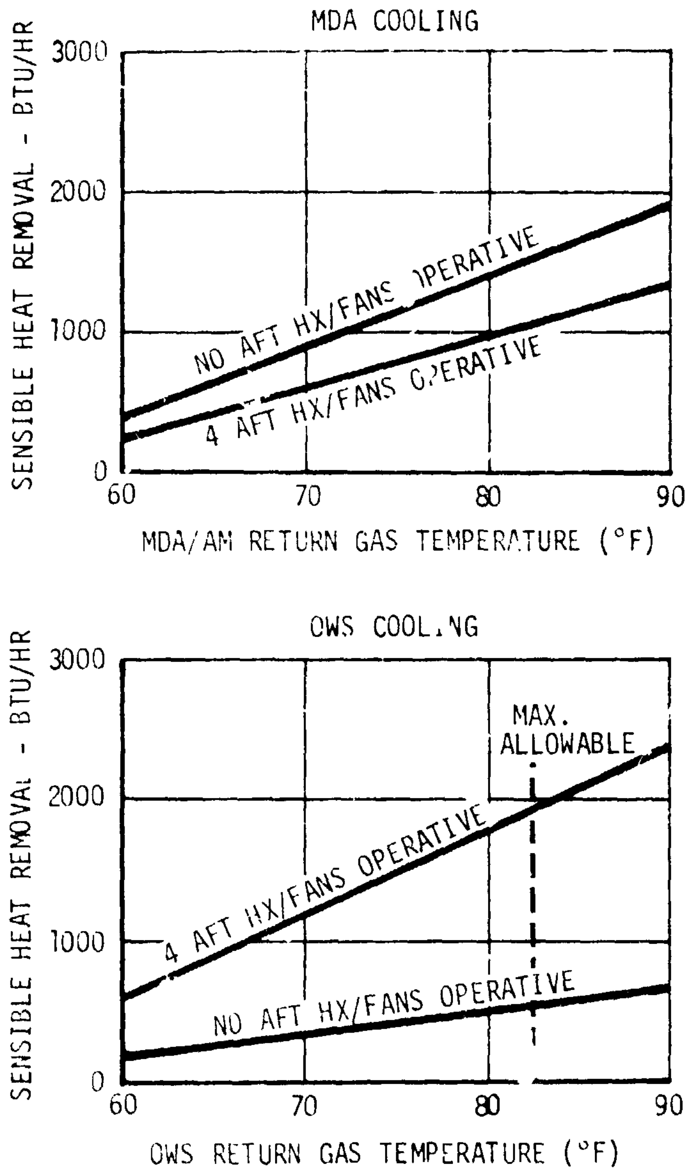
(2) BASED ON DATA ON MSFC DWG 20M42371 CORRECTED TO ρ = 0.261 LB/FT^3 AND 26 VDC FAN INPUT.

(3) BASED UPON NORMAL MOLECULAR SIEVE OPERATION:

- MOLE SIEVE SYSTEM A HAS ONE COMPRESSOR AND ONE CONDENSING HEAT EXCHANGER OPERATING WITH SELECTOR VALVES CYCLING.
- MOLE SIEVE SYSTEM B HAS ONE COMPRESSOR AND ONE CONDENSATE HEAT EXCHANGER OPERATING WITH SELECTOR VALVES BOTH FIXED IN ISOLATE POSITION.

FIGURE 2.5-26 VENTILATION FLOW RATES DELIVERED TO MDA

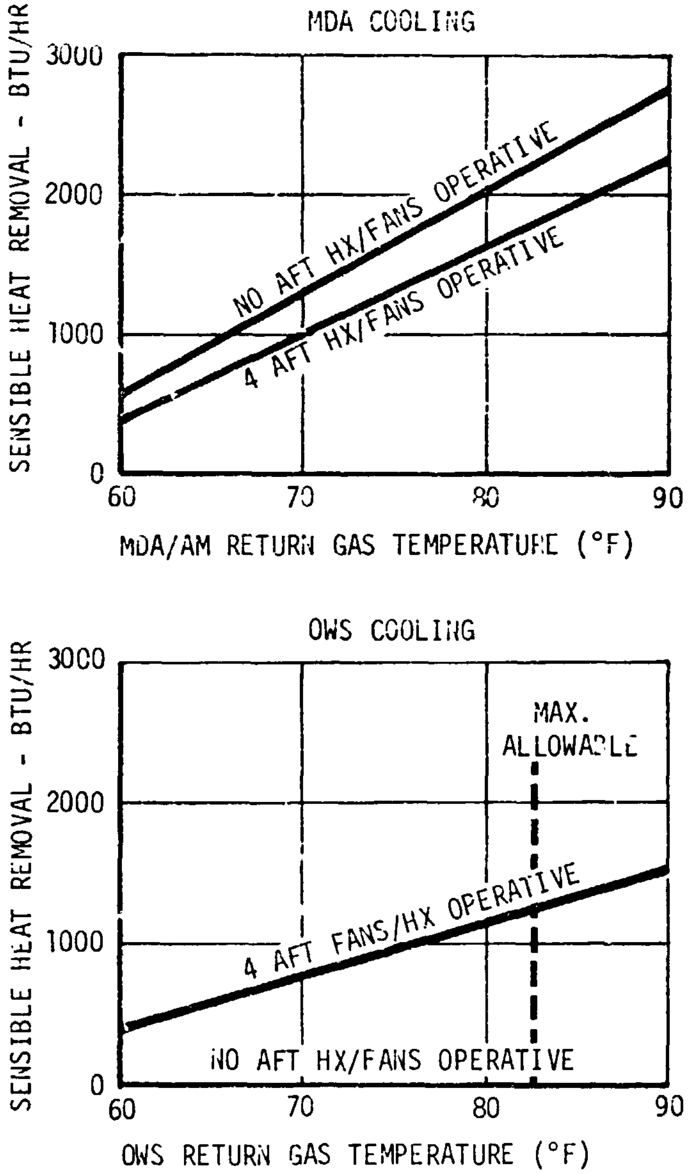
D. Temperature Control - Temperature control of the AM atmosphere and internal surfaces was achieved by a combination of cooling provided by gas circulation through condensing and cabin heat exchangers, and heating provided by internal equipment heat generation and thermostatically controlled wall heaters, as well as by passive means described in Section 2.4. With normal cluster heat loads, the AM atmospheric temperature control system was designed with the capability of maintaining gas temperature between 60°F and 90°F by controlling heat exchanger fan operation. Fan control was from ECS control panel 203, Figure 2.5-13. The atmospheric sensible heat removal capability with the condensing heat exchanger gas flow diverted to the OWS is presented in Figure 2.5-27. The atmospheric sensible heat removal capability with the condensing heat exchanger gas flow diverted to the MDA is presented in Figure 2.5-28. Temperature control of the atmosphere during the mission is described in Para. 2.5.4.2.



NOTES:

1. TWO COOLANT LOOP WITH ONE PUMP OPERATIVE IN EACH = 230 LB/HR PER LOOP
2. COOLANT TEMPERATURE ENTERING CONDENSING HEAT EXCHANGERS = 47°F
3. CLUSTER LATENT HEAT LOAD = 750 BTU/HR
4. FAN AND COMPRESSOR VOLTAGE = 26V
5. GAS FLOW RATES
   - CONDENSING HX THROUGH OPERATIVE MOLE SIEVE = 34.2 CFM (53.5 LB/HR)
   - CONDENSING HX THROUGH INOPERATIVE MOLE SIEVE = 29.3 CFM (45.9 LB/HR)
   - AFT FAN/HX = 39.5 CFM (61.8 LB/HR)
   - STS FAN/HX = 55.7 CFM (87.2 LB/HR)
   - INTERCHANGE DUCT = 112 CFM (175.2 LB/HR)
6. NET SENSIBLE HEAT REMOVAL DOES NOT INCLUDE FAN AND MOLE SIEVE LOADS
7. OWS AND MDA RETURN GAS TEMPERATURES ASSUMED EQUAL
8. AM DUCTING EMISSIVITY = 0.1

FIGURE 2.5-27 ATMOSPHERIC COOLING CAPABILITY – CONDENSING HEAT EXCHANGER FLOW DIVERTED TO OWS



NOTES:

1. TWO COOLANT LOOP WITH ONE PUMP OPERATIVE IN EACH = 230 LB/HR PER LOOP
2. COOLANT TEMPERATURE ENTERING CONDENSING HEAT EXCHANGERS = 47°F
3. CLUSTER LATENT HEAT LOAD = 7[illegible]0 BTU/HR
4. FAN AND COMPRESSOR VOLTAGE = 26V
5. GAS FLOW RATES
   CONDENSING HX THROUGH OPERATIVE MOLE SIEVE = 34.2 CFM (53.5 LB/HR)
   CONDNESING HX THROUGH INOPERATIVE MOLE SIEVE = 29.3 CFM (45.9 LB/HR)
   AFT FAN/HX = 39.5 CFM (61.8 LB/HR)
   STS FAN/HX = 55.7 CFM (87.2 LB/HR)
   INTERCHANGE DUCT = 112 CFM (175.2 LB/HR)
6. NET SENSIBLE HEAT REMOVAL DOES NOT INCLUDE FAN AND MOLE SIEVE LOADS
7. OWS AND MDA RETURN GAS TEMPERATURES ASSUMED EQUAL
8. AM DUCTING EMISSIVITY = 0.1

FIGURE 2.5-28 ATMOSPHERIC COOLING CAPABILITY – CONDENSING HEAT EXCHANGER FLOW DIVERTED TO MDA

2.5.2.3 Condensate System

The condensate system provided the capability of removing, storing, and disposing of condensate from the condensing heat exchanger water separator assemblies. The system also provided the capability of removing, storing, and disposing of gas from the EVA/IVA liquid/gas separator assembly installed in the SUS water loops. In addition, it provided support for the servicing/deservicing operations discussed in Section 2.5.2.4. The condensate system, shown schematically in Figure 2.5-29, was controlled from panel 216, shown by Figure 2.5-30, circuit breaker panel 200, mole sieve condensing heat exchanger control panels 230 and 232, lock compartment control panel 316, HX plate servicing panel 303, condensate dump panel 393, and OWS control panels.

The system employed redundant check valves in the transfer line, overboard dump line exits, solenoid valves, and electrical heaters. Freezing of water within the transfer and overboard dump lines was prevented by gas purge and by wrapping spacecraft coolant lines and dump lines together with aluminum foil tape and insulating the bundle. The condensate tank module was in-flight replaceable and a spare condensate module was provided for module replacement, if required.

A. Removal - Adequate removal was provided by maintaining the pressure level between 0.5 and 6.2 psi below cabin pressure but above the triple point pressure of water. This range is compatible with removal of water from the water separator plate assemblies on one condensing heat exchanger at rates exceeding the normal production rate, with operation of the EVA/IVA water separator, and with servicing/deservicing support. Pressures within this range were provided by the stowage and dump functions.

B. Stowage - During all manned operations except EVA stowage was provided in the OWS by a holding tank containing 2 bellows with 10 ft^3 of water collection volume. The design stowage duration for this primary stowage tank was 30 days. For stowage during EVA and backup operation, a 0.256 ft^3 condensate tank module was provided in the STS. The design stowage duration for this alternate tank module was 18 hours. These times are based upon the performance characteristics shown in Figures 2.5-31 and 2.5-32 for the primary and alternate tanks, respectively, specification leakage rates, the maximum water generation rate, and a final tank pressure of 4.3 psia.

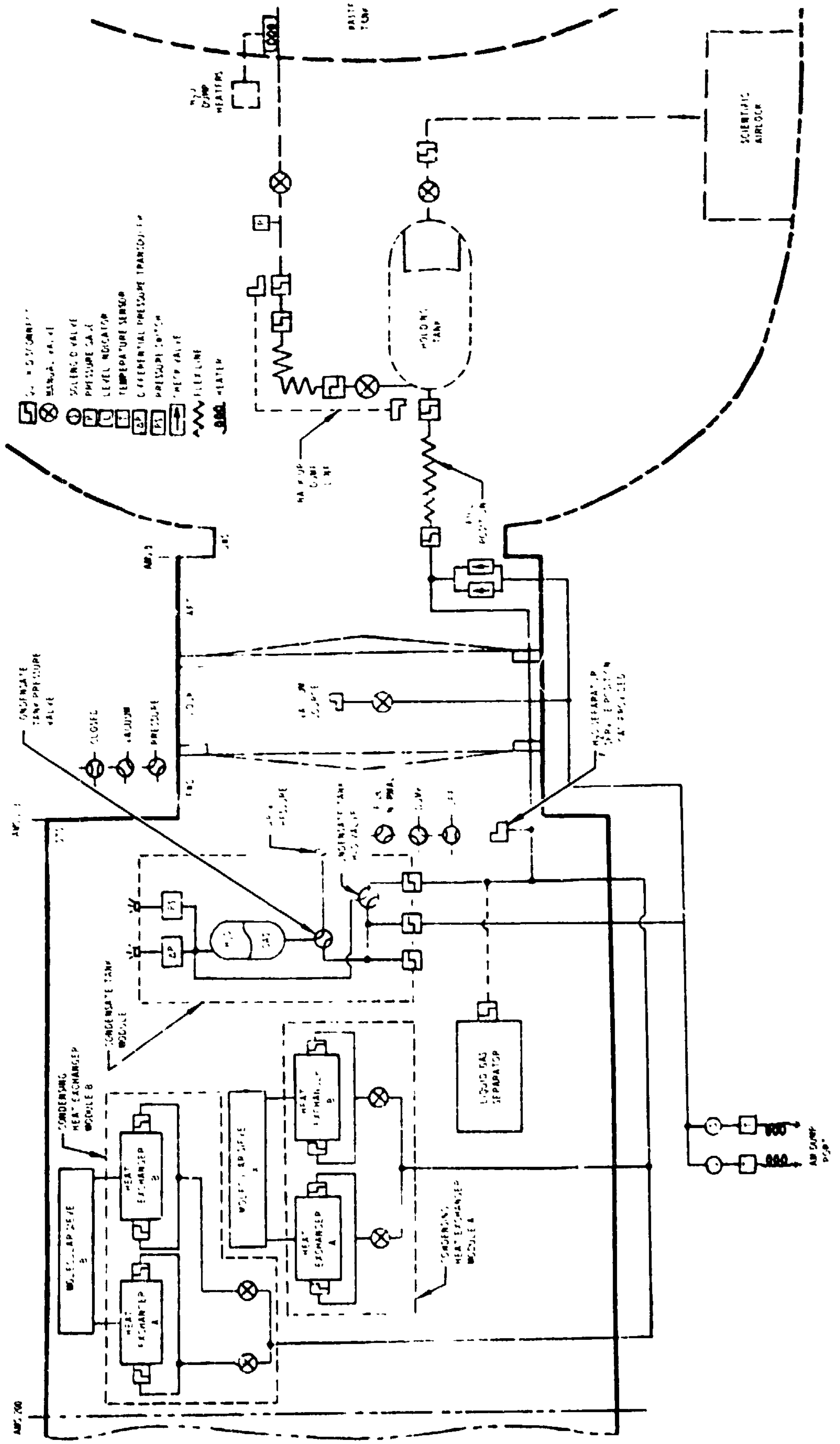


FIGURE 2.5–29 AM CONDENSATE SYSTEM

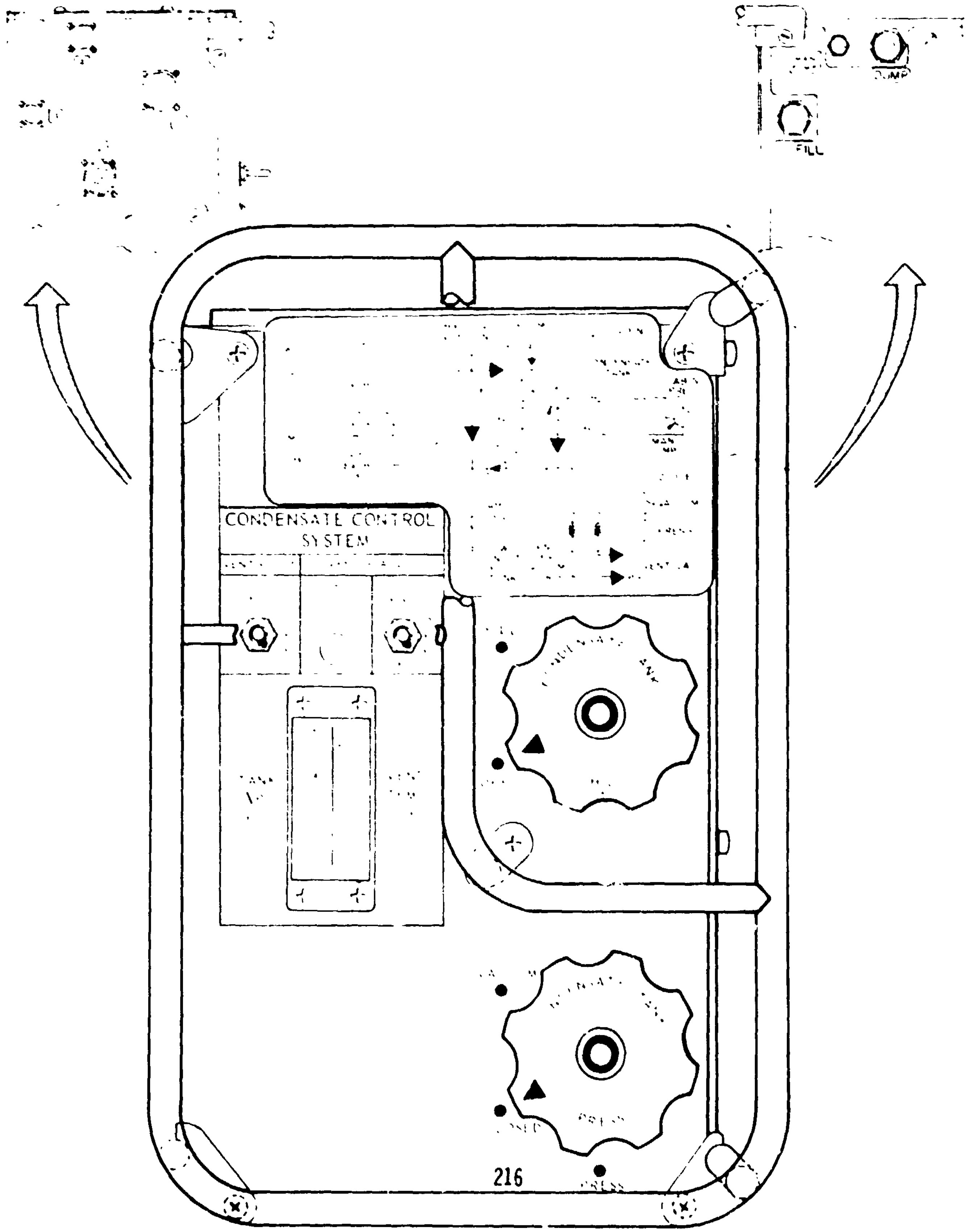


FIGURE 2.5-30 CONDENSATE CONTROL PANEL 216

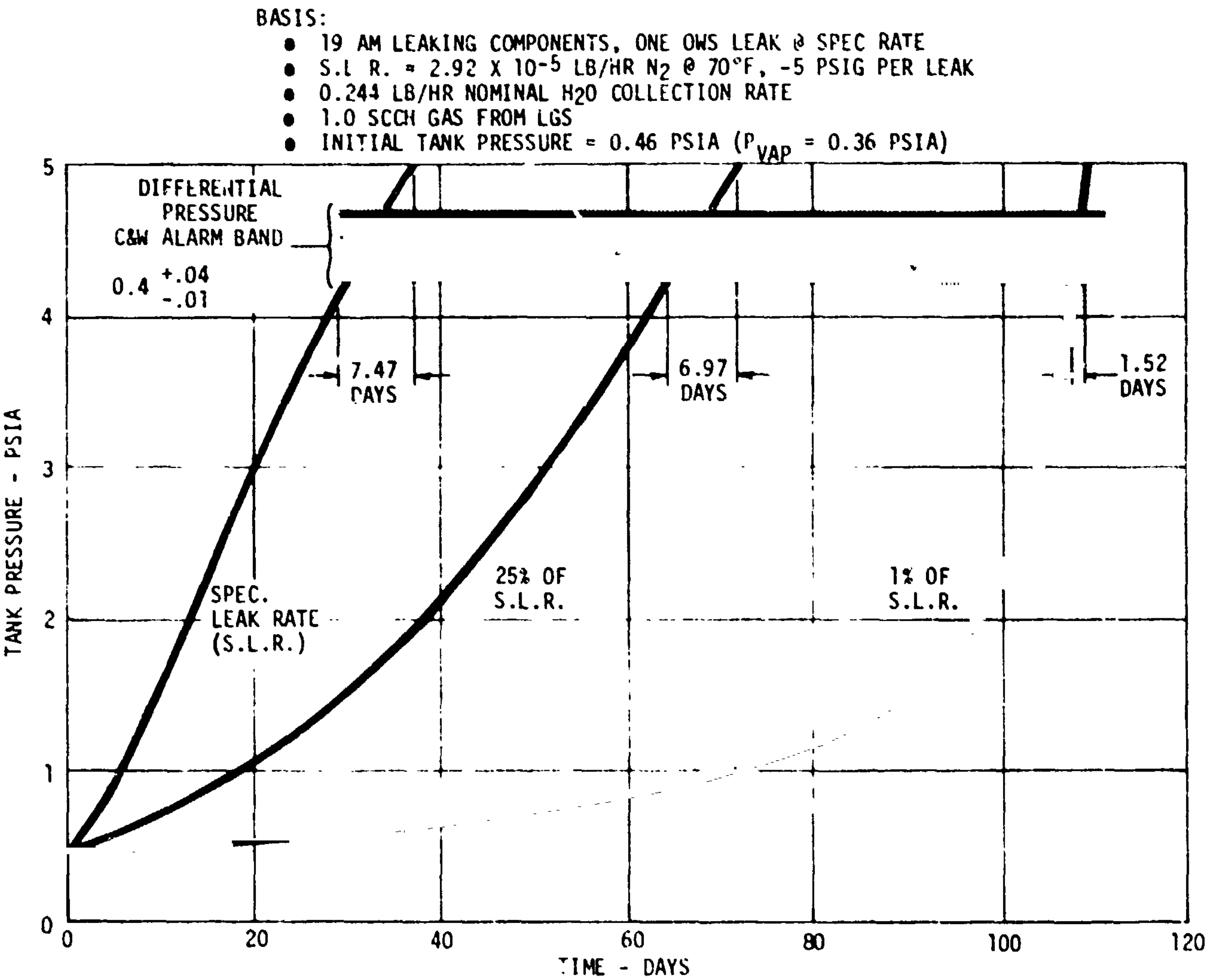


FIGURE 2.5-31 EFFECT OF CABIN GAS LEAKAGE ON OWS HOLDING TANK PRESSURIZATION

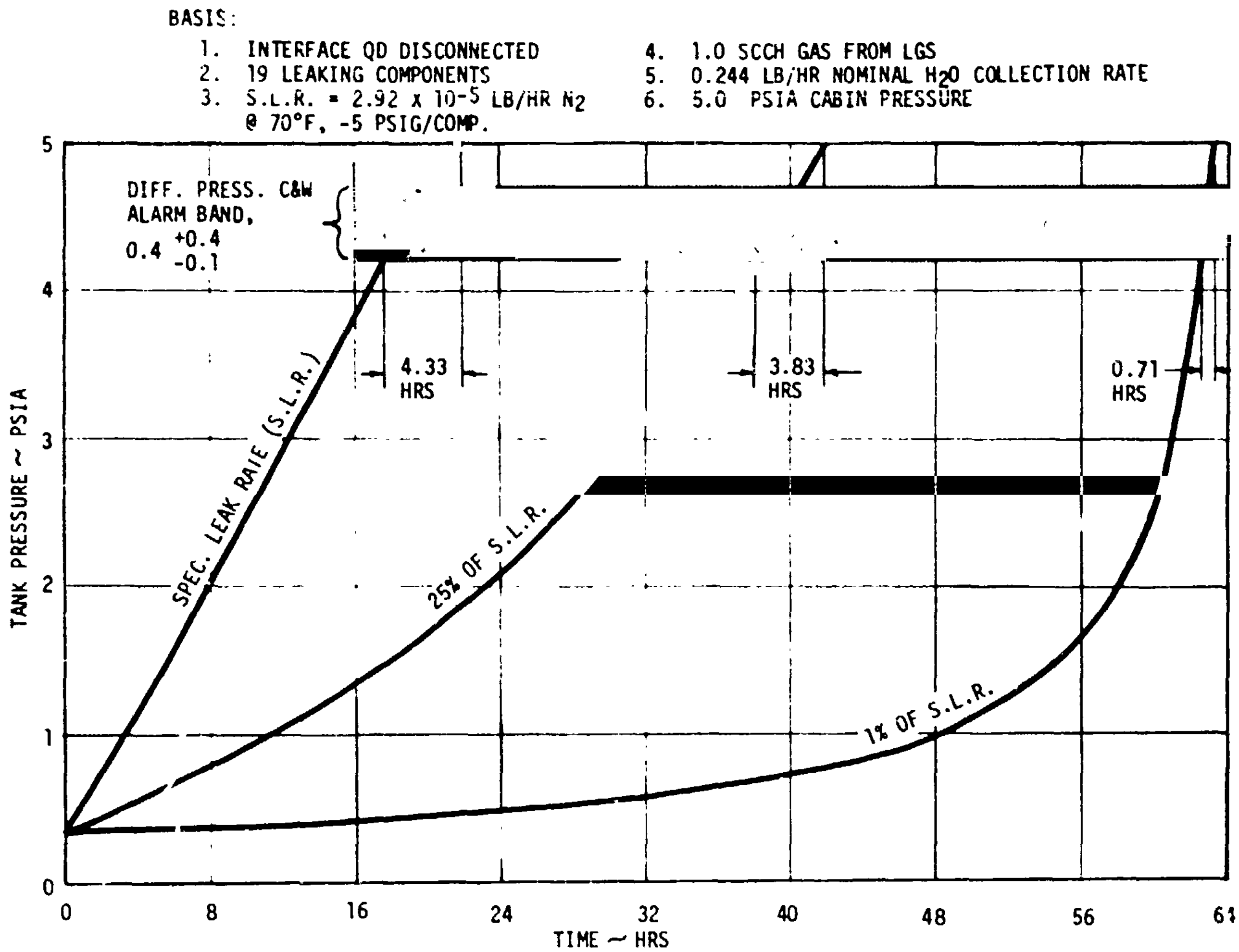


FIGURE 2.5-32 AM CONDENSATE TANK PRESSURE BUILDUP

C. Disposal - Prime dumping was to the OWS waste tank. Normally, only the OWS holding tank was dumped to the waste tank, but the line from the AM interface could be connected directly to the OWS waste tank for dumping the alternate tank.

Dumping of the OWS holding tank was accomplished by actuating the OWS probe heater, disconnecting the condensate inlet QD from the holding tank, venting the gas side of the holding tank to the cabin, and opening the condensate outlet shutoff valve. When the position of the OWS tank bellows and H_2O dump pressure indicated the water was dumped, the valve in the dump line was closed, the condensate inlet QD's were reconnected, the gas side of the OWS tank was vented to vacuum, and the heater was deactivated.

Alternate dumping was directly overboard through the AM overboard dump port. The system was operated from condensate control panel 216 (Figure 2.5-30). Prior to AM collection tank overboard dump operation, the AM exit line temperature was monitored via an on-board display. Dumping of the AM collection tank through the AM overboard dump port was accomplished by activating the AM exit heater as required, positioning the manual condensate tank pressure valve in the "press" position so that cabin pressure was applied to the gas side of the bladder, placing the manual condensate tank H_2O valve in condensate "dump" position, and opening the solenoid valve in the dump line. When the position of the condensate tank bladder indicated the water was dumped, the condensate tank pressure valve was positioned to "vacuum" to evacuate the gas side of the bladder and purge water from the lines. Purge was completed when tank ΔP gage was 3.8 psi or greater. At completion of purge, the condensate tank pressure valve and exit solenoid valves were closed, the H_2O valve was placed in "fill" position, and the exit heater was deactivated.

Two solenoid valves were provided at the exit for redundancy. The condensate overboard vents were located at AMS 157.25, with the primary vent 28° 37' off the +Z axis toward the -Y axis and the secondary vent 49° 7' off the +Z axis toward the -Y axis.

D. Servicing/Deservicing Support - The servicing/deservicing support provided by the condensate system consisted of providing 13 lbs of condensing heat exchanger separator assembly wetting solution stored within the spare condensate tank module, and a low pressure sink for servicing/deservicing of equipment discussed in Section 2.5.2.4. The water solution contained 10% Roccal (biocide), and 1% Sterox NJ (wetting agent) by volume prepared in accordance with P.S. 20531. The spare module was launched in the STS on cabinet 168. For plate servicing, it was strapped down adjacent to the H_2O separator service position QD in the STS. After servicing the necessary plates, the spare module was stored in the OWS on the refrigeration pump package. Access to the low pressure sink was through fittings on panels 316 and 303.

2.5.2.4 In-flight Water Servicing/Deservicing

Provisions were made for in-flight servicing of water separator plate assemblies, and servicing/deservicing equipment as well as servicing support for ATM C&D Panel/EREP and EVA/IVA water systems. In addition, provisions were made for in-flight deservicing of the servicing/deservicing equipment and deservicing support for LSU/PCU.

A. Water Separator Assemblies - The water separator assemblies were serviced in conjunction with the spare condensate tank module, hose, adapter, and servicing disconnect on panel 303. The tank module was to be strapped down adjacent to the H_2O separator service position QD in the STS and connected sequentially to each separator plate by an adapter as shown in Figure 2.5-33. Valving on the spare module was used to force

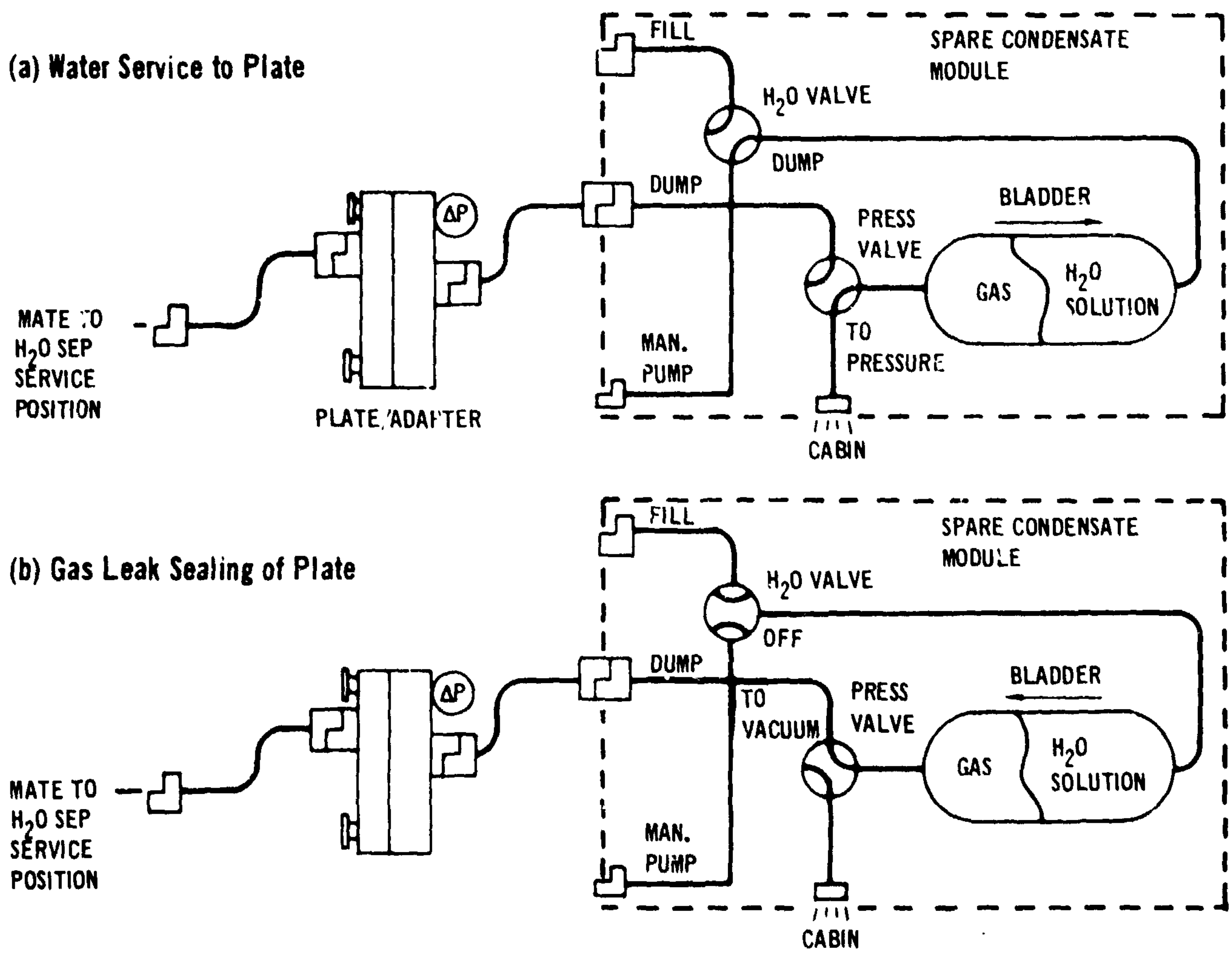


FIGURE 2.5-33 WATER SEPARATOR PLATE SERVICING

water from the tank through the plate. Water flow time was a function of condensate collection ΔP and varied from 12 seconds at a ΔP of 5 psid to 25 seconds at a ΔP of 1 psid. After water flow was established, the valves were switched to the position for gas leak seal. After 2 minutes at these positions, the PRESS valve was positioned to OFF and the pressure gage monitored. A plate was considered leak sealed if the gage reading did not drop more than .25 psi in 2 minutes. The serviced plate was then attached to the heat exchanger. After servicing the necessary plates, the spare module was stored in the OWS.

To preserve fluid and the normal service capability after an in-flight condensate module replacement the jumper hose (Figure 2.5-34) permitted transfer of service fluid between condensate modules. When water solution was unavailable from either module, the manual pump could be used as a pressurant source to service with condensate from the installed module. The QD nipple on the plate would exhaust to cabin for this servicing mode. In addition, the manual pump could have been used with the spare module for a fast service of a single plate in the OWS or STS. During flight, servicing was performed normally so these alternates were not required.

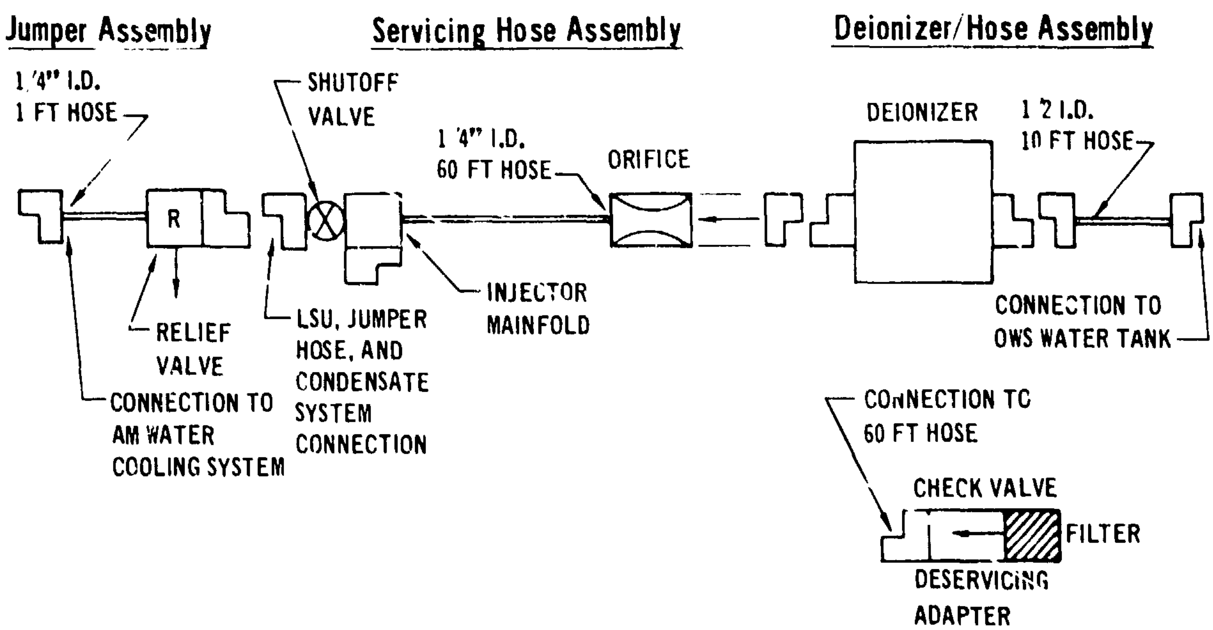


FIGURE 2.5-34 INFLIGHT WATER SERVICINC

B. Other Equipment - The servicing/deservicing equipment illustrated in Figure 2.5-34, ATM C&D Panel/EREP System, and EVA/IVA Water System were serviced using the OWS water system, the assemblies shown in Figure 2.5-34 and the condensate system. The water allocated for in-flight servicing was 126 lbs contained in OWS tank no. 9. The allocation included: (1) filling 6 umbilicals - 36 lbs, (2) servicing suit cooling system twice - 24 lbs, (3) servicing the ATM C&D panel/EREP cooling system twice - 24 lbs, (4) 50% additional amount for contingencies - 42 lbs. The water in the OWS water tank contained Iodine and Potassium Iodide which were removed by the deionizer during system servicing. The water was pressurized to 35 $\pm$ 2 psig by a metal bellows and gas pressure. The water in the OWS tank was deaerated to contain a maximum of 6.9 ppm dissolved air, so excessive amounts of dissolved gases, which could degrade pump life, would not be introduced into the water loops upon servicing. An orifice in the servicing hose assembly limited flow to a maximum of 2.5 lb/min. A relief valve in the jumper hose assembly limited maximum pressure to 15 psig, thus protecting reservoirs in water loops from overpressurization.

The deionizer/hose assembly was launched serviced while the servicing hose and jumper hose assemblies were launched dry. The servicing equipment was stored in the OWS when not in use. The servicing/deservicing equipment and LSU/PCU were deserviced using the assemblies shown in Figure 2.5-34 and the condensate system.

2.5.3 Testing

Prelaunch tests were performed to provide information needed by engineering for design, to qualify a particular part numbered component, to verify that the particular part and serial numbered components operated properly, to verify that U-1 and U-2 modules and systems functioned properly, and to support verification that the vehicle was ready for flight. Post launch tests were conducted to provide information needed for real time mission planning. Information on the test philosophy is presented in Section 5 of this report.

2.5.3.1 Development Tests

Development tests were performed on components and systems to obtain data on the functional characteristics needed to support the design process. Test requirements were specified by Test Request (TR).

A. Performance Tests - Performance tests were conducted to establish the performance of new components and systems. Some were conducted by vendors to satisfy requirements identified in Specification Control Drawings (SCD). Those tests conducted by MDAC-E are summarized below.

- Title - PLV Fan and Cabin Heat Exchanger Flow Test

  Background - The PLV Fan had never been used in conjunction with a heat exchanger.

  Objective - Obtain flow rate versus delta pressure data of the PLV fan and the PLV fan in combination with the cabin heat exchanger for conditions that would be seen on Airlock (5 psia).

  Results - Flow rate versus delta P data was obtained for the PLV fan and the PLV fan in combination with the cabin heat exchanger, Reference TR 061-068.29.

- Title - Condensing Heat Exchanger Gas "Breakthrough" Point Test

  Background - The condensing heat exchanger was used on Gemini in the closed loop circuit, where it had continuously high humidity inlet gas. On Airlock it would be used in [illegible] n open loop system and at times would have [illegible] mid- ity and possibly higher temperature inlet gas. [illegible] was no data at these conditions.

Objective - Determine as a function of atmospheric temperature and humidity, the point at which gas would flow through the initially wet water separator plates of the condensing heat exchanger.

Results - Test results indicate that the breakthrough point occurred in successively shorter times as the gas inlet dew point is reduced from the coolant inlet temperature (40°F for this test). There was an apparent minimum time required for breadthrough at approximately 5.5 hours under a low humidity condition. The results indicated that a dew point above the coolant inlet temperature would be required to condense sufficient water to prevent breakthrough, Reference TR 061-068.31.

- <u>Title - Condensing Heat Exchanger Thermal Performance Test</u>

Background - The condensing heat exchanger used on Airlock was the same as the suit heat exchanger used on Gemini. Performance characteristics for this unit were known, but at different flowrate and temperature combinations than would be seen on Airlock.

Objective - Verify flowrate and heat transfer performance at coolant/gas flowrate, temperature, and humidity combinations that were expected for Airlock.

Results - Two separate heat exchanger insulation/outlet duct configurations were tested. In addition, gas flow - differential pressure characteristics of the compressor were obtained to describe various gas flow conditions. Data required for the thermodynamic evaluation of the condensing heat exchanger was obtained from this test, Reference TR 061-068.34.

- <u>Title - Cabin Heat Exchanger/PLV Fan Subsystem Development Test</u>

Background - Performance data was needed to define the operation of the cabin heat exchanger assembly and heat exchanger/fan combination, planned to be used for the Airlock Project.

Objective - Determine the characteristics of the cabin heat exchanger/PLV fan assembly.

Results - Performance characteristics were obtained. All test results were satisfactory, Reference TR 061-068.36.

- Title - Water Servicing Development Test for Condensing Heat Exchanger (Airlock P/N 52-83700-1193)

Background - The plates which separate water from the gas in the condensing heat exchanger must be completely saturated with water to prevent gas leakage.

Objective - Develop a procedure for inflight water servicing of the separator plates.

Results - A procedure to service the separator plates was developed, Reference TR 061-068.37.

- Title - O_2/N_2 Two-Gas Control System Development Test

Background - The AM two-gas control system was required to maintain atmospheric total pressure and O_2/N_2 composition within desired limits.

Objective - Verify the functional adequacy of the two-gas control system at normal and extreme environmental conditions encountered during flight.

Results - The system controlled the partial pressure of oxygen within specification limits, Reference 061-068.39.

- Title - Water Separator Plate Assembly Performance Tests

Background - The initial design did not facilitate easy inflight replacement of the separator plate assemblies.

Objective - Evaluate a quick-change water separator plate assembly fastener design.

Results - This test demonstrated that the proposed fastener design for inflight replacement would perform adequately, Reference TR 061-068.59.

- Title - Cabin Pressure Relief Valve (52-83700-1213) Freezing Development Test

Background - The relief valve in the lock compartment had to vent to prevent the pressure from exceeding 6.0 psia when the astronauts were suited and preparing for EVA.

Objective - Determine if the poppet valve in the cabin pressure relief valve would be held open due to an ice buildup.

Results - Test results showed that no condensing or freezing could occur under the test conditions, Reference TR 061-068 65.

- Title - O_2 and N_2 Regulator Performance Test at Low Temperature

Background - There was concern whether the regulators would perform at low inlet gas temperatures.

Objective - Determine the effects of low temperature inlet gas on performance.

Results - The regulators did not function satisfactorily. Vent lines installed on the regulator sense and relief valve ports proved an effective means of preventing frost formation in sensing ports, and O-rings were changed from Viton to fluorosilicon. Reference TR 061-068.88 and TR 061-068.88.01.

- Title - Water Servicing and Gas Breakthrough for Condensing Heat Exchanger

Background - The heat exchanger configuration was changed to allow for inflight replacement of the separator plates.

Objective - Determine if the latest configuration of water separator plates in the condensing heat exchanger was able to be serviced (wetted) by moist gas.

Results - The water separator plates were serviced satisfactorily for six of the eight test conditions. Reference TR [illegible]-068.89.

- Title - Water Servicing Technique for Plate Wetting to Alleviate Breakthrough Problem for Condensing Heat Exchanger

Background - A period of over 10 hours was required for the heat exchanger to self-wet at a 50°F dewpoint and 3-1/2 hours at a 58°F dewpoint.

Objective - Demonstrate the feasibility of positive plate wetting.

Results - The positive plate wetting technique worked satisfactorily for all conditions tested, Reference TR 061-068.89.01.

- Title - Evaluation of Water Servicing Technique for Plate Wetting Using Squeeze Bulb and Spare Condensate Tank

Background - Positive wetting was achieved by water flow through the plates followed by leak check.

Objective - Demonstrate the feasibility of positive plate wetting using the squeeze bulb to provide flow through the plates from the spare condensate tank.

Results - The above plate servicing method was demonstrated satisfactorily, Reference TR 061-068.89.03.

• Title - Effect of Rapid Depressurization on Moist Cabin Heat Exchanger

Background - Condensate formed in the gas circuit might effect the heat exchanger when it is exposed to vacuum pressure.

Objective - Expose the heat exchanger to the vacuum environment expected when the aft compartment was depressurized for EVA.

Results - The vacuum exposure test verified that condensate formed in the gas circuit did not damage or adversely affect the heat exchanger, Reference TR 061-068.97.

• Title - Condensate System Performance Test Using Exit Port Nozzles

Background - New exit nozzles had been incorporated to eliminate the formation of ice cones.

Objective - Verify that the condensate system, modified by the new nozzles was capable of meeting all system requirements

Results - The system operated satisfactorily, Reference TR 061-168.04.

• Title - Condensing Heat Exchanger Vacuum Exposure Development Test

Background - The condensing heat exchanger might be exposed to a vacuum environment.

Objective - Verify the ability of an operating, wetted condensing heat exchanger to withstand exposure to a hard vacuum.

Results - The heat exchanger operated satisfactorily during the test, Reference TR 061-168.05.

• Title - Water Separator Plate Servicing and Operational Test on Airlock Condensing Heat Exchanger.

Background - Inflight use of the condensing heat exchanger required examination for breakthrough after installing wetted separator plates.

Objective - Determine effectiveness of service and leach rate of Sterox and Roccal from serviced plates after installation.

Results - It was concluded that the water separator plates were satisfactorily serviced using the inflight servicing procedures and hardware, Reference TR 061-168.08.

- Title - 52-88715-43 CO_2 Detector Verification Development Test Using TR 061-168.18 Test Setup.

  Background - During the manned altitude tests of U-1 both detectors used to sense $PPCO_2$ at the mole sieve inlet gave erroneous readings.

  Objective - Verify proper operation of CO_2 detectors with sense line located upstream of the condensing heat exchanger.

  Results - A change in sensing line location from downstream to upstream of condensing heat exchanger did not work; reduction of flow solved the problem, Reference TR 061-168.09.

- Title - Operation of Condensing Heat Exchanger Water Separator Plates with Near-Vacuum Downstream Pressure

  Background - Condensate transfer had been changed from the condensate tank to the OWS holding tank.

  Objective - Verify that the plates would operate satisfactorily with a downstream pressure of 0.1 psia for 14 days.

  Results - The heat exchanger operated satisfactorily throughout the test, Reference TR 061-168.13.

B. Endurance Test - An endurance test, designated ET-1 and documented by report TR 061-068.35, was conducted to verify that system components had the endurance to function properly during a complete mission. The test hardware included more than 70 Airlock flight configuration components assembled into functional systems. The test was designed to load the components and make them perform under conditions expected during flight. The test followed the proposed Skylab mission plan which consisted of 3 Active Phases and 2 Orbital Storage Phases covering a real time period of 8 months.

All components initially assembled into the ET-1 Environmental Control System functioned adequately except three; namely, the 120 psig O_2 regulator, a humidity sensor, and the cabin pressure switch. The 120 psi regulator exhibited excessive internal leakage found to be caused by teflon particles generated by out of specification finish on valve stem.

Valve stems on all valves were inspected and reworked as required. Both the humidity sensor and the cabin pressure switch were rejected for being out of tolerance. Since these failures could not be duplicated,further use of these particular components were sepcially controlled. After replacement of these components early in the test sequence, testing was continued and concluded satisfactorily.

2.5.3.2 Qualification Tests

Qualification tests and documentation are available for all Airlock components and systems. Test results are summarized in MDC Report G499, Volume V.

2.5.3.3 Acceptance Tests

Acceptance tests were conducted to prove the delivered components and systems function properly.

A. Acceptance Test - An acceptance test had to be passed at the vendors plant before shipment to MDAC-E. Acceptance test requirements were specified in the Acceptance Test Procedures. These procedures were prepared by the vendor and approved by MDAC-E.

B. Pre-Installation Acceptance Test - A pre-installation acceptance (PIA) test had to be passed at the MDAC-E plant to prove that the hardware arrived in good condition prior to going into the crib which supplied parts for U-1, U-2, and spares. PIA test requirements were defined by MDAC-E Service Engineering Department Report (SEDR).

2.5.3.4 System Tests

System tests were conducted to verify that modules and systems operated properly. System test requirements were specified by SEDR.

A. Major Subassemblies - Major subassemblies were tested prior to installation during vehicle buildup. A tabulation of subassemblies tested prior to installation is shown below.

SUBASSEMBLY TESTS

D3-G51	Misc. Fluid Sys. Funct. Tests	Oxygen, Nitrogen, Coolant
D3-M51	Misc. AM Fluid Sys. Mfg. Tests	Miscellaneous
D3-G52	Molecular Sieve Functional Tests	Ventilation
D3-G62	O_2 Supply Subassembly Functional Test	Oxygen
D3-G63	O_2/N_2 Control Subassy., Test	Oxygen, Nitrogen
D3-G64	H_2O Condensate Module	Condensate
D3-G65	N_2 Supply Subassy. Tests	Nitrogen
D3-G67	N_2 Supply System Test	Nitrogen
D3-G68	Condensing Heat Exchanger Module	Coolant, Ventilation

B. Systems - Final system and integrated acceptance test flow at the contractor facility are depicted in Figure 2.5-35 through 2.5-37 for the Atmospheric Control, Gas, and Condensate System. Problems encountered during these tests are presented below:

(1) Atmospheric Control System - The first system level test, SEDR D3-N70, was performed to validate all components and modules at the system level, i.e., fan operation, mole sieve cycle tests and leak checks. These tests included verification of all instrumentation and caution and warning parameters.

- The mole sieve gas selector valves leaked excessively during SEDR D3-N70 because of improper positioning of the pointer on the selector valve. The pointer was repositioned by MRR A12AM3 and retested by MPS AVE 69; however, the retest was unacceptable due to excessive leakage. The mole sieve was replaced per MRR A12AM3.
- SEDR D3-E72 system validation test revalidated the ventilation system and mole sieves. The parameters that supply inputs to the caution and warning system were verified including flow sensors and carbon dioxide sensors.
- In an effort to improve the accuracy of the gas flow measuring system, the transfer duct flow sensor was relocated to an area of less turbulence and shielding on all sensor wiring was modified (Ref MRR A12AF9 and EJS 061-1774). Revalidation of the gas flow measuring system was performed by SEDR D3-E76.

(2) Gas System - The first test was a proof and leak check of the high pressure portion of the system per SEDR D3-N67.

- Some of the manual valves in the O_2 and N_2 gas system were replaced with updated units. High pressure lines within the O_2/N_2 module were replaced when it was determined the lines might have been fabricated from defective material. Retest of the O_2/N_2 module after rework was accomplished by MPS AVE 50. Testing after rework was performed during SEDR D3-N70.

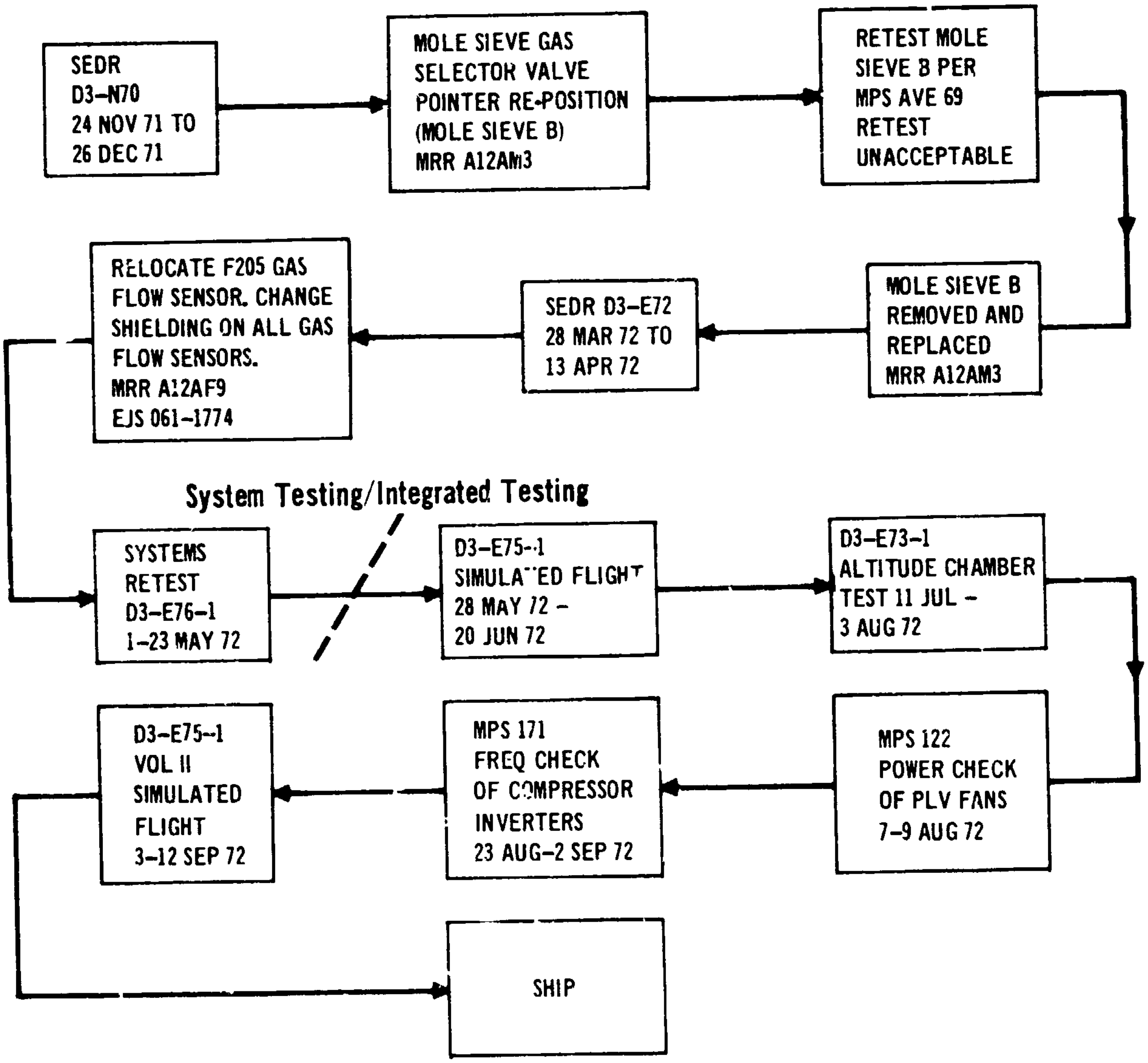


FIGURE 2.5-35 ATMOSPHERE CONTROL SYSTEM TEST HISTORY - MDAC-E

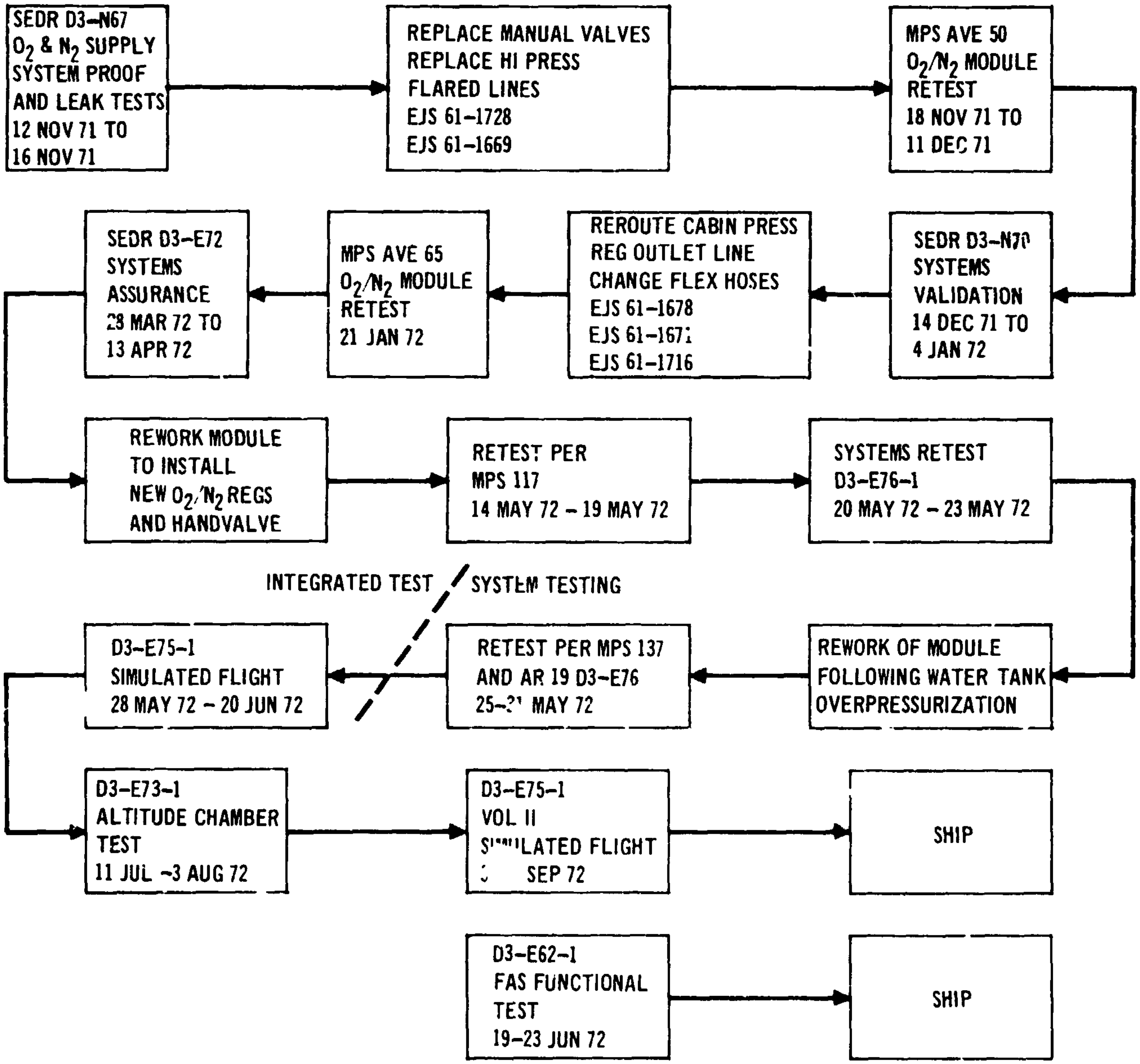


FIGURE 2.5-36 GAS SYSTEM TEST HISTORY - MDAC-E

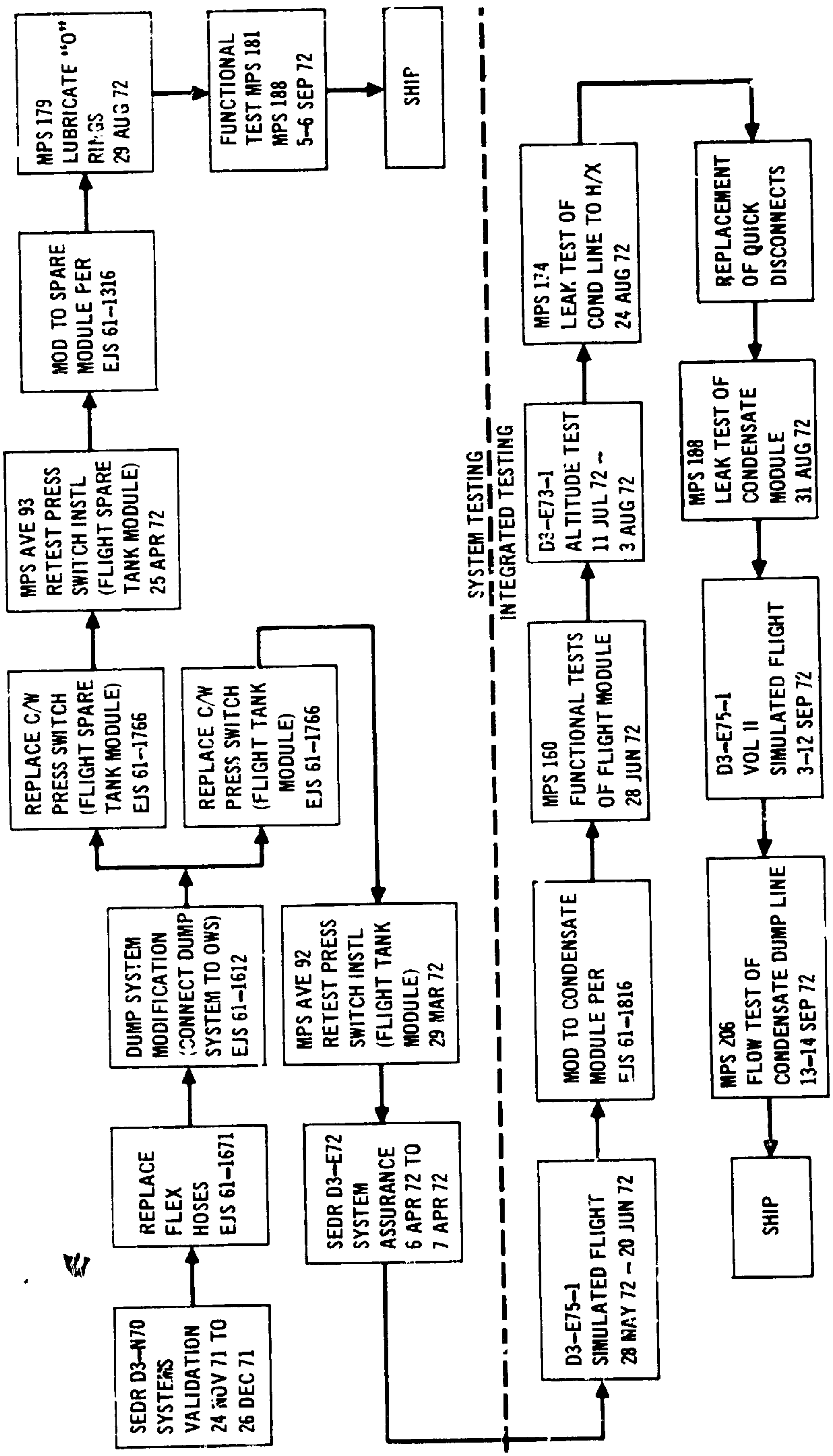


FIGURE 2.5-37 CONDENSATE SYSTEM TEST HISTORY – MDAC-E

- SEDR D3-N70 was the first system level test. During this test, the O_2 and N_2 gas system was leak checked, components were functionally checked, and system flow rates verified. System instrumentation and caution and warning parameters were validated along with verification of DCS controlled functions.
- During SEDR D3-N70 it was found that inadequate flow occurred from the cabin O_2/N_2 fill line. The O_2/N_2 module and fill line were both modified to correct the problem. The O_2/N_2 module was removed from the vehicle for the rework and retested per MPS AVE 65, prior to reinstallation. Retest of the fill line was deferred to SEDR D3-E72. Various flexible hoses in the system were updated to the latest configuration and some hardlines were insulated during the time of O_2/N_2 module modification.
- The O_2/N_2 system was revalidated during SEDR D3-E72. A functional check of the EVA/IVA O_2 system using a suited crew man at sea level ambient conditions was also performed. Following SEDR D3-E72-1, the O_2/N_2 module was removed from the vehicle and the high pressure O_2 and N_2 regulators were replaced because of a configuration change. During this time the shutoff valve for the OWS N_2 supply was replaced with an updated unit. Retest of the reworked module was accomplished by MPS AVE 117 prior to reinstallation in the spacecraft.
- The O_2/N_2 module was reinstalled in the vehicle and system retest was accomplished by SEDR D3-E76. During retest the water tank N_2 pressurization system was inadvertently over-pressurized. The O_2/N_2 module was removed from the vehicle and some components were replaced. Retest of the module after component replacement but prior to reinstallation was accomplished by MPS AVE 137.
- The O_2/N_2 module was reinstalled in the spacecraft and the O_2 and N_2 gas system retested by AR 19 versus SEDR D3-E76.

(3) Condensate System - The initial system level test was performed by SEDR D3-N70. This test included system leak and functional checks at operating pressures and a dump cycle test using N_2. Instrumentation and caution and warning parameters were verified. Several modifications were performed to improve the system. These included: (1) replacement of flexible hoses to provide a more reliable hose where the hose is used with a quick disconnect, (2) connecting the dump system to the OWS waste tank to eliminate ice particle generation outside the vehicle as a result of an overboard dump, (3) replacement of the condensate tank module pressure switch because of a dielectric failure following vibration testing of the pressure switch. Following rework, the condensate tank modules were retested by MPS AVE 93 for the flight spare and MPS AVE 92 for the flight unit. System functional and leakage tests were performed during SEDR D3-E72. This testing also included verification of caution and warning and overboard dump heater functions.

2.5.3.5 Integrated Tests

Integrated tests were conducted to verify that the vehicle was ready for flight both at the factory prior to delivery to the launch site and at the Kennedy Space Center (KSC) prior to launch.

A. Factory Tests - The tests conducted by MDAC-E in St. Louis are summarized below. Integrated test requirements were specified by SEDR.

- Atmospheric Control System - The system was operated in a support mode during simulated flight (SEDR D3-E75, Vol. I). The support operation included verification of fan operation and molecular sieve valve cycling. The mole sieve beds were baked out at the end of the test. Proper operation of the ventilation system was verified under simulated altitude conditions during the D3-E73-1 altitude chamber test. This included cabin environment contamination removal, humidity and temperature control, and verification of proper flow rates. Tests were conducted during unmanned and manned altitude runs.

Power consumption of each of the 8 PLV fans was determined by tests per MPS AVE 122. Frequency output of each mole sieve compressor power inverter was determined after the altitude chamber test (SEDR D3-E73) per MPS 171.

The system was operated again in a support mode during simulated flight (SEDR D3-E75, Vol. II). At the end of the test the mole sieve beds were baked out in preparation for shipment.

- Gas System - During simulated flight (SEDR D3-E75, Vol. I) the O_2 and N_2 gas system was operated in a support mode.

  During the altitude chamber test (SEDR D3-E73) the system was operated in a flight mode except the gas storage tanks were not pressurized. O_2 and N_2 was supplied from GSE connected to the vehicle high pressure lines. This included supplying N_2 for mole sieve valve cycling, O_2 for EVA/IVA operations, and O_2 and N_2 for cabin pressurization.

  During simulated flight (D3-E75, Vol. II) the gas system was operated in a support mode. Mole sieve beds were cycled and instrumentation parameters monitored during this test. Proof pressure and leakage measurement tests of that portion of the O_2 system installed in the FAS were conducted per SEDR D3-L62. This included all plumbing and components from the O_2 tank check valves to the FAS/AM interface. The tank assemblies had been previously verified by SEDR D3-G62.

- Condensate System - The system was operated in a support mode during simulated flight (SEDR D3-E75, Vol. I). Caution and warning checks and overboard dump heater functional tests were performed again during this test.

  After the simulated flight test, condensate tank modules were modified to allow use of gas from the gas side of the tank to purge water from dump lines after a normal condensate dump operation. Checkout of the module after modification was accomplished by MPS AVE 161 for the flight spare and MPS AVE 160 for the flight unit.

The system was used without the tank module for condensate removal during the manned altitude chamber test (the tank module could not be used due to the 1-g environment).

After the manned altitude chamber test, the condensing heat exchangers were treated with a germicide and dried. Also, one of the condensate removal flexible hoses was reorientated to provide a better installation. Retest of the hose was accomplished by MPS AVE 174.

Some quick disconnects were replaced and O'rings relubed to the latest Q.D. configuration. After Q.D. modification retest was accomplished by MPS AVE 188. The system was operated in a support mode during simulated flight test (SEDR D3-E73, Vol. II).

The final test at the MDAC-E facility was a system flow test to determine pressure drop in the condensate system between the tank module and the OWS/AM interface. This test was performed by MPS AVE 206.

B. KSC Tests - Launch site test requirements were specified in MDC Report E0122, Specification and Criteria at KSC for AM/MDA Test and Checkout Requirements, and KSC Report KS2001, Test and Checkout Plan. Results of testing at KSC together with factory test results are presented in Figures 2.5-38 through 2.5-40 for the gas system, atmosphere control system, and condensate system, respectively.

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
1.0 COMPARTMENT PRESSURE RELIEF VALVES						
1.1 FWD, LOCK & AFT valves (3) crack and reseat pressure valve 30[illegible], 313, 391	Crack and reseat between 5.5 and 6.0 psid	[illegible]-N57-1	Crack Press FWD 5.65 psid LOCK 5.72 psid AFT 5.70 psid	KM-0033	5.68 psid 5.72 psid 5.78 psid	Leakage 5.5 35 psid 1[illegible] scc, 10 sccm 0 sccm
1.2 FWD, LOCK AFT relief valve manual shutoff	Each valve operates manually. Stops align with panel indications.	D3-N70-1	Verified		Verified	
2.0 COMPARTMENT VENT VALVES		MDA-[illegible]-2000				
2.1 MDA four-inch vent valves (2) leak check	1.37 SCCS max at 5.0 + 0.1 psid N_2		#1 0.047 sccs #2 0.033 sccs		0.86 sccs 0.86 sccs	
2.2 MDA four-inch vent valves (2) and manifold [illegible] leak test	0.44 SCCS max with MDA pressurized to 5.0 ± 0.2 psid air.		0.44 sccs		5.678x10[illegible] sccs	
2.3 MDA S191 Window Cover Mechanism leak test						
2.3.1 Leakage of window cover latch mechanism with cover in the simulated closed position	6.7 x 10^{-2} SCCS max with MDA pressurized to 5.0 + 0.2 psid air.	MMC	Part of overall leak test		0.[illegible] sccs	
2.3.2 Leakage of window cover crank mechanism with cover in the simulated closed position	6.7 x 10^{-2} SCCS max with MDA pressurized to 5.0 + 0.2 psid air.	MMC	Part of overall leak test		4.88x10^{-3} sccs	
3.0 COMPARTMENT EQUALIZATION VALVES						
3.1 MDA docking port valves (2)	Each valve operates manually. Valve vanes indicate OPEN and CLOSED positions. Valve handle locks in the OPEN and CLOSED positions	D3-E72-1	Verified		Verified	
4.0 MDA VENT VALVE RESPONSE VERIFICATION						
4.1 Verify vent valve closing time.	8 seconds maximum for each valve from command to close to "CLOSED" indication.	D3-E75-1	#1 6.2 sec. #2 6.2 sec	KS-0045	Verified	
GAS STORAGE						
1.0 O_2 SYSTEM						
1.1 Proof test lines between tank check valves and 120 psi regulator at 6000 $^{+100}_{-0}$ psig	No visible permanent deformation when pressure held for 3 minutes		N/A	KM-3000	Verified	
1.2 Leak test braze connections/weld seams from tanks to check valves at 2400 + 100 psig helium	No allowable leakage	D3-G62-1	Verified	KM-5001	Verified	See note 2. High He background accepted per waiver No. MDAC-AM-WR-19
1.3 Leak test braze connections between tank check valve and inlet of 120 psi regulator at [illegible] + [illegible] psig helium	No allowable leakage		N/A	KM-3000	Verified	See note 2
1.4 Service the [illegible] bottles for flight						
1.4.1 [illegible]	Total [illegible] shall be [illegible] ± 11 lbs as determined by individual bottle pressure-temperature relationship.		[illegible]	KM-3002	[illegible] lbs	

FIGURE 2.5-38 ECS GAS SYSTEM REQUIREMENT VERIFICATION (SHEET 1 OF 6)

REQUIREMENTS		VERIFICATION				[illegible]
DESCRIPTION	SPECIFICATION	FACTORY		[illegible]		
		PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
GAS STORAGE (continued)						
[illegible] gas sample	[illegible] sample [illegible] shall meet [illegible] 64M00129 except [illegible] level shall not [illegible] except moisture content which shall be 1.9 ppm [illegible]		N/A	KM-5002	High halogenated hydrocarbon count [illegible] tanks No. [illegible] 4 and [illegible]	[illegible] halogenated hydro- [illegible] per waiver MDA-AM-W[illegible] [illegible] 30
[illegible] Final bottle leak-age	No detectable leakage allowed		N/A	[illegible]	Verified	[illegible]
[illegible] TM pressure parameter operation	Indications from TM parameter shall be within [illegible] the [illegible] indications during servicing		N/A	KM-5002	Verified	
[illegible] TM temperature parameter qualitative operation	Qualitative check of [illegible] temp parameters		N/A	KM-5002	Verified	
[illegible] SYSTEM						
[illegible] Leak test accessible braze connections weld seams from tanks to check valves at 1500 + 100 [illegible] helium	No allowable leakage	[illegible]	[illegible]	[illegible]	Verified	[illegible]
[illegible] Leak test braze connections on T[illegible] #4 and [illegible] charge [illegible] disconnect at 3000 + 150 [illegible] helium	No allowable leakage	[illegible]	[illegible]	KM-3000	Verified	See [illegible]
[illegible] test brazed connections in system [illegible] and #3 to the M[illegible] shutoff valve and to the [illegible] regulator inlet at [illegible] helium	[illegible]	[illegible]	[illegible]	KM-3000	[illegible]ified	[illegible]
[illegible] service the six N[illegible] bottles for flight						
[illegible] 4 [illegible] mass	Total N[illegible] shall be [illegible] lbs [illegible] determined by [illegible] bottle pressure temperature relationship		N/A	KM-5002	[illegible]	
[illegible] 4 [illegible] gas sample	Gas sample from each bottle shall meet requirements [illegible] MSFC Spec 234A, Type [illegible] except particle count level shall not apply		N/A	[illegible]M-5002	High halogenated hydrocarbon count	Accepted per waiver No MDAC-AM-WR-21
[illegible] 4 [illegible] Final bottle leak-age	No detectable leakage allowed		N/A	[illegible]	[illegible]fied	Detectable leakage defined [illegible] a [illegible]4 psi decrease below the fill density from time of stabilization after servicing thru redline monitoring
[illegible] 4 TM pressure parameter operation	Indications from TM param[illegible] shall be within + 1.5 [illegible] the G[illegible] indication during servicing		N/A	KM-5002	Verified	
[illegible] 4 [illegible] TM temperature parameter qualitative operation	Qualitative check of bottle temp TM parameter	[illegible]	[illegible]	KM-5002	Verified	

FIGURE 2.5-38 ECS GAS SYSTEM REQUIREMENT VERIFICATION (SHEET 2 OF 6)

REQUIREMENT		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
GAS SUPPLY						
1.0 SYSTEM LEAK TESTS						
1.1 Leak test system from 120 psi regulator to EVA/IVA disconnects, O_2 fill shutoff valves and cabin pressure regulator at 120, +10, -0 psig N_2.	2.28 SCCM max.	D3-E76-1	0.0 sccm	[illegible]-0003	1.01 sccm	IDR 072 High leakage rate at QD. Upgraded to DR AM1-07-0079. (Verified). Retest due to H_2O bladder replacement was 3.0 sccm. Ref waiver MDAC-AM-WR-34.
1.2 Leak test system from 150 psi regulator to OWS N_2 [illegible] Mol Sieve A&B [illegible] interface, inlet of 5 psi regulator, OWS O_2/N_2 press. line interface at 150, +10, -0, psig N_2.	2.85 SCCM max.	D3-E76-1	0.0 sccm		0 sccm	
1.3 OWS pressure sense line leakage at 15.0 ± 0.1 psig N_2.	0.2 SCCM max.	D3-N70-1	Zero		0.19 sccm	
1.4 Leak test system from 5 psi regulator to ATM coolant reservoir inlet and suit coolant reservoir inlet (sys 1&2) at 5 psig N_2.	1.52 SCCM max.	D3-E76-1	0.34 sccm		0.80 sccm	This test invalidated by replacement of bladders in VAB.
1.5 Leak test 150 psi regulator vent line with 5.0 ± 0.2 psid across line.	No detectable leakage at GSE 474 using bubble check technique.	D3-N57	Verified		Verified	
2.0 FUNCTIONAL TESTS						
2.1 Valve operation						
2.1.1 Primary and secondary O_2 shutoff valves	Evidence of flow/no flow through individual valves upon manual switching and DCS command. TM events occur properly as valves are commanded.	D3-E76-1				
2.1.2 Primary and Secondary N_2 shutoff valves	Evidence of flow/no flow through individual valves upon manual switching and DCS command.					
2.1.3 120 psi regulator shutoff valves.	Evidence of flow/no flow through individual valves.					
2.1.4 Cabin pressure regulator shutoff valves.	Evidence of flow/no flow through individual valves.					
2.1.5 OWS N_2 shutoff valve	Evidence of flow/no flow through valve. (8 ± 1 turns full travel)					
2.1.6 150 psi regulator shutoff valves.	Evidence of flow/no flow through individual valves.					
2.1.7 Primary and secondary N_2 fill shutoff valves.	Evidence of flow/no flow through individual valves upon manual switching and DCS command.					
2.1.8 Primary and secondary O_2 fill shutoff valves	Evidence of flow/no flow through individual valves upon manual switching and DCS command.					

FIGURE 2.5-38 ECS GAS SYSTEM REQUIREMENT VERIFICATION (SHEET 3 OF 6)

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
GAS SUPPLY (Continued)						
2.1.9 5 psi regulator shutoff valves.	Evidence of [illegible]/no flow through individual valves.	D3-E76-1	Verified	KM-0003	Verified	
2.1.10 Primary and secondary OWS fill shutoff valves.	Evidence of flow/no flow through individual valves upon manual switching and DCS command.					
2.1.11 Primary and secondary AM fill shutoff valves.	Evidence of flow/no flow through individual valves upon manual switching and DCS command.					
2.1.12 N_2 primary/secondary selector valve	Evidence of flow through valve upon manual operation to each position.					
2.1.13 Aft compartment N_2 bottle shutoff valves (2)	Evidence of flow/no flow through individual valves.					
2.1.14 Aft compartment M509 shutoff valve.	Evidence of flow/no flow through valve. (8 ± 1 turns full travel)					
2.1.15 Aft compartment M509 umbilical vent valve.	Evidence of flow/no flow through valve. (8 ± 1 turns full travel)					
2.1.16 Calibrate hand valve	Evidence of flow through valve. (8 ± 1 turns full travel)					
2.1.17 Vacuum hand valve	Evidence of flow/no flow through valve.					
3.0 CHECK VALVE LEAK TESTS						
3.1 120 psi regulator check valves at 140 ± 5 psig N_2	Reverse leakage: 0.2 SCCM max per valve	D3-G63-1	Zero		Zero	
3.2 150 psi regulator check valves at 175 ± 5 psig N_2.	Reverse leakage: 0.2 SCCM max per valve	D3-G63-1	Zero		Zero	
3.3 Check valve at outlet of O_2 fill shutoff valve at 175 ± 5 psig N_2.	Reverse leakage: 0.2 SCCM max	D3-G63-1	Zero		Zero	
3.4 O_2 check valves at inlet of O_2 fill shutoff valve at 175 ± 5 psig N_2	Reverse leakage: 0.2 SCCM max per valve.	D3-G63-1	Zero		Zero	
4.0 PRESSURE SWITCH OPERATION.						
4.1 OWS fill switches (2)	Each switch closes at 4.8 psia minimum.	D3-E76-1	(P) 4.9 psia (S)4.95 psia		5.8 psia pri 5.1 psia sec	
	Each switch opens at 5.0 ± 0.2 psia.		(P)4.95 psia (S)4.95 psia		4.95psia pri 4.95psia sec	
4.2 AM fill switches (2)	Each switch closes at 4.8 psia minimum.	D3-G63-1	(P)4.97 psia (S)4.98 psia		(P) 5.0 psia (S)4.97 psia	
	Each switch opens at 5.0 ± 0.2 psia.		(P)4.96 psia (S)4.95 psia		(P) 5.0 psia (S)4.15 psia	
5.0 REGULATOR AND RELIEF VALVE OPERATION						
5.1 150 psi regulators A & B	REGULATIONS:					
	a. 150 ± 17 psig at flow of 1.7 x 10^{-3} lb/sec N_2 to 2.3 x 10^{-3} lb/sec N_2. Flow measured at OWS O_2/N_2 interface exhausting to ambient.	D3-E76-1	(A)2.25x10^{-3} lb/sec (B)2.22x10^{-3} lb/sec		2.15 x 10^{-3} lb/sec @ 145 psig 2.1[illegible] x 10^{-3} lb/sec @ 146 psig	(A) 168 psia Reg Outlet @ St. Louis (B) 164 psia Reg Outlet @ St. Louis

FIGURE 2.5-38 ECS GAS SYSTEM REQUIREMENT VERIFICATION (SHEET 4 OF 6)

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
GAS SUPPLY (continued)						
5.1 (Continued)	b. 150 ± 17 psig at flow of 3.1×10^{-3} lb/sec N_2 to 4.0×10^{-3} lb/sec N_2. Flow measured at OWS N_2 interface exhausting to ambient.	D3-E76-1	(A)3.7×10^{-3} lb/sec (B)3.72×10^{-3} lb/sec	KM-0003	3.55×10^{-3} @ 146 psig 3.65×10^{-3} @ 148 psig	(A) 168 psia Reg Outlet @ St. Louis (B) 166 psia Reg Outlet @ St. Louis
	c. Lockup: 187 psig maximum after 10 minutes.	D3-E76-1	175 psia		156.5 psig - GSE 175 psia - Panel 225	
5.2 150 psi reg A & B relief valves.	180 psig minimum reseat pressure. 210 psig maximum cracking pressure.	D3-G63-1	(A)182.5psig (B)187.5psig (A)196 psig (B)197 psig		199psig-open GSE 185psig Reseat 200psig-open 187psig Reseat	
5.3 120 psi regulators A & B	REGULATIONS a. 120 ± 16 psig at flow of 5.7×10^{-3} lb/sec N_2 to 7.3×10^{-3} lb/sec N_2. Flow measured at OWS O_2/N_2 interface exhausting to ambient.	D3-E76-1	(A)5.78×10^{-3} lb/sec (B)5.85×10^{-3} lb/sec		5.12×10^{-3} lb/sec (Pri) @105 psig 5.12×10^{-3} lb/sec (Sec) @106 psig	(A) 124 psia Reg Outlet (B) 128 psia Reg Outlet
	b. 120 ± 16 psia with qualitative verification of flow (through pri & sec AM fill valves).	D3-E76-1	Verified		Verified	
	c. Lockup: 151 psig maximum after 10 minutes.	D3-E76-1	145 psia		142 psia	
5.4 120 psi regulators A & B relief valves	144 psig minimum reseat pressure 170 psig maximum cracking pressure/	D3-G63-1	(A)160 psig (B)157 psig (A)167 psig (B)166 psig		153.0 psig 154.8 psig 164.0 psig 164.5 psig	
5.5 5 psia regulators A & B	REGULATIONS a. Flow: 5.0 ± 0.4 psia at flow of 7.7×10^{-4} lb/min to 7.7×10^{-5} lb/min N_2. Flow measured at ATM tank module.	D3-E76-1	(A)2.3×10^{-4} lb/min (B)2.56×10^{-4} lb/min		3.10×10^{-4} lb/min @ 4.9 psia 3.10×10^{-4} lb/min @ 5.01 psia	
	b. Lockup: 5.5 psia maximum after 10 minutes. (@ GSE)	D3-E76-1	5.22 psia (D23A) 5.35 psia (GSE Gage)		P_A=5.31psia P_B=5.46psia	
5.6 5 psia regulators A & B relief valves	5.6 psia minimum reseat pressure. 6.2 psia maximum cracking pressure.	D3-G63-1	(A)5.86psia (B)5.7 psia (A)6.0 psia (B)5.9 psia		(A)6.12psia (B)6.11 psia (A)6.2 psia (B)6.01 psia	Valves were replaced & retested in VAB due to ATM C&D Loop bladder rupture.
5.7 5 psia cabin regulator A & B	REGULATIONS: a. 1.55×10^{-2} lb/min N_2 to 2.01×10^{-2} lb/min N_2 with outlet pressure maintained at 4.90 +0.05, -0.00 psia. Regulator inlet pressure maintained at 150 ± 5 psig N_2.	D3-E76-1	(A)1.84×10^{-2} lb/min (B)1.84×10^{-2} lb/min		1.65×10^{-2} lb/min (A) @ 4.80 psia 1.91×10^{-2} lb/min (B) @4.85 psia	(A) 4.8 psia Reg Outlet @ St. Louis (B) 4.83 psia Reg Outlet @ St. Louis
	b. Lockup: 5.3 psia maximum after 5 minute monitor. (@ GSE) 5.55 psia maximum (@D256)	D3-E76-1	5.19 psia 5.25 psia		5.15 psia 5.13 psia	

FIGURE 2.5-38 ECS GAS SYSTEM REQUIREMENT VERIFICATION (SHEET 5 OF 6)

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
GAS SUPPLY (continued)						
6.0 O_2/N_2 CONTROLLER FUNCTIONAL TEST						
6.1 PPO_2 sensors flooded with N_2 for Control & Monitor positions 1, 2, 3	TM parameters (PPO_2) indicate: $0.0 \begin{smallmatrix} +17.0 \\ -0.0 \end{smallmatrix}$ mm Hg Cabin PPO_2 meters indicate: $0 \begin{smallmatrix} +0.33 \\ -0.00 \end{smallmatrix}$ psia PPO_2 LOW light illuminates.	D3-E73-1	1) 2.4 mmHg 2) 0 3) 0 1) 0.2 psia 2) 0 3) 0	KM 0003 ↓	11 mm Hg 6 mm Hg .06 mm Hg 0.3 No units 0.1 " 0.2 "	
6.2 PPO_2 sensor operation at ambient (air) conditions	TM parameters (PPO_2) indicate: 160.3 ± 17.0 mm Hg Cabin PPO_2 meters indicate: 3.1 ± .33 psia PPO_2 LOW light remains out.	D3-E70-1	#1 153mm Hg #2 157mm Hg #3 150mm Hg #1 2.9 psi #2 2.8 psi #3 2.9 psi		D0237 159 mm Hg D0239 170 mm Hg D0240 152 mm Hg 3.0 psi 3.1 psi 2.95 psi	St. Louis AMB PPO_2 156 mm Hg
6.3 N_2 solenoid valve operation - Flood sensors alternately with O_2/N_2 mixture (35% O_2/65% N_2) and 100% N_2.	N_2 solenoid valves cycle open and closed as evidenced by gas flow through valves.	D3-E76-1	Verified		Verified	
6.4 Flow rate from M509 recharge station using 1-2 position and 3-6 position with 3000 +100, -0, psig @ GSE 505 & GSE 529.	Qualitative flow verification.	D3-E76-1 D3-E72	Verified	↓	Verified	D3-E76-1 Flow Check @ 1000 psia D3-E72-1 Fast fill test of bottle @ 2800 psig.

GENERAL NOTES:

1. Allowable leakage specified as 1 x 10^{-3} SCCS helium maximum shall be leak-checked using the helium mass spectrometer leak detector (sniffing mode). Points that are unacceptable shall be rechecked using a Uson leak detector. Only those points where leakage can be verified using the Uson detector are unacceptable.

2. No leakage allowed shall be defined as no detectable leakage, above background, using a helium mass spectrometer (sniffing mode) set to maximum machine sensitivity. Background must be stable and shall not exceed an indicated background value of 1 x 10^{-8} sccs helium.

FIGURE 2.5-38 ECS GAS SYSTEM REQUIREMENT VERIFICATION (SHEET 6 OF 6)

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
1.0 LEAK TESTS						
1.1 Condensing H/X module A&B check valve reverse leakage.	5.8 in. H_2O minimum ΔP with mole sieve compressor operating.	3-N70-1	MS A) 6.52" H_2O B) 7.58" H_2O	KM-0003		
1.2 Molecular sieve A&B overboard vent duct and valve at 5.5 + 0.1 psid (cabin to duct)	4.14×10^{-6} lb air/min maximum leakage.		A) 3.95×10^{-7} B) Zero ($< 1\times10^{-7}$)		$< 2.16\times10^{-6}$ 3.5×10^{-6}	
2.0 FAN/COMPRESSOR OPERATION						
2.1 Mol sieve compressors (4) (pri & sec) operation via manual switching.	Presence/absence of flow at mol seive outlet. TM events occur properly during system operation. Compressor delta pressure: MS "A" - 7.0 + 1.2 in. H_2O MS "B" - 7.0 + 1.2 in. H_2O		MS A) 6.52" H_2O B) 7.58" H_2O		Flow verif. TM events occured properly. Mol SV A ΔP Fan 1 7.55" H_2O Fan 2 6.27" H_2O Mol SV B ΔP Fan 1 7.48" H_2O Fan 2 6.63" H_2O	ΔP recorded for two comp. only during ΔP Xducer C/O check valve leakage checked @ module level @ 10" H_2O - 1.2 sccm for "A". 1.5 sccm for "B" Mol SV A & B Beds 1 & 2 in storage position. Cond. HX air vlvs in "B position.
2.2 AM interchange duct fan (1) (high/low position)	Presence/absence of flow.		Verified		Hi/Lo flow verified	
2.3 OWS cooling fans (4) operation via manual switching (AM).	Presence/absence of flow in duct. TM events occur properly during fan operation.	D3-E76-1			Verified	KS-0045 verified flow for each fan.
2.4 Cabin H/X fans (3) (high/low position)	Presence/absence of flow at fan inlet.	D3-E75-1				
2.5 MDA cabin fans (2) (high/low position)	Presence/absence of flow at duct outlet.	D3-E75-1				
2.6 CSM port fan (high/low position)						
2.6.1 Normal electrical connector	Presence/absence of flow in duct.	D3-E75-1				
2.6.2 Spare electrical connector.	Presence/absence of flow in duct.	D3-E75-1				
2.7 MDA fan diffuser adjustment	Diffusers adjust with one hand.	D3-E75-1				
3.0 FLOW RATES & SELECTION						
3.1 MDA/OWS air selector valve 234	Air flow is directed to MDA/OWS upon manual operation of valve.	D3-N70-1				
3.2 Mol Sieve condensing H/X air flow valve 233 and 239	Mol sieve air flow is noted upon manual operation of each valve (all positions).	D3-N70-1				
3.3 Mole sieve A flow sensor	TM indicates a flow of 18.3 cfm minimum for each compressor (2), for each H/X selector valve position (3), and with sorbent bed selection (ADSORB) For STORAGE position minimum flow is 15.0 cfm.	D3-E75-1 Vol. II	P) 27 cfm S) 26 cfm		No. OWS HX Fans On / Flow F210 cfm 4 28.6 3 30.1 2 26.5 1 21.0 0 26.7	(St.L)H/X valve - B position Bed 2 - Adsorb (KSC) Pri fan only. CHX vlv - position B only. Bed 1 adsorb Bed 2 storage

FIGURE 2.5-39 ECS ATMOSPHERIC CONTROL SYSTEM REQUIREMENT VERIFICATION (SHEET 1 OF 3)

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
3.4 Mol sieve B flow sensor	TM indicates a flow of 18.3 cfm minimum for each compressor (2), for each H/X selector valve position (3), and with sorbent bed selection (ADSORB) For STORAGE position minimum flow is 15.0 cfm.	D3-E75-1 Vol. II	P) 30 cfm S) 27 cfm	KM-0003	No. OWS HX Fans On / Flow F211 cfm 4 30.1 3 27.0 2 27.7 1 29.7 [illegible] 29.7	Pri Fan only CHX valve - position B only Beds 1 & 2 - storage.
3.5 Interchange duct flow sensor	TM indicates a flow of 104 cfm minimum.	D3-E76-1	120 cfm		109 cfm	
3.6 OWS cooling duct flow sensor	For single fas operation 45 cfm minimum (each of 4 fans) For multiple fan operation: (2) fans 106 cfm minimum. (3) fans 147 cfm minimum	D3-E73-1 D3-E76-1	1) 109 cfm 2) 100 to 153 cfm 3) 123 to 156 cfm 4) 101 to 149 cfm (2 fans) 162 cfm (3 fans) 190 cfm		109 cfm (2 fans) 173 cfm	Single fan operation verif for fan 1 only. IDR 132: F209 read UPU with all 4 OWS HX fans off and AM Duct fan - Lo. Upgraded to DR AM 1-07-0156 Flowmeter F209 retested in KS-0045 with same results. Explained by duct turbulence. Operated normally with OWS duct fans on during KS-0045.
4.0 MOLECULAR SIEVE OPERATION - The following items are applicable to each Mol Sieve (A & B). Test all combinations of timer valves/manual inter-connect valves						
4.1 Gas selector valve	Valve travels to commanded position (manually and pneumatically). TM events occur properly during operations.	D3-E72	Verified		Verified	
4.2 Solenoid valves (4)	Valves open/close (gas flow/no flow) upon command (manual and timer).	D3-E72	Verified		Verified	
4.3 Timer	a. Operates solenoid valves at intervals of 880 ± 15 seconds, resulting in the beds alternately adsorbing and desorbing.	D3-E72	Verified		Verified	Also verified in KS-0045, Seq. 35-010 & 018. DR AM1-07-0011. Mole Sieve A timer short cycle.
	b. C&W SV TMR light illuminates when power is removed from timer circuit.	D3-E72	Verified		Verified	
4.4 Mol sieve A & B vent valve internal leakage.	Maximum allowable leakage is 4.87×10^{-7} lb/min at 5 PSID air. (Cabin to Duct)	D3-N70-1	A) 2.6×10^{-7} B) 0		4.09×10^{-7} lb/min	IDR 106-couldn't pull a vacuum - Reason mol sieve vent valve left open.
4.4.1 Leak test mol sieve TF 13 connection.	Maximum allowable leakage 4.15×10^{-6} lb/min per Mol sieve @ 5.5 psi below ambient.	D3-N70-1	A) 0 B) 0		3.58×10^{-7} lb/min 3.5×10^{-7} lb/min	
4.4.2 Leak test mol sieve gas selector valves @ ADSORB side of STORAGE band (pointer on inside edge of band).	Maximum allowable leakage 3.63×10^{-6} lb/min air per mol sieve @ 5.5 psi below ambient	D-N70-1 D-G52	A) Zero B) Zero		Mol Sieve A 9.8×10^{-6} lb/min. Mol Sieve B 2.9×10^{-6} lb/min	Measurement in 42-143 exceeds spec. value of 3.63×10^{-6} lb/min. Retest per Dev 1296 passed with measurement of 2.9×10^{-7} lb/min.
4.5 Bakeout of sorbent beds (A & B)	Controller starts cycling when panel 203 indication is 330 to 440°F. SIEVE TEMP HIGH light remains out during system operations.	D3-E73-1	A) 380°F B) 380°F Verified	KS-0016	Verified	
4.6 Dryout of Condensing Heat Exchanger (4)	Verify water vapor at GSE 407 on each heat exchanger is less than 3442 ppm (20°F)	D3-E87-1	Verified	KS-0016	Verified	

FIGURE 2.5–39 ECS ATMOSPHERIC CONTROL SYSTEM REQUIREMENT VERIFICATION (SHEET 2 OF 3)

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
5.0 PCO_2 SENSOR TEST						
5.1 PCO_2 sensor qualitative test	Qualitative response to presence of CO_2. TM events and indications occur properly during operation.	D3-F76-1	Verified	KM-0003	Verified	IDR 214 - Procedure error (Manner of valving pressure sensor) IDR 225 - Additional testing resulted in new IDR 246 (Replaced by DR AM 1-70-0162) Erroneous pressure sensor reading - Fix by install. of orifices & repl'c filter. Closed by Dev 2265.
6.0 PURGE REQUIREMENTS						
6.1 AM/MDA purge with N_2	Purge AM/MDA with 225 lbs N_2 minimum for 45 minutes min.	D3-E56-1	Verified			
6.2 MDA superinsulation purge	Gas flow from vent holes in S190 external window cover with cover in latched position	D3-E56-1	Verified			
6.3 MDA superinsulation purge line leak test.	No evidence of leakage at the FAS/Trus #4 interface connection, using bubble check solution.		N/A			
6.4 Nose cone purge duct	No evidence of audible leakage at the PS/FAS interface or at the PS cylinder/PS cone interface.		N/A			

FIGURE 2.5-39 ECS ATMOSPHERIC CONTROL SYSTEM REQUIREMENT VERIFICATION (SHEET 3 OF 3)

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
1.0 LEAK TEST Perform leak test on H_2O side of condensate system at 5 psig below ambient under the following conditions:						Actually run TPS AM-07-0147 IDR 149 (Procedural error) (corrected by Dev 1605) Test Run in KM-0003 with tank press vlv "Press" Tank H_2O vlv - "Off"
1.1 Plates wet and QD's connected. Condensate tank H_2O fill valve in fill position. Pressure valve in press position, condensate S/O valves (4) open.	Maximum allowable leakage is 5.56 SCCM.	Different configuration tested at factory.		KS-0045	1.42 sccm	Leakage at QD condensate tank fill side. Removed and replaced QD. (Dr AM 1-07-0142)
1.2 Vacuum shutoff valve open. Condensate tank H_2O valve in dump position. Press valve in press position, condensate S/O valves (4) open, one CHX condensate line in stowed position.	Maximum allowable leakage is 2.49 SCCM.	Different configuration tested at factory.		KM-0003	0.21 sccm	
1.3 Condensate tank H_2O valve in FILL position. Pressure valve in VACUUM position, one condensing ht. exch. condensate line in stowed position. Condensate shutoff valves (4) open.	Maximum allowable leakage is 1.92 SCCM.	Different configuration tested at factory.			0.58 sccm	Run by TPS AM-07-0147
2.0 SENSOR FUNCTIONAL TESTS						
2.1 Cabin indicator - delta pressure.	Tank ΔP indicator within $\pm$ 0.4 psid of GSE indication. TM indication within + 0.2 psid of GSE indication.	D3-N70-1	See Remarks Column		See Remarks Column	GSE Gage PSID / Tank ΔP Meter PSID / TM PSID Factory: 0 / 0 / 0.1; 2.4 / 2.4 / 2.5; 5.0 / 5.1 / 5.0; 2.5 / 2.4 / 2.5; 0 / 0 / 0.1 KSC: 2.49 / 2.4 / 2.48; 5.0 / 5.0 / 5.07
2.2 C&W - delta pressure	CNDST TANK ΔP light - ON when delta pressure is 0.3 to 0.8 psid. CNDST TANK ΔP light - OUT when delta pressure is 0.8 psid or more.	D3-N70-1	Sp. 0.5 psid Flt.0.45psid		0.50 0.49	
3.0 VALVE OPERATION						
3.1 Condensate tank H_2O valve	Evidence of flow/no flow.	D3-G64-1	Verified		Verified	
3.2 Condensate tank pressurization valve	Evidence of flow/no flow.	D3-G64-1	Verified		Verified	
3.3 Condensate control system vent valves (PRI & SEC)	Evidence of flow/no flow.	D3-G64-1	Verified		Verified	
3.4 Lock compartment vacuum source valve, 316.	Evidence of flow/no flow.	D3-N70-1	Verified		Verified	

FIGURE 2.5-40 ECS CONDENSATE SYSTEM REQUIREMENT VERIFICATION

2.5.3.5 Mission Support Tests

Mission support tests were conducted during the flight of U-1 using both the Static Test Unit (STU) and U-2. Specific requirements for these tests were generated during U-1 flight.

A. STU Simulation - Normally the STU was operated to duplicate U-1 flight conditions. Special tests were conducted on an as required basis. A summary of special ECS tests performed utilizing the Skylab Test Unit is presented below. Test details as well as descriptions of STU are presented in the ECS/TCS Skylab Test Unit Report No. TR 061-068.99.

- Title - Oxygen Regulator Test with Cold Gas

  Background - U-1 oxygen tank temperatures were low. Oxygen temperatures at the regulator inlet port could have been as low as -30°F at 2150 psig pressure.

  Objective - Evaluate operation of the 61A830383 oxygen regulator at low inlet gas temperature, Reference TR 061-015-600.03.

  Results - Oxygen regulator performance was satisfactory for all test conditions, Reference TM252:667.

- Title - Deionizer Filter Assembly, 1B89235-505, High Temperatu Test.

  Background - T deionizer filter assembly may have been exposed to peratures which could cause overpressurization ring storage in U-1.

  Objective - etermine if exposure to high temperature (130°F) after eing fully serviced with water at 60°F would cause permanent deformation and thus leakage at operating pressure of 40 psig, Reference TR 061-015-600.09.

  Results - Deionizer filter assembly exposure to high temperature did not cause deformation nor leakage, Reference TM 252:650.

- Title - CO_2 Detector Filter Cartridge Performance Verification Test.

  Background - During the SL-2 mission the inlet CO_2 detectors indicated cabin CO_2 levels of 1 to 2 mm Hg less than the M171 mass spectrometer values.

Objective - Compare the output of inlet CO_2 detectors using fresh filter cartridges with the output of the same detector using the filter cartridges returned from the SL-2 mission, Reference TR 061-015-600.11.

Results - Outputs from detectors SN 262 and 260 were found to agree within 0.1 to 0.2 mmHg, respectively. However, there was an offset of 0.6 to 1.1 mmHg from the PIA calibration curves, Reference TM 252:670.

- Title - Mole Sieve Compressor Power Inverter Test

Background - Circuit breaker opened in U-1 when mole sieve "B" compressor #2 was turned on.

Objective - Determine current required for compressor startup, Reference TR 061-015-600.15.

Results - Compressor start and run currents were 4.4 amps and 2.9 amps, respectively. Time for compressor startup was approximately 20 seconds. Reference TM 252:681.

- Title - Mole Sieve Flowmeter Test

Background - When the U-1 AM fill valve was opened, mole sieve "A" flowmeter (F210) indicated a low flow (25 CFM) that activated C&W while mole sieve "B" flowmeter (F211) indicated high flow (off scale).

Objective - Determine the effect of flow through the AM fill valve on mole sieve "A" and "B" flowmeters under simulated U-1 flight conditions, Reference TR 061-015-600.40.

Results - When the STU AM fill valve was opened under simulated U-1 flight conditions, mole sieve "A" flowmeter reading decreased but the "low flow" light did not illuminate. Mole sieve "B" flowmeter indicated high flow (off scale), Reference TM 252:732.

- Title - Exploratory Test of the High Pressure N_2 Regulator Performance Characteristics

Background - During SL-2 and SL-3 mission, the high pressure N_2 regulator gradually decreased below specification limits. The gradual decreasing regulator control pressure had not been experienced in tests including the mission support simulation tests performed on the STU.

Objective - Determine if another regulator, installed in STU and subjected to U-1 conditions (except for Zero-G) would exhibit U-1 regulator operating characteristics, Reference TR 061-015-600.49.

Results - The high pressure N_2 regulator performed within specification limits at test conditions, Reference TM 252:748.

B. U-2 Testing - A summary of ECS test activity performed utilizing the U-2 vehicle in support of the U-1 mission is presented below.

- <u>Problem - High OWS ambient temperatures required possible work-around plan to reduce temperature levels.</u>

  Activity - A fit check of the MDA flex duct between the OWS and AM air return duct was performed, this was in response to AR-102 and was a method to increase air interchange between the OWS and AM to reduce high OWS temperatures.

- <u>Problem - Mission time lines required possible simultaneous pressurization of AM and OWS. Evaluation of orifice flow characteristics by testing on U-2 was desired.</u>

  Activity - A study of simultaneously pressurizing the OWS and AM was made and MPS 147 versus U-2 was generated. This test successfully demonstrated the capability of pressurizing both modules simultaneously while still maintaining acceptable orifice flow control in the system.

- <u>Problem - Possible excessive leakage from vehicle condensate system resulted in need to determine leakage characteristics/dead band of condensate pressure valve (Ref SAR 6)</u>

Activity - MPS 157 was issued to test the alignment and dead band of the condensate module pressure valve. Results of the testing indicated that the pressure valve had approximately ± 25 degrees travel from the "off" position before it was in the "press" or "vacuum" mode. Possibility of leakage due to misalignment was doubtful considering amount of deadband.

- Problem - The S/L 2 crew commented during debriefing that while depressurizing the lock compartment for EVA frosting occurred on the pressure equalization valve. This frosting increases the lock equalization time and could possibly block the valve.

  Activity - Hardware to remedy this problem was fabricated and a trial fit was performed by MPS 167 vs. U-2. This new hardware was launched and used on SL-3.

- Problem - During SL-3 activation, mole sieve B secondary fan did not start and the circuit breaker opened. Troubleshooting established the problem to be with the inverter/electrical circuitry.

  Activity - A jumper cable (61A762368-1) was fabricated which permitted powering the mole sieve fan from another inverter. Also a carry on inverter with a Y cable (61A762369-1) to interconnect the inverter to fan & utility outlet was fabricated. This restored normal fan operation/redundancy by using the utility outlet to power the new inverter which in turn powered the mole sieve B secondary fan. Both the jumper cable and the inverter/Y cable assy were functionally tested on U-2 to verify proper fit/function and system compatibility.

- Problem - Gas leakage into water side of condensate system required daily dump of OWS holding tank.

Activity- A plan was devised to pressurize the condensate system plumbing to 35 psig and bubble leak check accessible joints. Pressurization source would be panel 500 in the OWS with the 60 ft. H_2O servicing umbilical and various other onboard adapters used to connect the pressurization source to the condensate system.

A fit check, except for panel 500 connection, of the umbilical and adapters was successfully conducted on U-2.

C. Bench Testing - A summary of ECS tests performed utilizing the PIA test-benches in support of the U-1 mission is presented below:

- Problem - Effect of high ambient OWS temperatures on the in-flight water servicing deionizer.

  Activity- Servicing of a qual deionizer was performed in support of a high temperature soak test. Results of the test indicated that the flight unit in SL 1/2 would not have been subjected to extreme pressures at the temperature levels experienced.

- Problem - Failure of meteroid shield to deploy necessitated need for work-around plan to reduce OWS temperatures.

  Activity- Various methods of shading the OWS were studied and some systems were built. Among these were several pneumatically deployed devices.

  (1) Three individual "K" bottle assemblies (approx. 50 cu. in. each) employing fixed orifices and shutoff valves were built and successfully tested on MPS spares 22. The orifices were independently tested and sized for a predetermined flowrate. The "K" bottle valve assembly portion of the units were proof and leak tested. The orifices were installed and the bottles pressurized to 123 psia. The depressurization time necessary for each bottle was recorded and was sufficient to meet the requirements for an inflatable shield.

(2) Another shielding method employed a three stage regulator assembly capable of regulating from 3000 psi inlet to .8 inch H_2O outlet. Testing of the regulator assembly was accomplished on MPS spares 20. Leak and functional checks of the individual components were made prior to and during the assembly build-up. A leak check of the entire unit was performed after complete assembly. The entire unit underwent a successful functional test in a vacuum environment. MPS spares 23 was written to run a negative leak check on the 61A830415 hose assembly. This hose assembly was intended for evacuation of the inflatable shield prior to pressurization.

(3) Other effort involved technical support for manufacturing on several deployable devices using compressed carbon dioxide.

2.5.4 Mission Results

The Airlock Module ECS satisfactorily performed all required functions throughout the Skylab mission, i.e. functions relating to prelaunch purge, ascent venting, gas supply and distribution, atmospheric control, and condensate removal plus some additional functions.

2.5.4.1 Gas System

In addition to performing all of the required functions, the gas system successfully performed unplanned, supplemental operations by pressurizing the cluster as part of a purge operation prior to SL-2 to eliminate possible contaminants and by pressurizing the cluster during storage following SL-3 for gyro six-pack cooling.

A. Prelaunch Purge - Airlock Module gas distribution in support of the nose cone purge, ATM canister purge, MDA insulation purge, OWS dome HPI purge, IU purge and AM/MDA purge was accomplished with no apparent problems.

B. Launch Ascent Venting - Ascent venting of the STS, lock and aft compartments was completely normal as shown by Figure 2.5-41.

C. Gas Storage - Prelaunch loading of O_2 and N_2 tanks is shown in Figure 2.5-42 and indicates that 6085.3 lb of O_2 and 1622.8 lb of N_2 were available for the mission.

During the initial structural integrity check at 5 psia following launch, there was essentially no change in cabin pressure over a 65-hr period. As a result of the low leakage rate, gas usage for normal operation of the cluster was well below design levels and significant quantities of O_2 and N_2 were still available following deactivation of SL-4. Figure 2.5-43 presents a summary of O_2 and N_2 consumables throughout the mission. It should be noted that the percentage of N_2 remaining is much lower than that for O_2 primarily due to unplanned N_2 pressurizations to purge SL-1 prior to crew arrival and to permit operation of fans for gyro cooling during the storage period following SL-3. Flight data during the final storage period would indicate a cluster storage configuration leakage rate of 2.25 lb/day at 5 psia and 50°F. No detectable leakage was encountered within the gas storage subsystem.

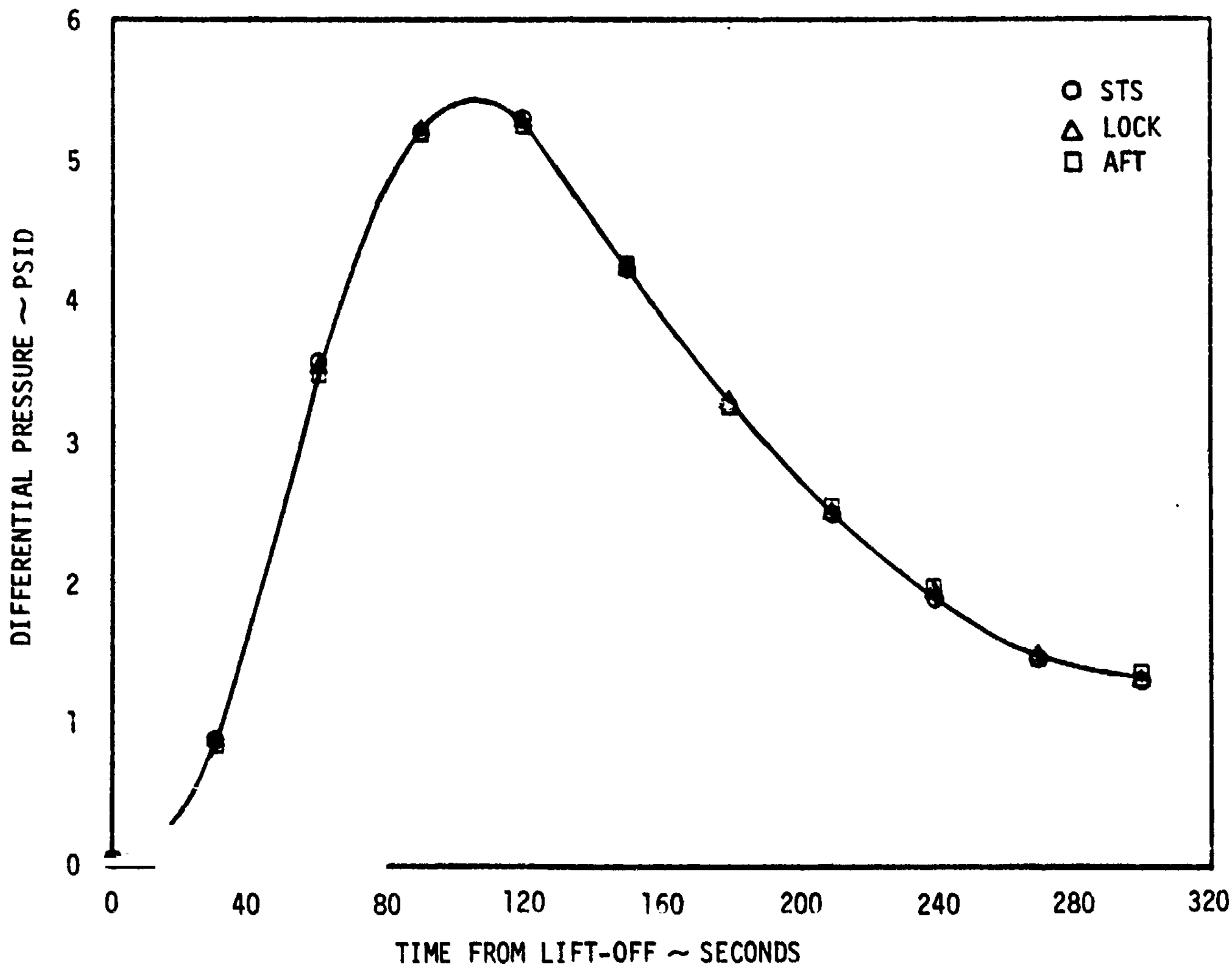


FIGURE 2.5-41 COMPARTMENT DIFFERENTIAL PRESSURES DURING ASCENT

TANK	PRESSURE (PSIA)	ALT. PRESSURE (PSIA)	TEMPERATURE (°F)	MASS (LBS)
O_2 Tank 1	2979.5	2974.4	70.1	1004.6
O_2 Tank 2	2984.3	2988.4	67.8	1011.5
O_2 Tank 3	2996.6	3014.0	70.4	1011.9
O_2 Tank 4	2980.0	2953.9	67.7	1019.3
O_2 Tank 5	2994.9	2989.6	68.9	1021.4
O_2 Tank 6	3013.6	3024.5	72.4	1016.6
TOTAL O_2				6085.3
N_2 Tank 1	2965.6	2981.9	66.9	271.7
N_2 Tank 2	2990.8	2981.5	70.9	271.4
N_2 Tank 3	2888.4	2936.3	63.3	268.0
N_2 Tank 4	2949.4	2953.0	63.9	272.4
N_2 Tank 5	2953.4	2948.4	66.1	271.4
N_2 Tank 6	2908.9	2961.0	66.3	267.9
TOTAL N_2				1622.8

FIGURE 2.5-42 PRELAUNCH LOADING OF AIRLOCK MODULE O_2 AND N_2 TANKS

EVENT	GAS USAGE (LBS)		GAS REMAINING (LBS)	
	O_2	N_2	O_2	N_2
PRESSURIZATION TO 5 PSIA	244	--	5841	1623
N_2 PURGE CYCLES (4)	---	276	5841	1347
PRESSURIZATION TO 5 PSIA (SL-2)	249	31	5592	1316
SL-2 MISSION (29 DAYS)	364	112	5228	1204
PRESSURIZATION TO 5 PSIA (SL-3)	222	45	5006	1159
SL-3 MISSION (60 DAYS)	913	196	4093	963
PRESSURIZATION TO 5 PSIA (STORAGE)	---	145	4093	818
REPRESS TO 5 PSIA (STORAGE)	58	---	4035	818
PRESSURIZATION TO 5 PSIA (SL-4)	237	19	3798	799
SL-4 MISSION (84 DAYS)	1189	190	2609	609

FIGURE 2.5-43 O_2 AND N_2 CONSUMABLE USAGE SUMMARY

As discussed in Paragraph 2.4.4, temperatures on O_2 tank 6 and on N_2 tanks 1 and 2 were higher than expected. As a result, the 160°F design limit on O_2 tank no. 6 was exceeded while at a pressure of approximately 2450 psia. Also, the higher temperature on N_2 tanks coupled with the higher than normal preflight loading resulted in pressures up to 3370 psia. No problems were identified as a result of these exposures to off-nominal conditions. It should also be noted that the high N_2 tank pressures could have been prevented by utilizing gas from these tanks for normal system usage; however, flight controllers elected to retain all N_2 in these tanks for recharge of M509 Experiment pressure vessels.

D. Gas Supply - Performance of the 120 psi O_2 and 150 psi N_2 pressure regulators is shown in Figures 2.5-44 and 2.5-45 respectively. Regulation of 120 psi O_2 was normal in all respects, while regulation of 150 psi N_2 pressure was satisfactory although control level decreased with operating time as seen in Figure 2.5-46. This figure also indicates that normal N_2 regulation was restored when a regulator had been shut off for a number of days. Efforts to duplicate regulator performance in ground tests were unsuccessful and extensive investigations failed to pinpoint the exact cause of such behavior. The problem was believed to have been caused by excessive friction in the regulators. The N_2 regulators were, therefore, cycled on and off as required during SL-3 to maintain desired pressure levels. Regulators performed normally on SL-4 as shown in Figure 2.5-47.

- Initial Pressurization - Gas flowrates for cluster pressurization were nominal based on the typical cabin pressure profile plotted on Figure 2.5-48. The pressurization rates shown by this figure would indicate O_2 and N_2 fill flowrates at 21.42 lb/hr and 7.03 lb/hr, respectively.
- Mole Sieve Actuation - The regulated N_2 supply provided normal cycling of the molecular sieve beds. An apparent problem during SL-3 activation when molecular sieve beds failed to cycle has been attributed to failure of the crew to properly open the Mole Sieve A Bed Cycle N_2 supply valve on Panel 221.
- Reservoir Pressurization - Nominal performance was exhibited by the 5 psia N_2 regulators used for pressurization of water reservoirs in the

EVENT	FLOWRATE (LB/HR)	REG PRESSURE (PSIA)
O_2 FILL	21.42	120.5
TWO-GAS CONTROL	<1.0	125 - 127
EVA	22.7 (NOMINAL)	125.2
IVA	15.8 (NOMINAL)	125.2
LOCKUP	ZERO	135 - 147

FIGURE 2.5-44 GAS SYSTEM REGULATED O_2 PRESSURES

EVENT	FLOWRATE (LB/HR)	REG PRESSURE (PSIA)
N_2 FILL	7.03	153
TWO-GAS CONTROL	<1.0	153 - 159 △1
LOCKUP	ZERO	166 - 180

NOTES: △1 VALUES SHOWN ARE FOR PERIOD IMMEDIATELY FOLLOWING ACTIVATION. SEE FIGURES 2.5-46 AND 2.5-47 FOR OTHER TIME PERIODS

FIGURE 2.5-45 GAS SYSTEM REGULATED N_2 PRESSURES

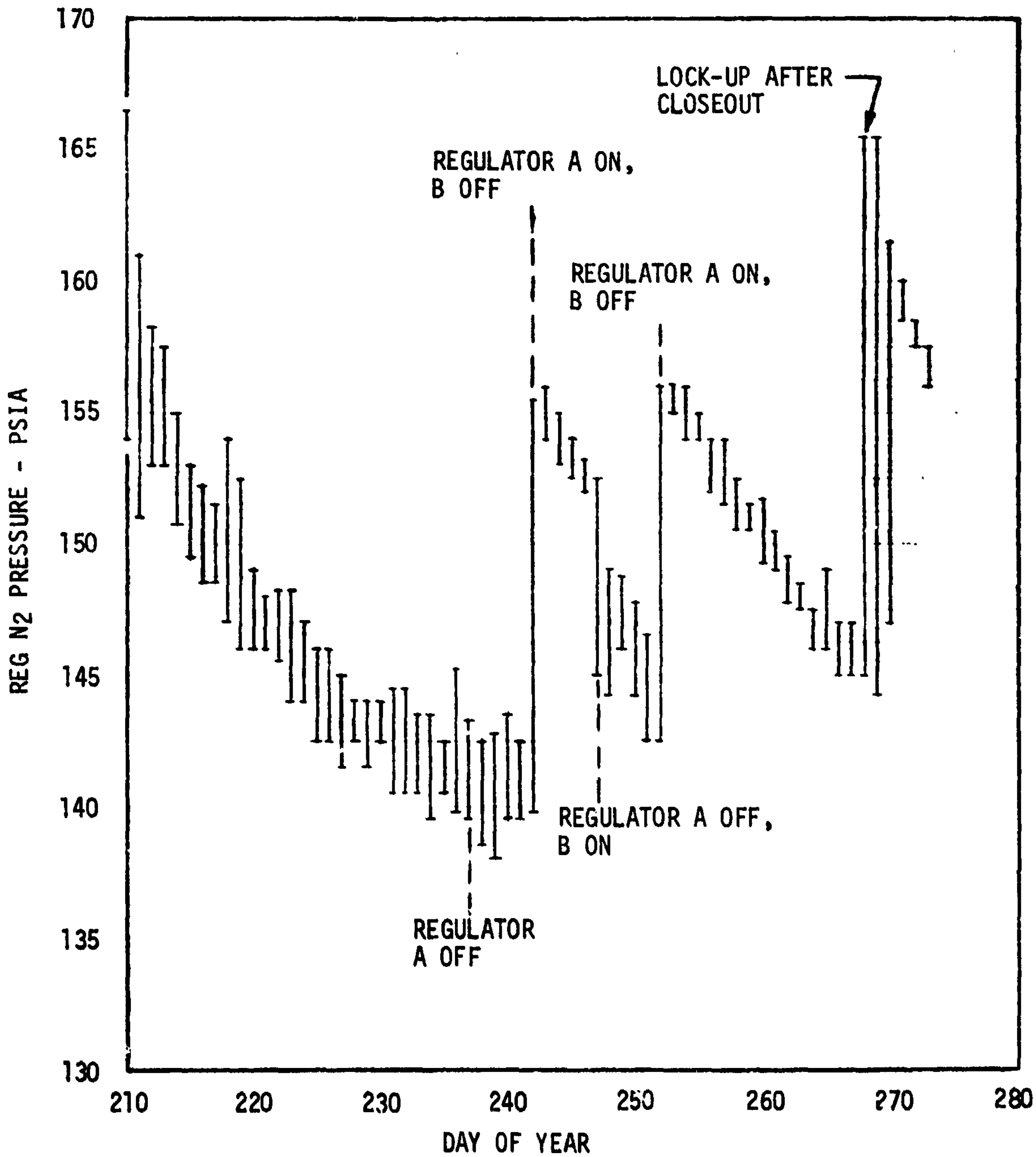


FIGURE 2.5-46 REGULATED N_2 PRESSURES DURING SL-3 (HIGH/LOW VALUES SHOWN)

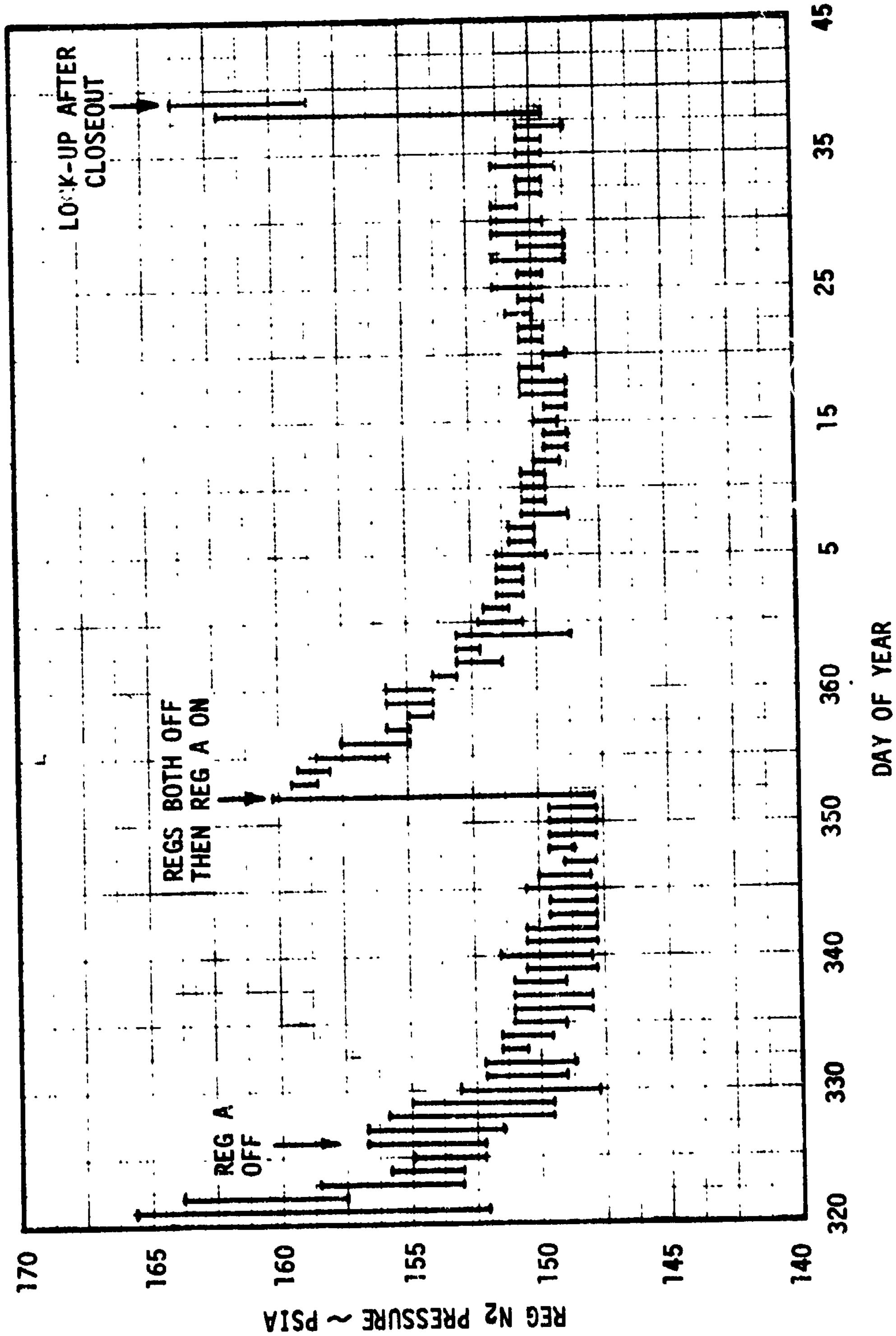


FIGURE 2.5-47 REGULATED N_2 PRESSURES DURING SL-4 (HIGH/LOW VALUES SHOWN)

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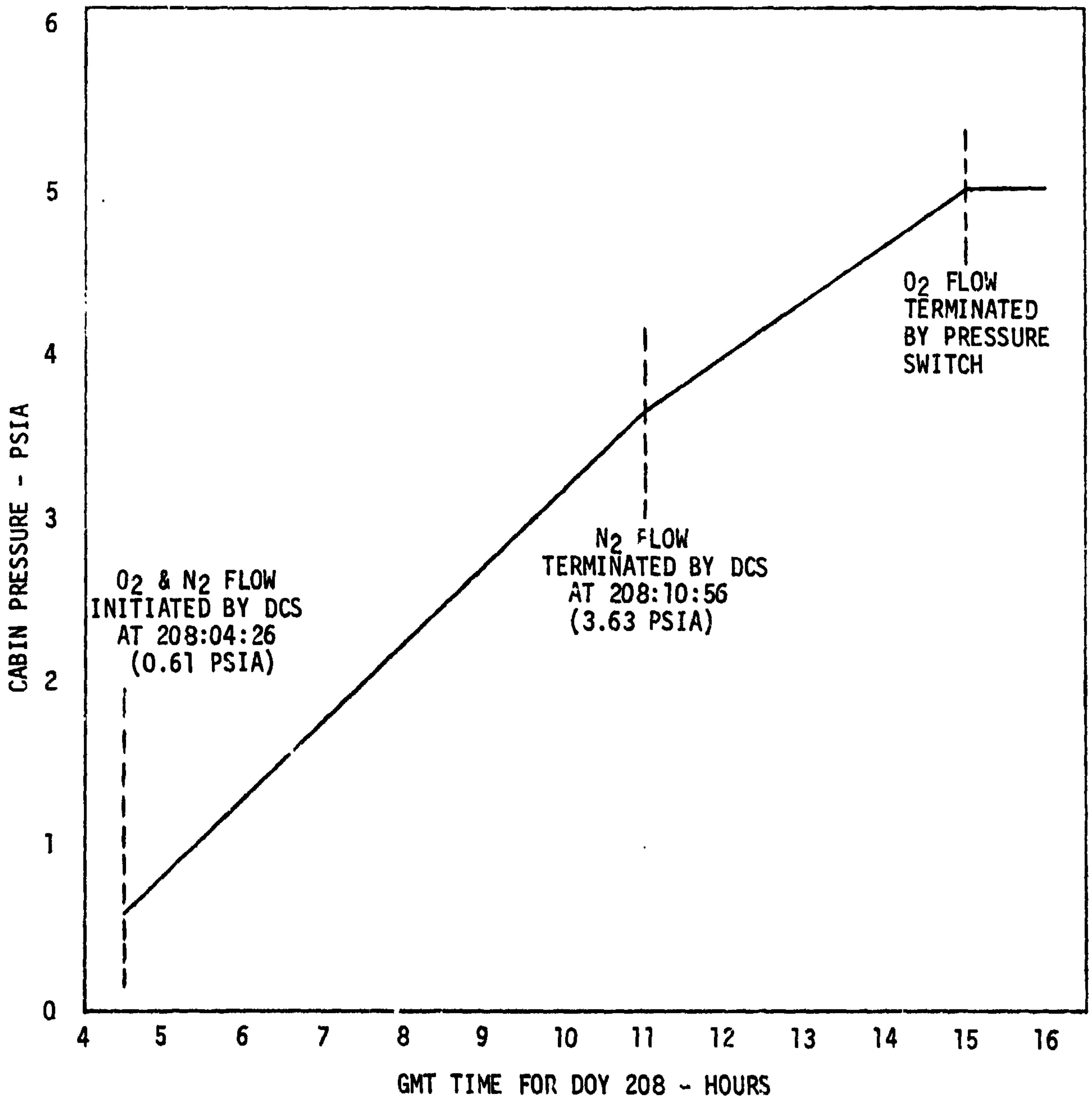


FIGURE 2.5-48 CLUSTER PRESSURIZATION PRIOR TO SL-3

ATM C&D Panel/EREP and Suit Cooling Systems. Following initial venting of the reservoirs during ascent, pressure was maintained between 4.95 and 6.0 psia throughout the mission with the higher level being attained during servicing of the Suit Cooling System 1 reservoir on DOY 361.

- EVA/IVA Support - EVA/IVA Support was normal as discussed in Section 2.6.
- Experiment and OWS Water System Support - The regulated N_2 supply provided normal OWS water system pressurization and experiment support. Recharge of M509 bottles was accomplished, without difficulty, on 48 separate occasions. The pressure in N_2 tanks 1 and 2, which had been reserved for tank top-off during this operation, was 3080 psia during the initial top-off on DOY 168 in SL-2 and 1900 psia following the final top-off on DOY 026 during SL-4. Tanks 1 and 2 were never opened to the remainder of the N_2 system.

E. Atmospheric Pressure Control

- Maintain Minimum O_2/N_2 Pressure - Control of cabin pressure was well within the 5.0 $\pm$ 0.2 psia allowable range and PPO_2 remained between 3.3 and 3.9 psia during all periods when operation of the two-gas control system was not overridden by gas addition and/or venting associated with performance of M509/T020, EVA operations, CM O_2 flow into the cluster or inadvertent crew action. Figure 2.5-49 presents PPO_2 and cabin pressure levels during a typical period when the above influences were not present and the two-gas control system was permitted to operate normally. Operation of PPO_2 sensors was completely normal with no degradation being observed during any of the manned missions based upon comparisons of outputs (control and monitoring) with M171 mass spectrometer readings. The capability to flow O_2 and N_2 for PPO_2 sensor calibration was successfully demonstrated on DOY 330.

On several occasions, the fact that flow capability of cabin pressure regulators was intentionally designed to a close limit permitted rapid recognition of abnormal cabin leakage when a decreasing cabin pressure was observed. For example, on DOY 211, a pressure decrease equivalent to an observed cluster leakage rate of approximately 4 lb/hr prompted a crew investigation which located an improperly positioned valve on the Trash Airlock in the OWS. Similarly, on DOY 257, a cabin pressure decrease of

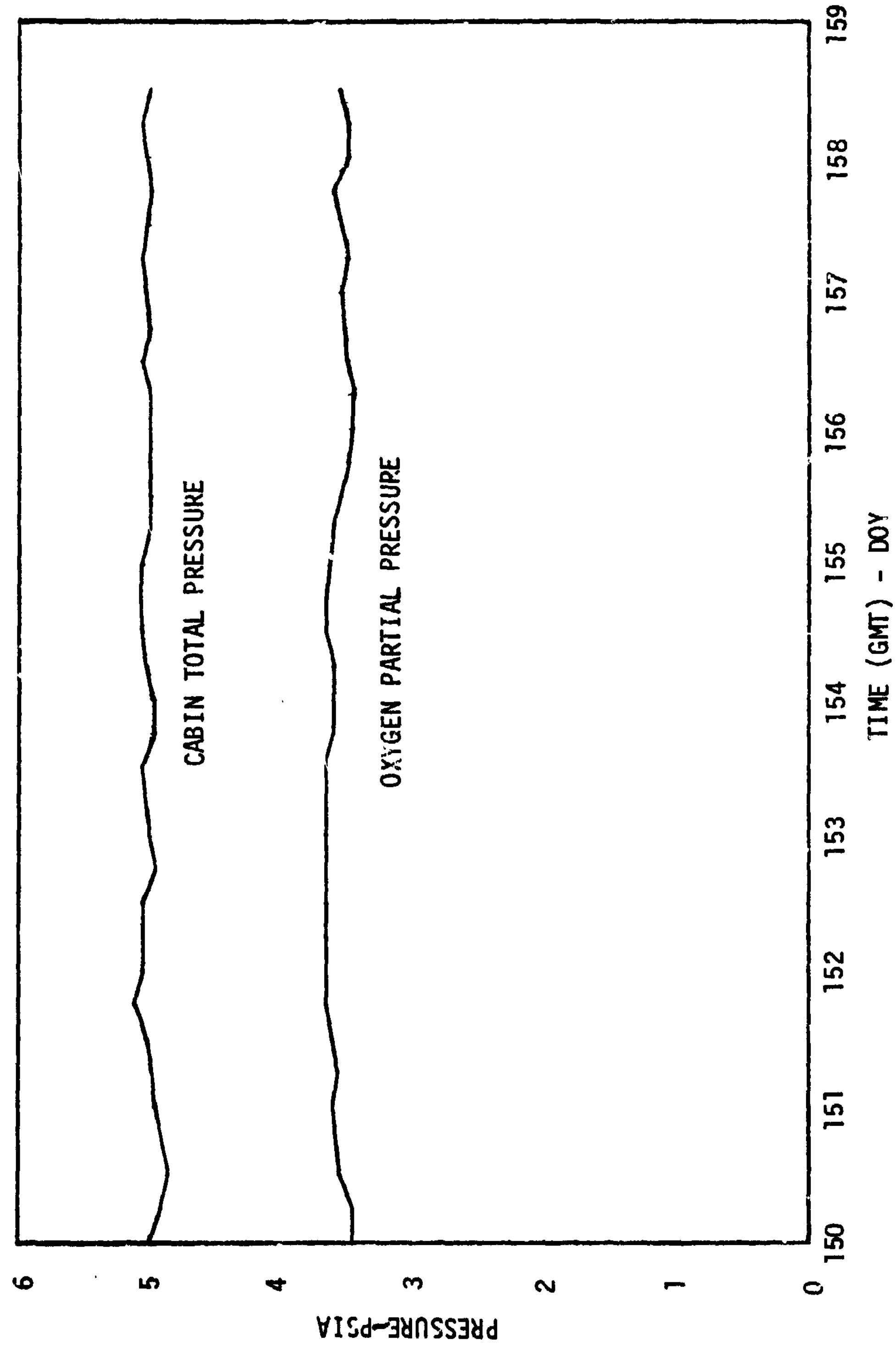


FIGURE 2.5-49 CABIN TOTAL AND OXYGEN PARTIAL PRESSURE CONTROL

approximately 0.1 psia over a six-hour period was found to be the result of an improperly sealed door on waste processors in the OWS. With a higher maximum flow rate regulator, identification of such leakage would not have been possible until the crew used these components again or until excessive gas usage from storage tanks was noted. In either case, a number of days may have elapsed and the overboard gas loss would have been significantly greater.

- Limit Maximum Atmospheric Pressure - Gas flow from the cluster atmosphere via cabin pressure relief valves occurred on two occasions during SL-4 due to a procedural error. Cabin pressure normally remained well below the relief valve minimum cracking pressure of 5.5 psid except during M509/T020 operations when pressure was permitted to increase to higher levels (5.95 psid maximum) after the manual shutoff valve on Airlock Module relief valves had been closed. However, the crew failed to close the aft compartment relief valve on SL-4 prior to M509 operations and overboard relief occurred as cabin pressure reached 5.7 psia on DOY 017 and 020. Protection against overpressurization of the cluster during periods when Airlock Module relief valves were closed was afforded by CM relief valves.

2.5.4.2 Atmospheric Control System

In addition to performing all of the required functions, the system provided more cooling than the maximum planned to assist cool down of the workshop crew quarters. No midmission molecular sieve bakeouts were required per the bakeout criteria (paragraph 2.5.2.2,B.) during the entire flight and none were performed during the 84 day SL-4 mission.

A. Humidity Control - Figures 2.5.50 through 2.5-52 present dewpoint histories for each of the missions and indicate an overall dewpoint range of 39.8°F to 63.5°F when activation periods are excluded. Normally, dewpoint was maintained between 46°F and 60°F except during EVA periods and on SL-4 crew shower days. During EVA, dewpoint decreased due to reduced moisture generation rates (only one crewman internal to the vehicle) and the smaller volume of atmosphere being conditioned. Dewpoint quickly returned to normal levels following EVA when the OWS hatch was again open and circulation with the larger volume re-established. On crew shower days, additional moisture was introduced into the cluster atmosphere and resulted in dewpoints exceeding 60°F on several occasions

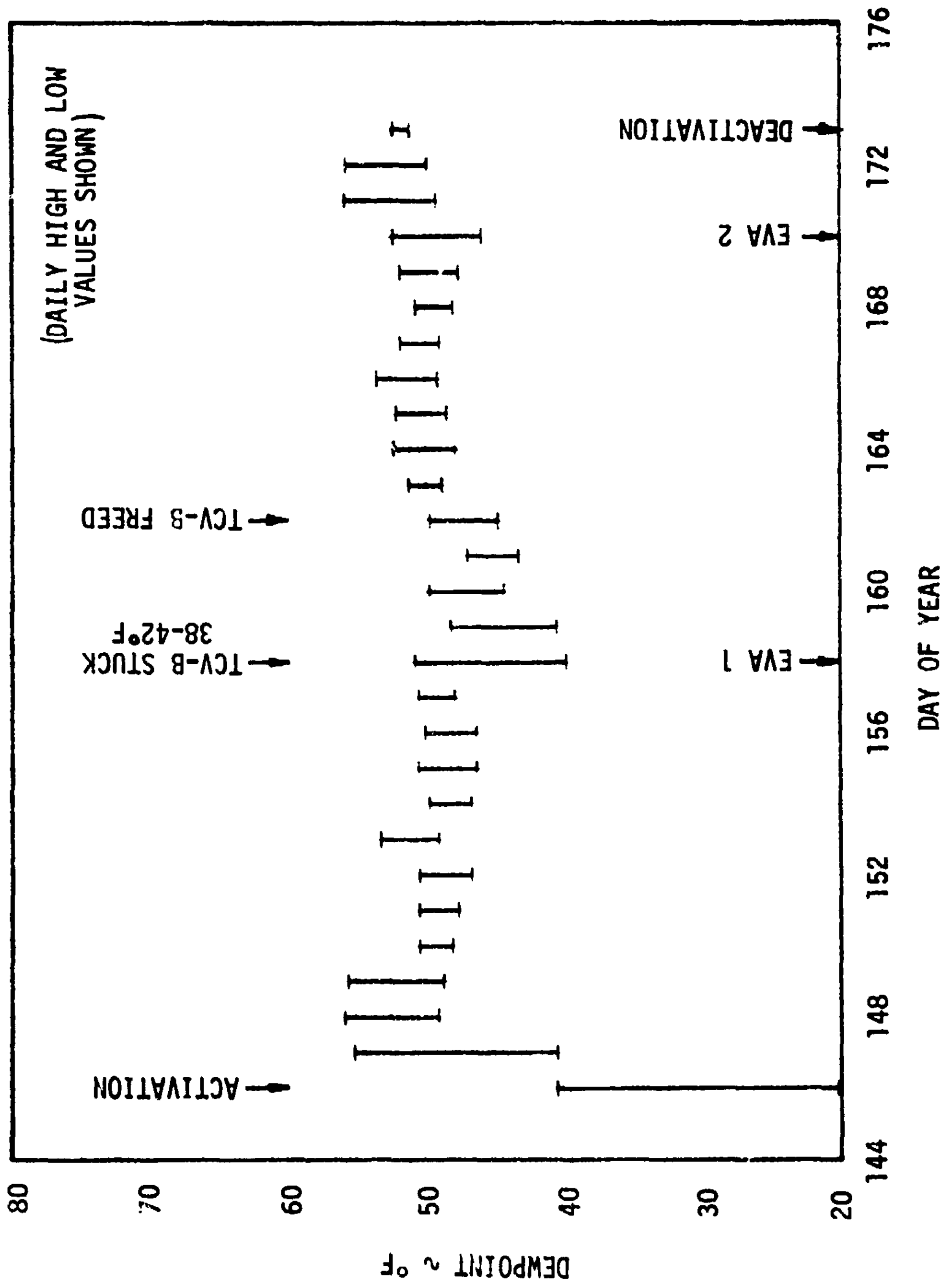


FIGURE 2.5-50 SL-2 DEWPOINT HISTORY

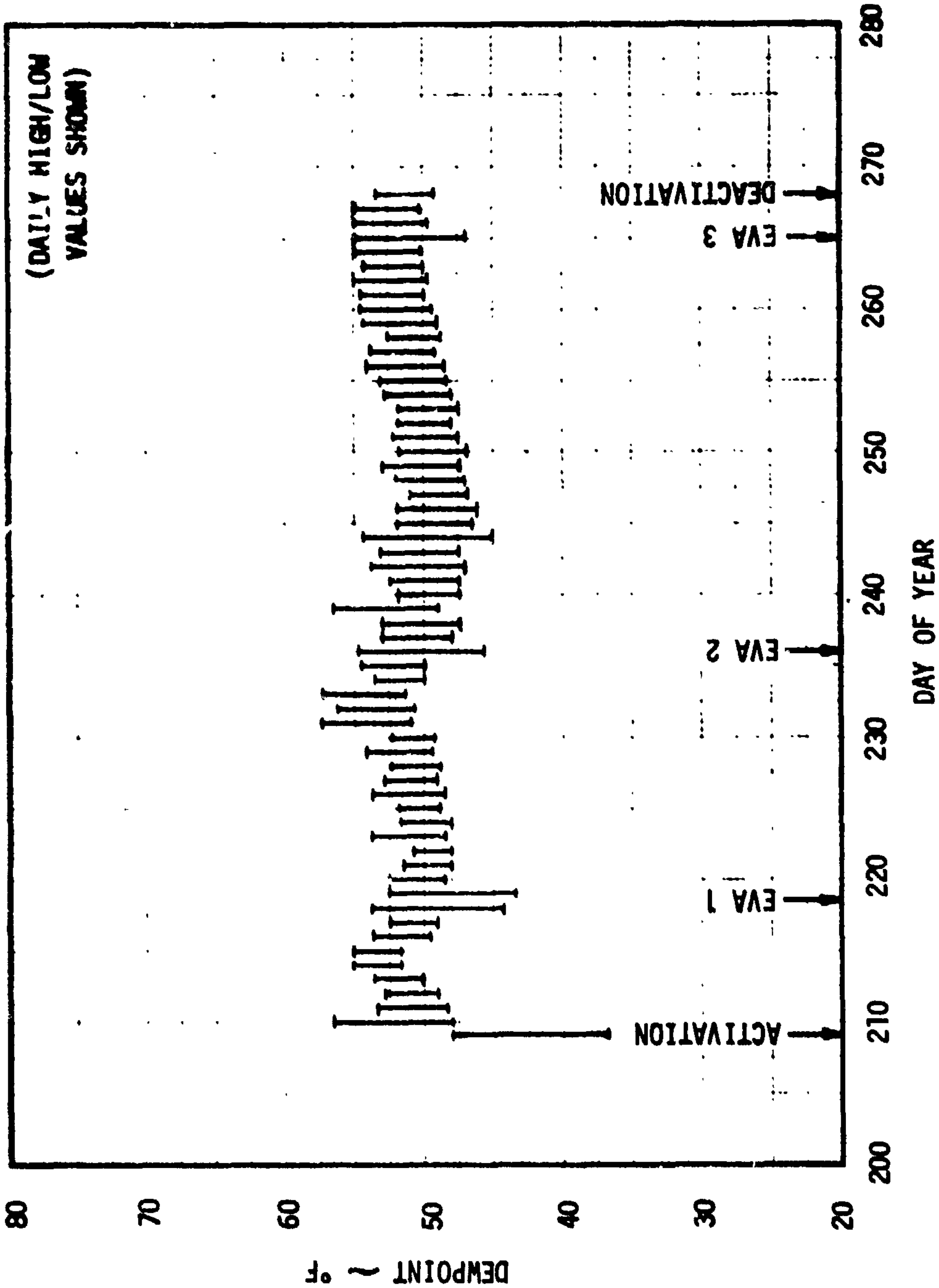


FIGURE 2.5-51 SL-3 DEWPOINT HISTORY

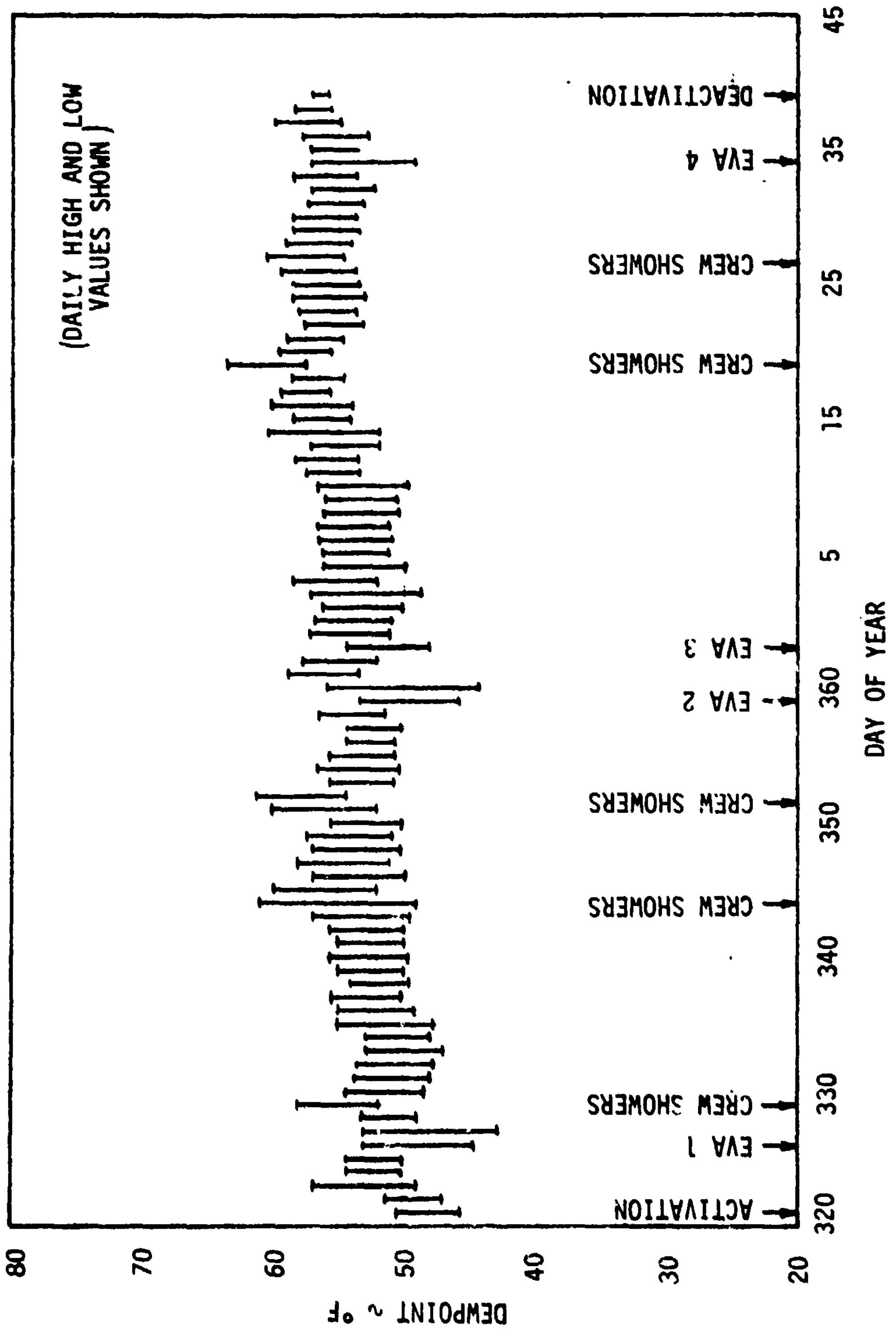


FIGURE 2.5–52 SL–4 DEWPOINT HISTORY

during SL-4. This is believed to be due to water spills discussed in the SL-4 debriefing report.

The original water separator plate assemblies were used throughout the entire mission. Liquid/gas separation was accomplished in a normal manner.

B. Purity Control - Molecular sieves performed in an outstanding manner as indicated by atmospheric CO_2 levels shown in Figure 2.5-53. It can be seen that average daily $PPCO_2$ was normally well below 5.5 mmHg. Somewhat higher levels were obtained during and following the SL-3 mid-mission bakeout period (DOY 231 to 234). Significantly lower levels indicated on EVA days were a result of reduced CO_2 generation rates (only one crewman internal to the vehicle) and the smaller volume of atmosphere being conditioned with the lock compartment forward hatch closed. Mole sieve performance indicated by the difference between measured inlet and outlet carbon dioxide partial pressures is shown in Figure 2.5-54. Three $PPCO_2$ measurements from Experiment M171 taken during the same time span were in close agreement with the values measured at the mole sieve inlets. Effects of crew activity on atmospheric $PPCO_2$ is apparent by the cycles of $PPCO_2$ levels during each 24 hour period. The $PPCO_2$ level decreased during crew sleep periods and increased during periods of crew activity.

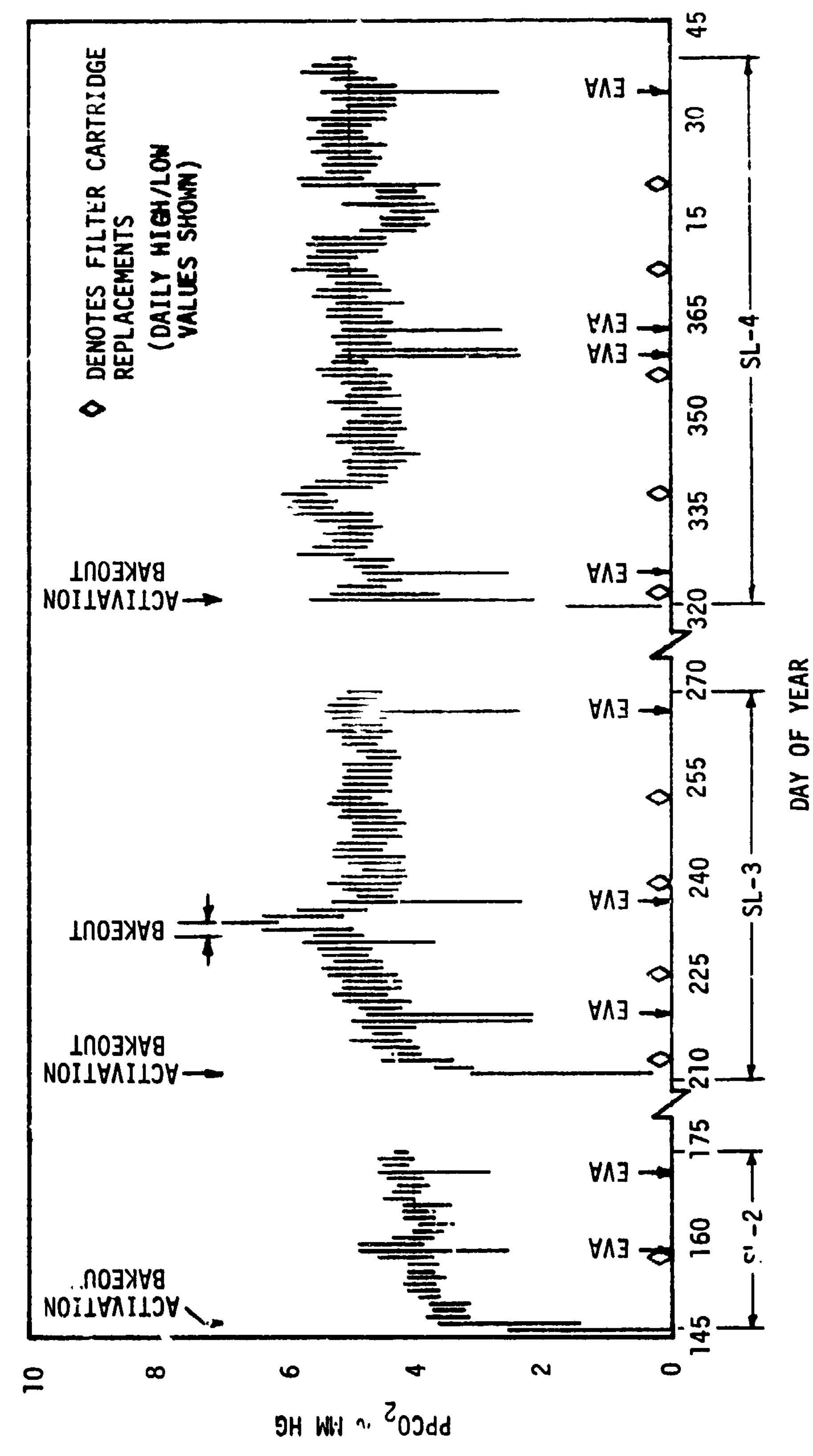


FIGURE 2.5-53 MOLECULAR SIEVE A INLET CO_2 PARTIAL PRESSURE

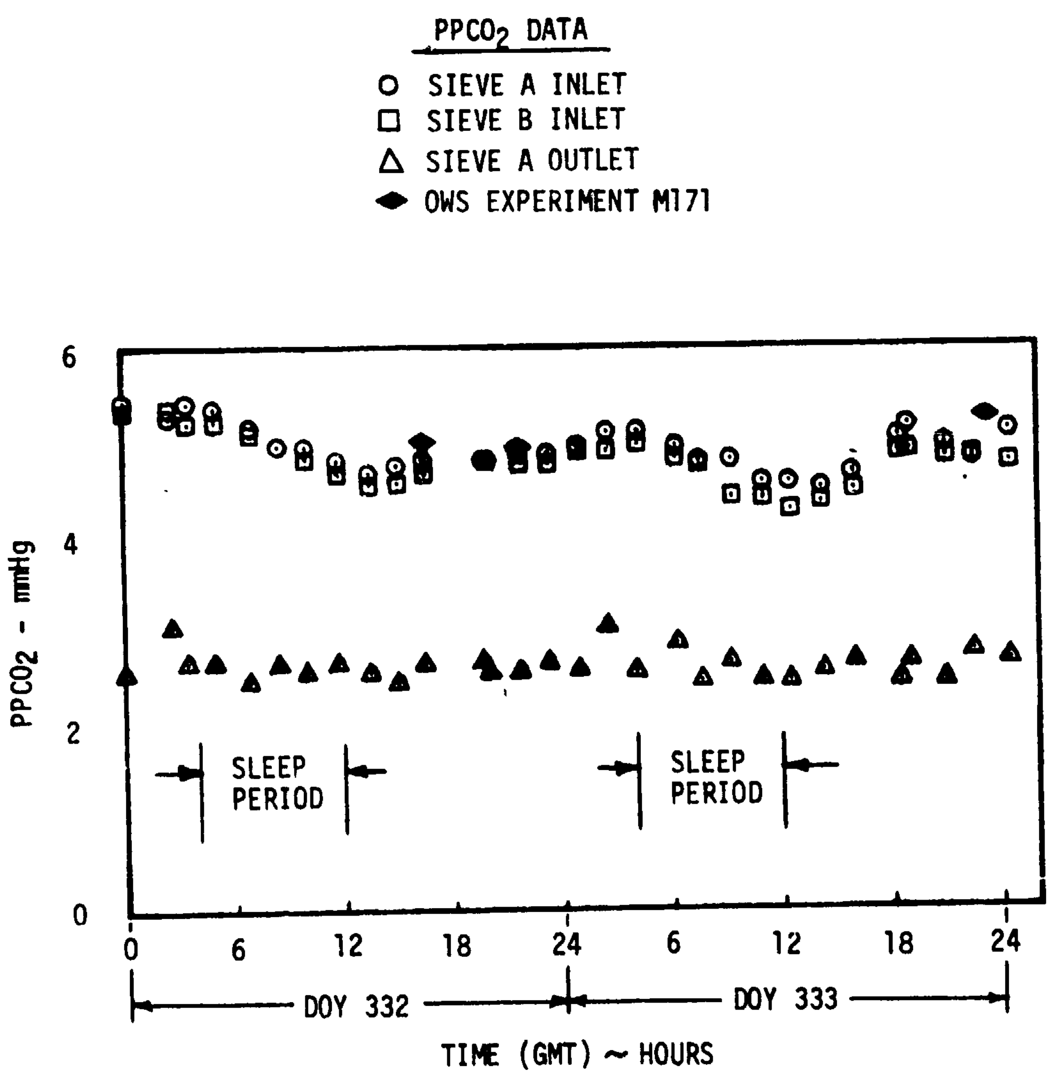


FIGURE 2.5-54 MOLECULAR SIEVE PERFORMANCE

Bed cycling was normal except for a 3 1/2 hour period following molecular sieve activation on SL-3 when beds failed to cycle due to insufficient N_2 flow as discussed in Section 2.5.4.2. This situation coupled with the delay in initiation of condensate removal subjected sieve beds to higher than normal moisture levels; however, beds quickly recovered and no performance degradation requiring premature bed bakeout was observed.

Figure 2.5-55 provides a summary of all bed bakeouts which were performed. Based on crew reports, they were normal in all respects. No hardware failures of any type were experienced on Mole Sieve A thereby negating any requirement for activation of Mole Sieve B. Although no midmission bakeouts were required during the entire flight, one was performed during the 59 day SL-3 mission.

Odor removal by charcoal canisters apparently was good since both the SL-2 and SL-4 crews commented on "freshness" of the atmosphere and all crew reports indicate no problems with undesirable odors in the cluster. Charcoal canisters were replaced on DOY 172, 245, 267 and 364.

MISSION	BED DESIGNATION	BAKEOUT DOY	BAKEOUT DURATION (HOURS:MINUTES)
SL-2	SIEVE A BED 1	146-147	6:35
SL-2	SIEVE A BED 2	147	5:07
SL-3	SIEVE A BED 1	210	10:45
SL-3	SIEVE A BED 2	210	6:40
SL-3	SIEVE A BED 1	231-232	13:00
SL-3	SIEVE A BED 2	232-233	13:20
SL-4	SIEVE A BED 1	321	6:00
SL-4	SIEVE A BED 2	321-322	5:45

FIGURE 2.5-55 SUMMARY OF MOLECULAR SIEVE BED BAKEOUTS DURING FLIGHT

Measurement of $PPCO_2$ levels proved to be adequate. Sieve A inlet and outlet $PPCO_2$ detectors provided data within their allowable range of accuracy (±1.4 mm Hg) while SL-2 and SL-3 data from Sieve B detectors were outside this range on the low side based on preflight predictions and M171 experiment data. Readings from the Sieve A outlet detector were believed to be correct while data from Sieve A and B inlet detectors were approximately 1 and 2-3 mm Hg lower than actual levels, respectively, during the SL-2 mission. The Sieve B outlet detector was only active for a short period of time on SL-2 when an additional check on monitoring capability was desired. During this period, outputs were initially about 2 mm Hg lower than actual before drifting to a much higher level when filter cartridges became saturated with water. Postflight testing of SL-2 inlet filter cartridges revealed no cartridge problem. Zero checks performed on SL-3 indicated that zero shift in detector electronics could not account for the low readings. Erratic outputs from the Sieve B inlet detector on SL-3 have been attributed to crew problems associated with improper replacement of filter cartridges. With all seals properly installed, outputs returned to normal levels. Following installation of a new o-ring on SL-4 at the Sieve B inlet detector, Sieve A and B inlet readings were in close agreement with M171 Experiment data.

C. Ventilation Control - Gas circulation provided by Airlock Module systems permitted normal accomplishment of all atmospheric control functions including dew point control, CO_2 removal, odor removal, and atmospheric cooling. A summary of fan flowrates is presented in Figure 2.5-56.

FAN(S)	FLOWRATE (CFM)			
	PREDICTED	INDICATED		
		SL-2	SL-3	SL-4
MOLE SIEVE A	34.2	42-55	35-52	35-48
MOLE SIEVE B	29.3	38-54	32-50	31-47
INTERCHANGE DUCT	112	89-165	58-147	41-103
AFT COMPT CABIN HX (4)	158	125-272	88-200	65-224
STS CABIN HX (3)	167	← NOT INSTRUMENTED →		

FIGURE 2.5-56 AIRLOCK MODULE FAN PERFORMANCE

Although a portion of the spread in flowrates is due to fluctuations in the output of flow sensors, a degradation in indicated flowrate was observed in the interchange duct and aft compartment cabin heat exchanger module as shown in Figures 2.5-57 and 2.5-58 respectively. Aft compartment cabin heat exchanger flow returned to normal levels after the inlet face of each heat exchanger was cleaned. Thereafter, flow was maintained at desired levels by periodic cleaning of heat exchanger inlets. Also, all four heat exchanger fans were replaced on DOY 018 in an attempt to obtain higher flow and increased cooling of the OWS during the period of high beta angles. However, no increase in flow was observed. A significant decrease in flow was obtained on DOY 019 when water was discovered in the heat exchangers. This condition occurred due to a high dewpoint level following crew showers and low coolant inlet temperatures with three coolant pump operation. Water removal was accomplished by vacuuming the heat exchangers and flow again returned to normal levels.

Cleaning of screens and installation of an alternate fan by the crew failed to improve the indicated interchange duct flowrate. It is likely that gas flow was normal and the low flow indication was a result of contamination on the flow sensor. It should be noted that interchange duct flow at the indicated levels would have caused no circulation problem.

The molecular sieve compressor actual flowrates were believed to be relatively constant and closer to the predicted values shown in Figure 2.5-56 than the indicated values. The inverter output frequency was set and matched with compressor performance during preflight altitude chamber tests to provide flowrates near the predicted values listed. The flowrate indications in flight as well as during preflight fluctuated on the high side of actual.

Although fan flow through STS cabin heat exchangers was not measured, adequate cooling was available at all times. Crew reports indicated that all fans were extremely quiet.

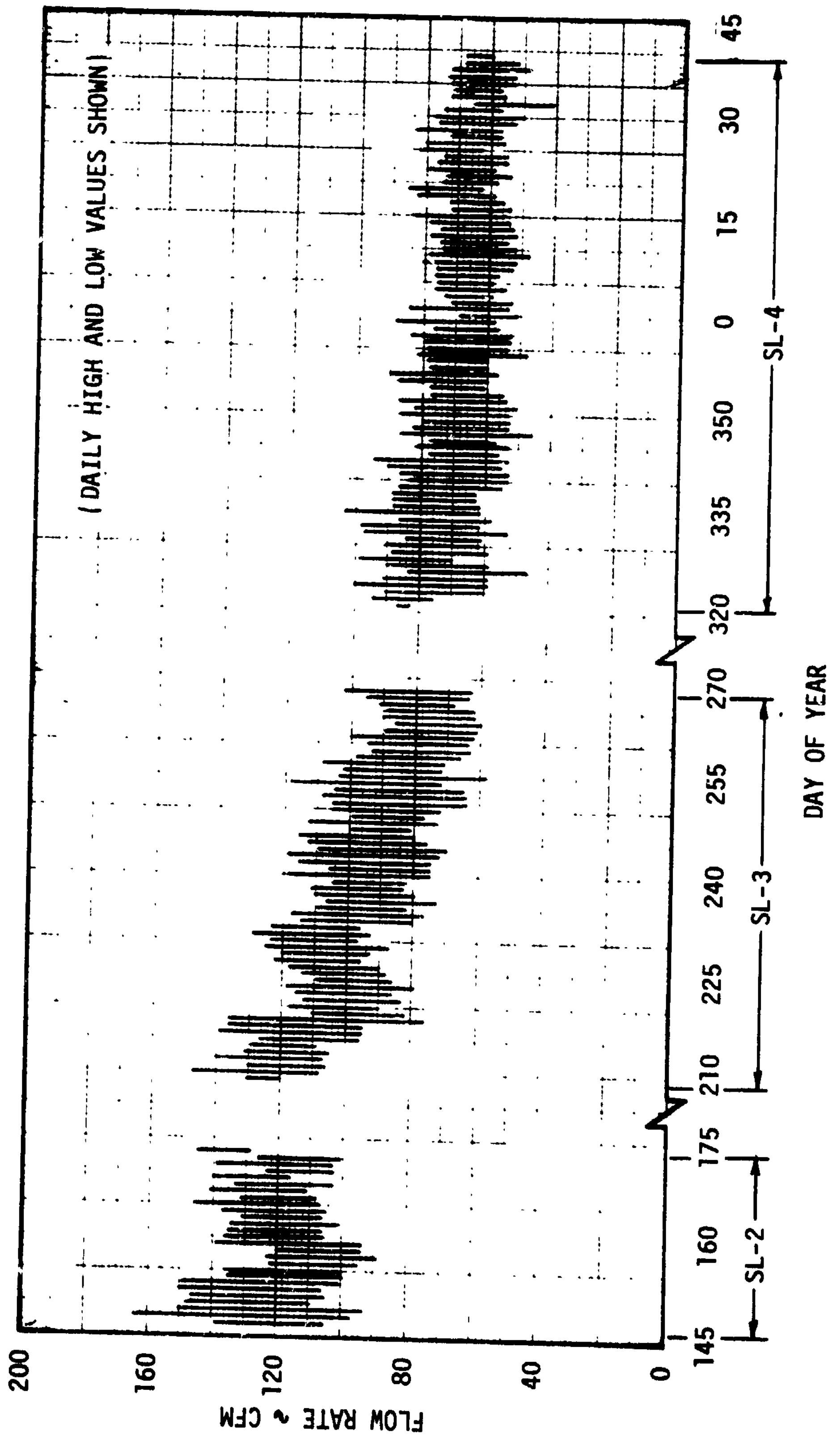


FIGURE 2.5-57 INTERCHANGE DUCT FAN FLOW RATE

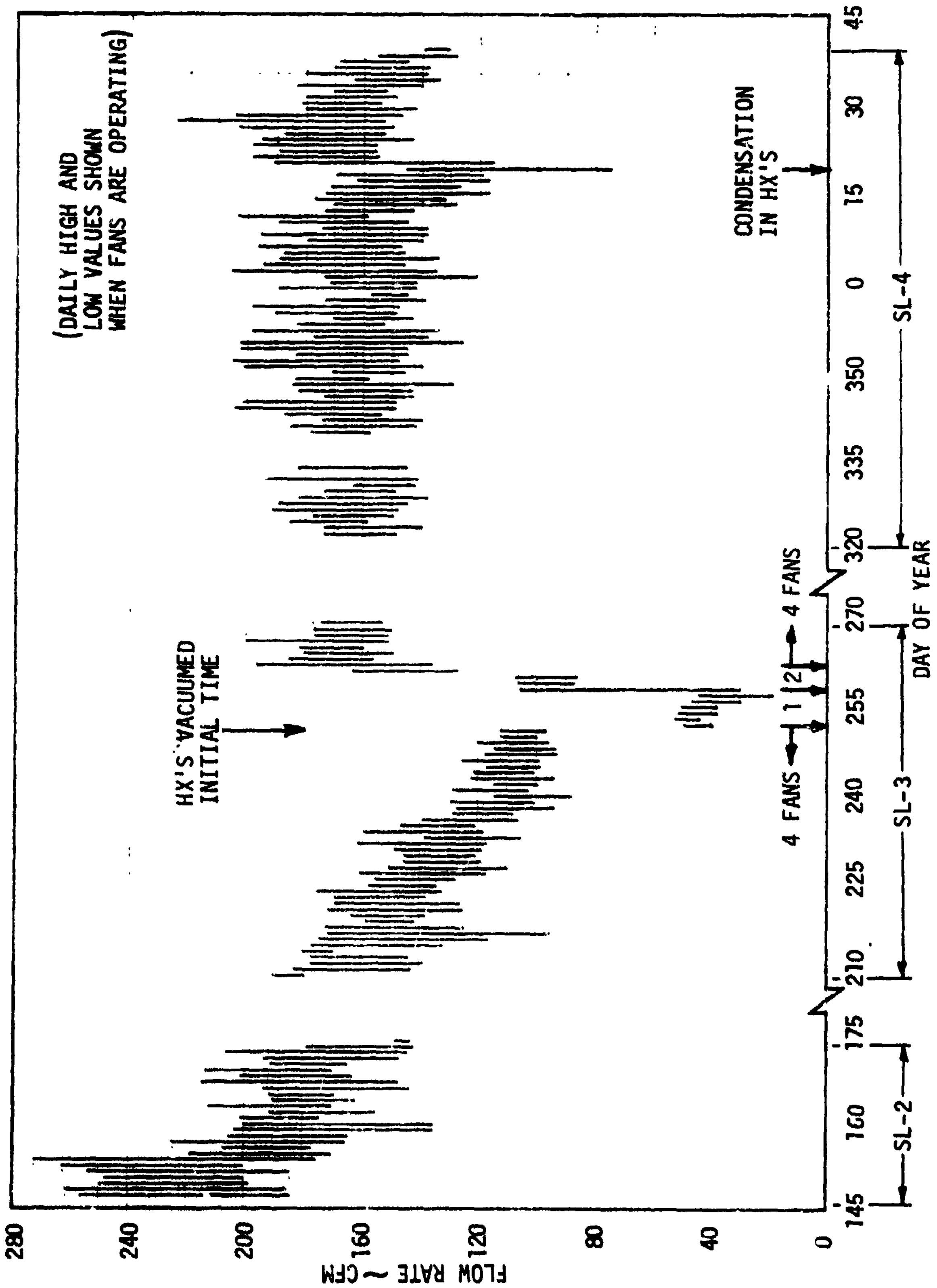


FIGURE 2.5-58 AFT COMPARTMENT CABIN HEAT EXCHANGER FAN FLOW RATE

The only failure associated with atmospheric circulation involved the molecular sieve B secondary fan which failed to start during SL-3 activation. The problem was believed to be with the inverter or electrical circuitry since installation of an alternate fan failed to correct the situation.

D. Temperature Control - Levels of atmospheric cooling provided during manned mission phases varied considerably depending on heat loads and resulting equilibrium temperature levels. Figure 2.5-59 presents atmospheric cooling accomplished by condensing heat exchangers, OWS (aft compartment) cabin heat exchangers, and STS cabin heat exchangers for the three missions. Higher than expected levels of atmospheric cooling were imposed upon Airlock Module systems due to high OWS temperature levels resulting from loss of the OWS meteoroid shield. Every effort was made to maximize AM cooling provided to the OWS. Three coolant pumps were operated for a short time on SL-4 to reduce temperature at the heat exchangers and a portable fan was periodically installed at the OWS interface with the aft compartment to increase circulation between the two modules. All increased demands were met by AM systems.

2.5.4.3 Condensate System

Satisfactory condensate removal, stowage and disposal was maintained throughout all manned mission phases as evidenced by the acceptable dewpoint levels shown in Figure 2.5-50 through 2.5-52. However, excessive gas leakage into the system resulted in the need for more frequent dumping of the system.

A. Removal and Stowage - Figure 2.5-60 presents condensate system ΔP following initial activation of SL-2. After the OWS condensate holding tank had been connected to the system a ΔP level of 3.5 to 4.5 psid was maintained at all times except during those periods when the holding tank was disconnected for EVA or a holding tank dump. With the holding tank disconnected, system ΔP decreased more rapidly than expected as seen in Figure 2.5-61 and 2.5-62. Since system pressure was not affected while connected to the holding tank, it was concluded that gas leakage into the gas side of the AM condensate tank was responsible. The spare condensate module was not installed, however, since EVA and holding tank dumps were performed infrequently and were of short duration. The system was deactivated normally at the end of SL-2.

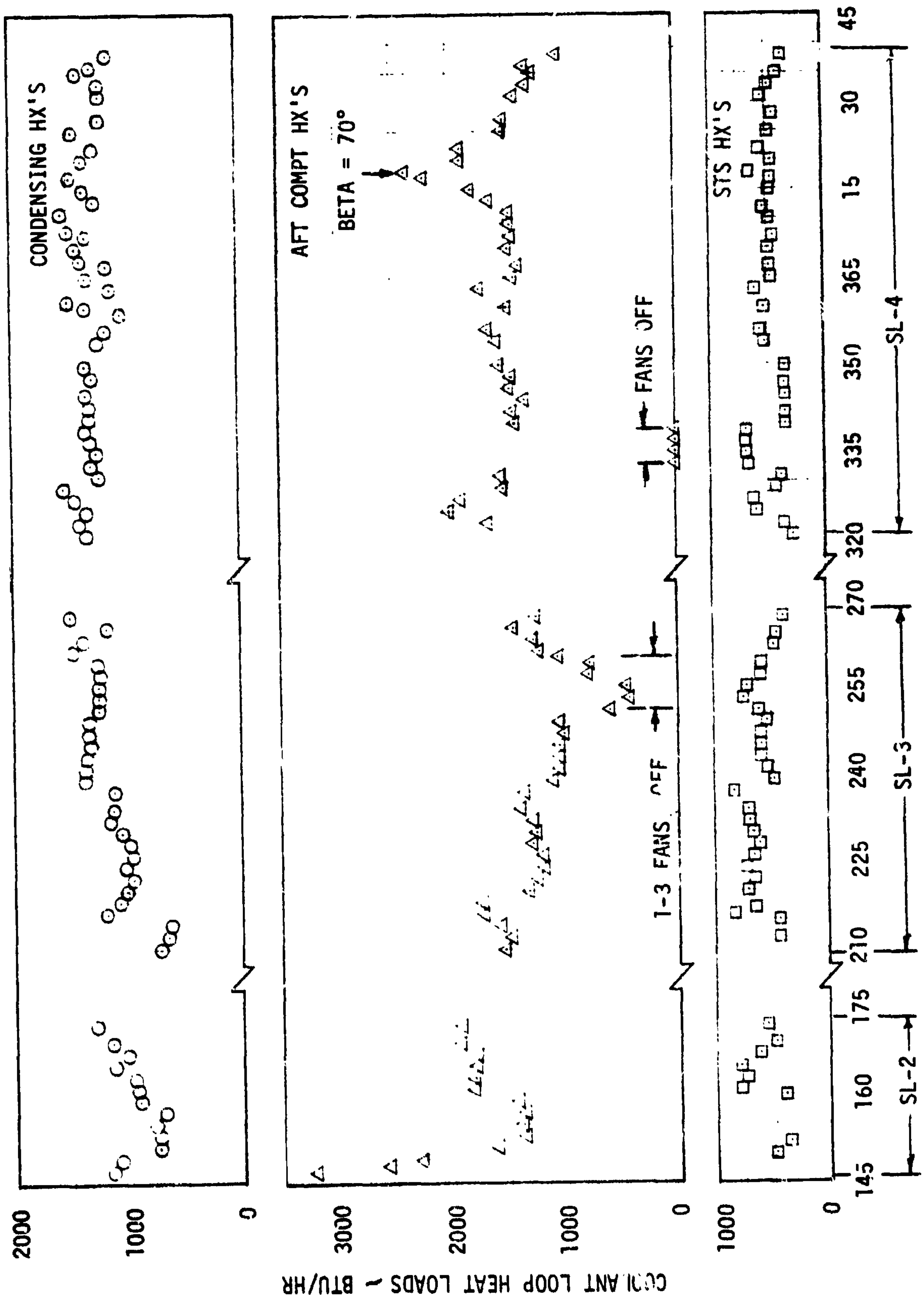


FIGURE 2.5-59 HEAT REMOVAL FROM CABIN ATMOSPHERE

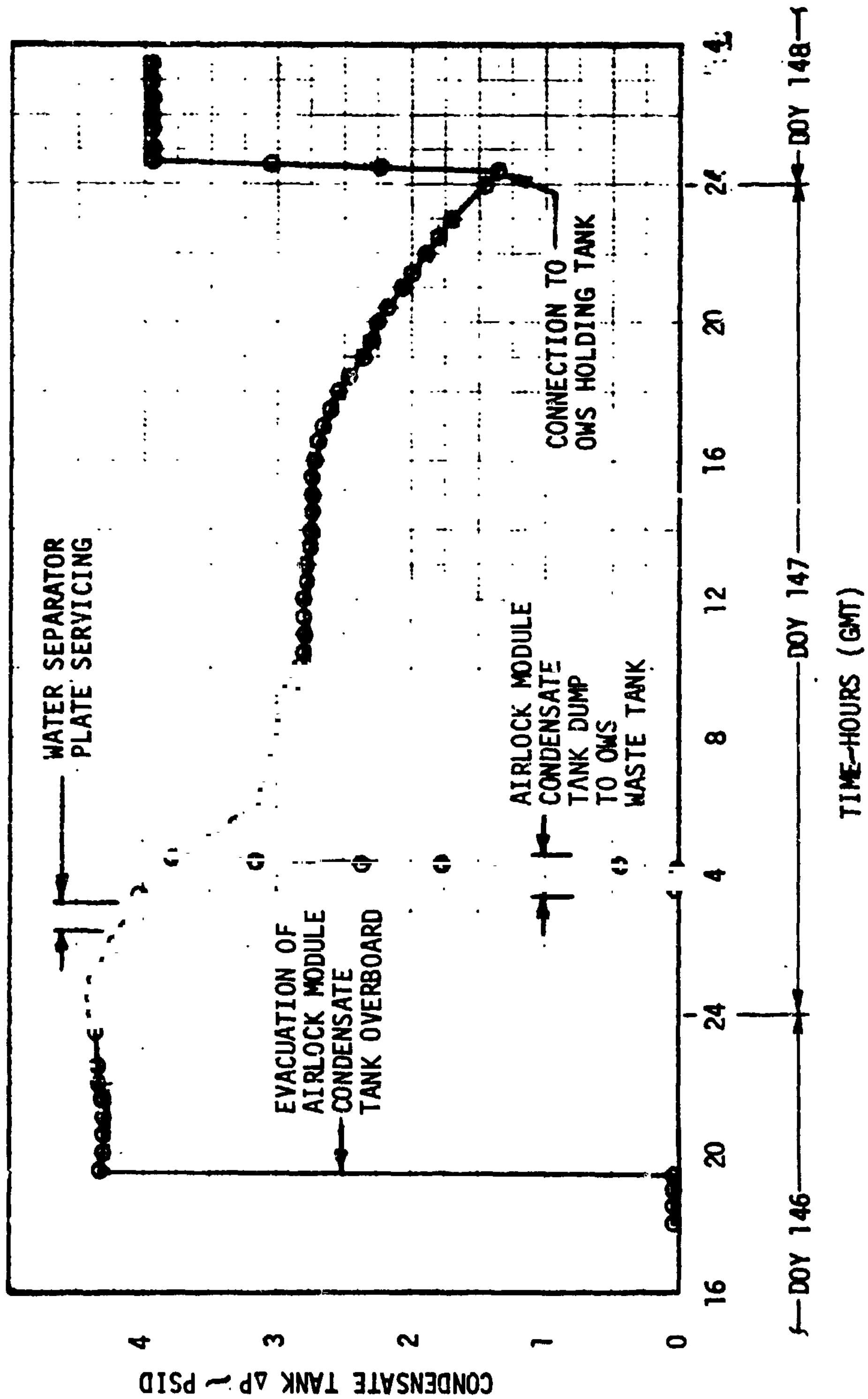


FIGURE 2.5-60 SL-2 CONDENSATE SYSTEM ACTIVATION

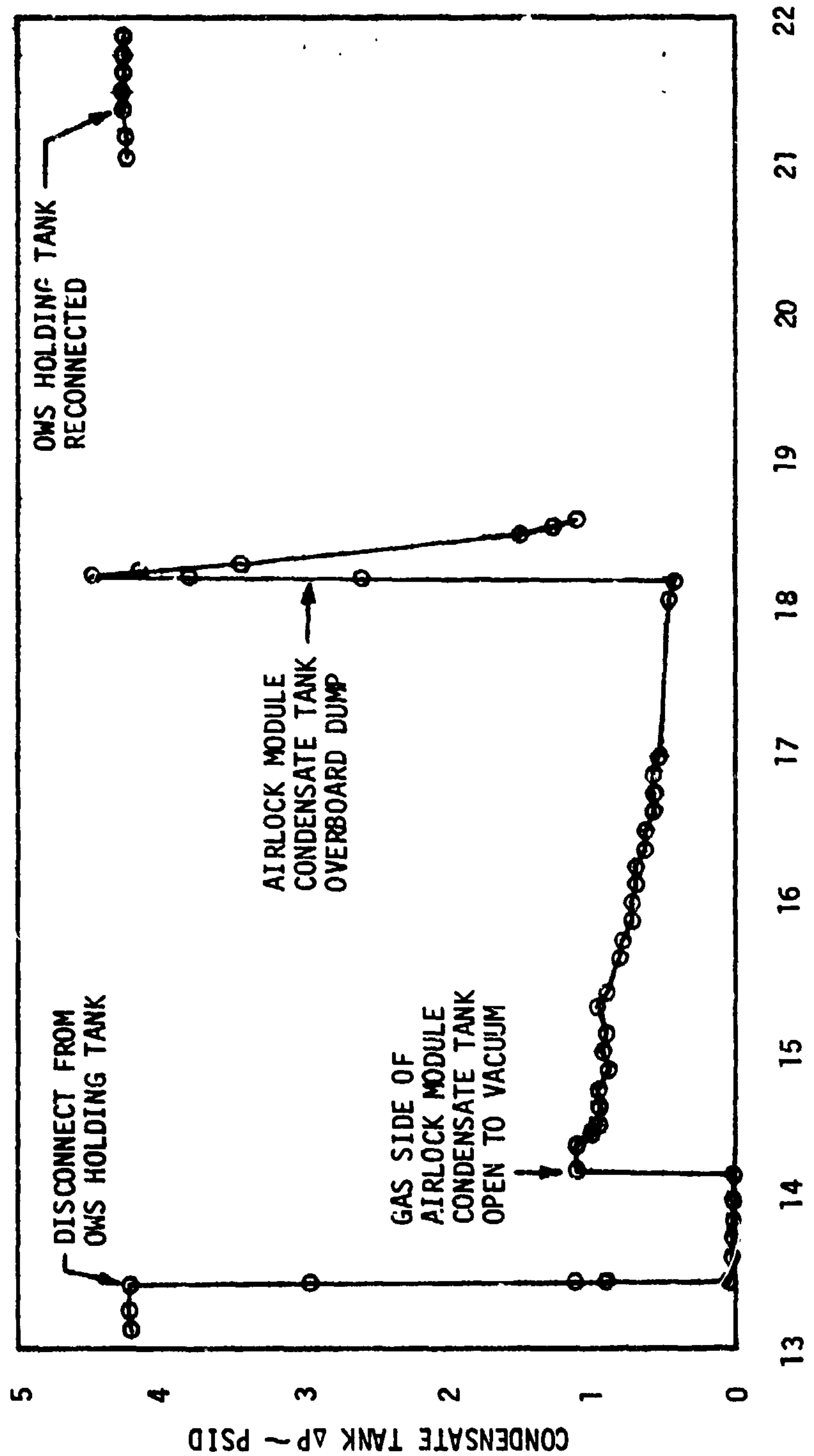


FIGURE 2.5-61 CONDENSATE SYSTEM PRESSURE DURING EVA ON DOY 158

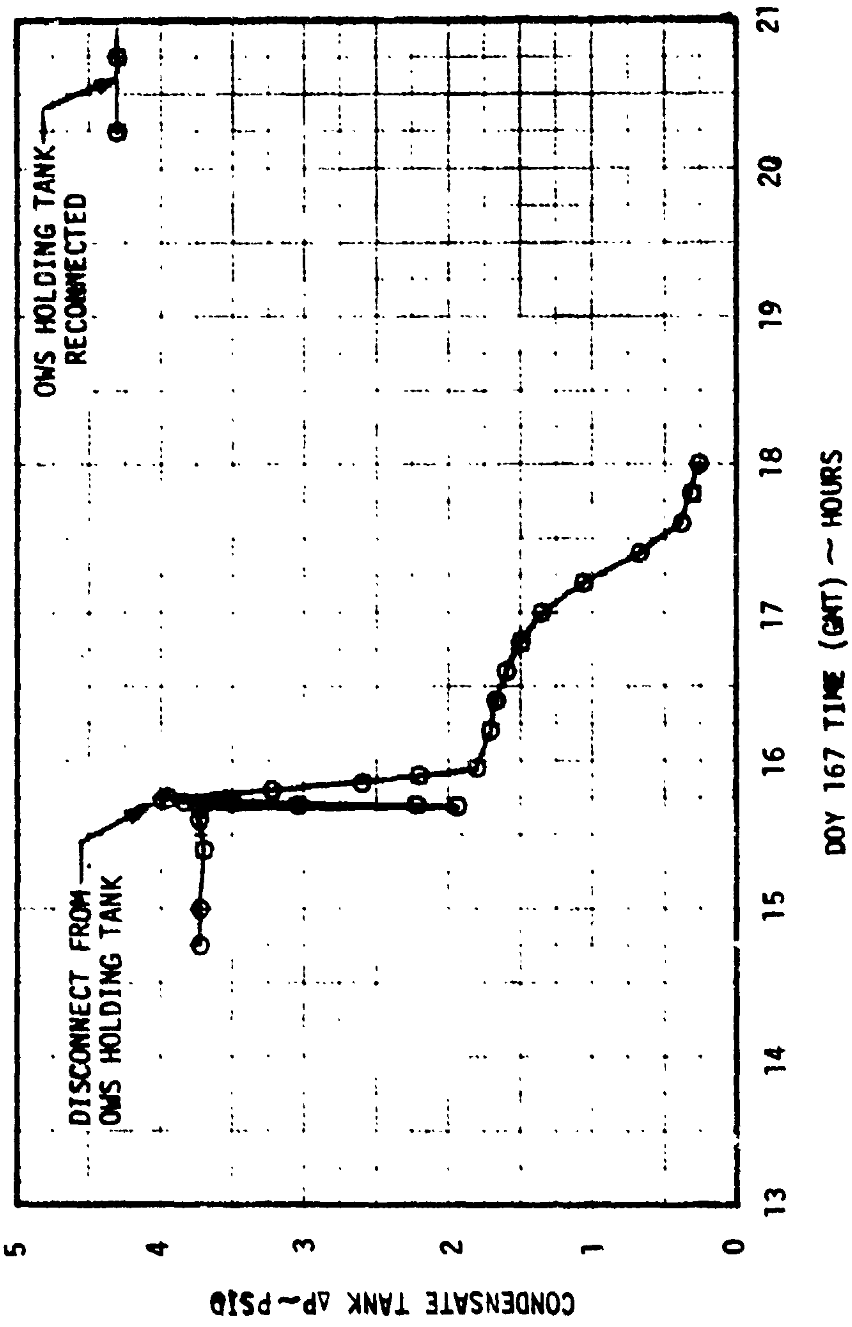


FIGURE 2.5-62 CONDENSATE SYSTEM PRESSURE DURING OWS HOLDING TANK DUMP

During SL-3 activation, the holding tank was reconnected to the system and the ΔP of 4.23 psid initially remained constant indicating a leak-free system. However, following use of the system for water separator plate servicing and transfer of CM waste water to the holding tank, ΔP had decreased to 3.6 psid and began a steady decline as shown in Figure 2.5-63. Although troubleshooting was performed and the gas leakage was isolated to plumbing within the Airlock Module, the exact location of the leak could not be established. Further evaluation led to the belief that leakage was occurring in one or more quick disconnects. As a result, procedures for lubrication of quick disconnects were developed and incorporated into crew malfunction procedures. Leakage disappeared on DOY 245 following disconnection of the dump QD from the condensate module. No further evidence of leakage was observed throughout the remainder of SL-3 and system deactivation was limited to closing the condensing heat exchanger condensate isolation valves.

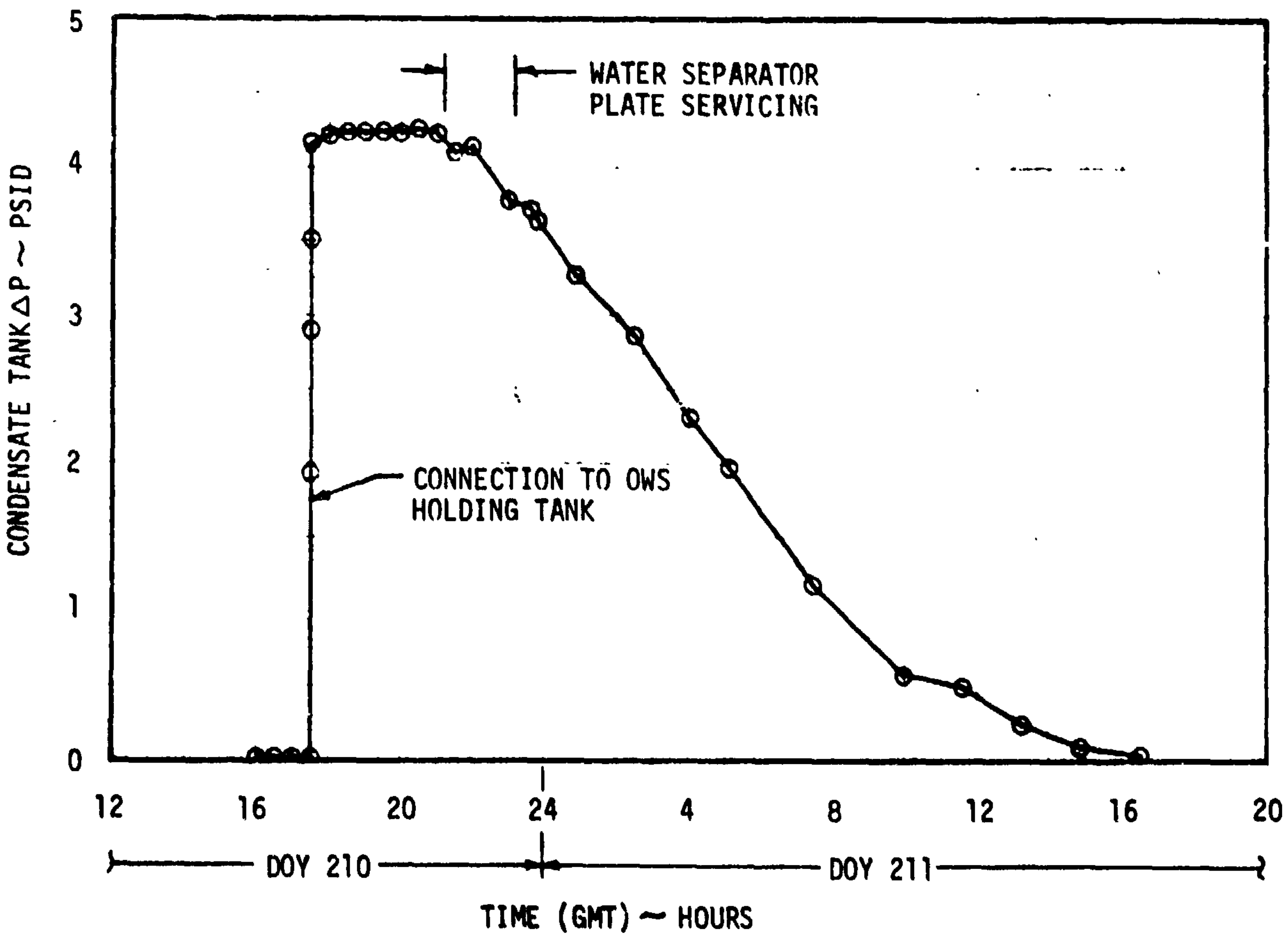


FIGURE 2.5-63 SL-3 CONDENSATE SYSTEM ACTIVATION

Condensate system activation was completely normal at the start of SL-4 with a system ΔP of 2.86 psid having been maintained during the storage period. No evidence of significant gas leakage into the system was observed until DOY 034 when the QD was disconnected from the liquid gas separator at Panel 217 after EVA operations. Following disconnection, system ΔP decreased to zero within approximately 15 minutes. After attempts to stop the leak using Krytox and universal sealant were unsuccessful, a cap launched on the SL-4 CSM was installed on the disconnected QD and no further evidence of leakage was observed.

B. Disposal - Essentially all condensate was dumped into the OWS waste tank via the OWS holding tank. The holding tank was dumped twice on SL-2, daily on SL-3 until DOY 245 when system leakage was stopped, and three times on SL-4. Prior to holding tank activation on DOY 148, the AM condensate tank was dumped directly to the OWS waste tank on one occasion. Overboard dumps via the AM overboard vent were limited to mixtures containing a high percentage of gas and only occurred during system activation on SL-2 (DOY 146), during EVA on DOY 158, and as a part of system trouble-shooting during SL-3.

C. Servicing/Deservicing Support - Servicing/deservicing functions supported by the condensate system relating to water separator plate assemblies and other equipment were accomplished in a normal manner.

2.5.4.4 Servicing/Deservicing

The original condensing heat exchanger water separator plate assemblies performed so well they were serviced and reused at the beginning of both SL-3 and SL-4. All servicing/deservicing operations went as planned.

2.5.5 Development Problems

Problems were encountered with the functional performance of five components during development of the ECS. The components were the gas flowmeters, $PPCO_2$ sensors, PLV fan and heat exchanger assemblies, 120 psi O_2 pressure regulators, and cabin pressure regulators.

A. Gas Flowmeters - Erratic gas flowmeter outputs were observed during systems testing. Investigation revealed that a combination of gas turbulence and flowmeter sensitivity was causing the erratic outputs. Performance of the flowmeter located in the AM/OWS interchange duct was improved by relocating the sensor to an area of lower turbulence. Testing also identified an incompatibility in the wiring shield grounding configuration which was causing a reduction in the sensitivity of the flowmeters. The shields for all gas flowmeter systems were reterminated. Testing the modified configurations resulted in acceptable data.

B. $PPCO_2$ Sensors - Premature depletion of $PPCO_2$ sensor cartridges was experienced during systems testing. Early depletion was caused by the flow control orifices being sized for higher than planned flowrate and caking of the sodium hydroxide in the cartridges which blocked flow. Two modifications were made to correct these problems. First, new orifices were installed to restrict sample gas flowrates for all sensors to approximately 6 cc/min at 5 psia pressure and second, the active $PPCO_2$ sensor cartridge ingredients were changed to materials which removed CO_2 and water effectively. The Ascarite, NaOH, was replaced with $LiOH \cdot H_2O$ and the water absorber, Drierite, was replaced with silica gel. In addition, the molecular sieve inlet $PPCO_2$ sensor gas sample port was relocated upstream of the condensing heat exchanger to eliminate any possibility of condensate being transferred into the sensor cartridge along with sample gas flow. As a result of these modifications, the cartridges successfully passed final tests and were requalified for useage life periods of 14 days and 28 days, respectively, for the mole sieve inlet and outlet locations.

C. PLV Fan/Cabin Heat Exchanger Assemblies - Operating characteristics of the PLV fan/cabin heat exchanger assemblies were evaluated for various operational conditions and assembly configurations. There was concern regarding blockage of gas flow by the surface tension of condensation occurring within heat exchanger fins. Results of extensive development tests and systems analyses showed that for the predicted range of moisture generation rates, condensation should not occur in the cabin heat exchangers during normal operations provided that two condensing heat exchangers were utilized for humidity control and no more than one coolant pump per coolant loop were operated for temperature control. At these operating conditions, atmospheric dew point temperature would be below the temperature of the cabin heat exchanger fins, and consequently, condensation would not occur.

D. 120 psi O_2 Pressure Regulator - Excessive leakage was detected from the pressure relief valve ports on the 120 psi O_2 pressure regulator assembly during systems testing. Erratic pressure control was also detectable. The relief valve leakage was attributed to improper sealing of the relief valves caused by hardening of relief valve seat material at low temperature. The erratic pressure control was caused by ice formation. Both occurred when flowing oxygen at high EVA/IVA oxygen flow rates (EVA/IVA flow rates are discussed in Section 2.6). Icing resulted from a combination of high cooling rates due to expansion of oxygen flow from high to low pressure and the humid atmosphere surrounding the regulator assembly during EVA/IVA periods. Except for EVA/IVA activity during orbital operations, high oxygen flow rates would occur during cluster repressurization periods, only. During these periods, gas leakage was no problem and the cluster atmosphere was dry, and consequently, condensation and freezing would not occur. To prevent leakage, regulator O-ring material was changed from LS-53 to Silicone and relief valve poppet seals were changed from Viton to Silastic 675 (Fluorosilicon). To prevent condensation and freezing during EVA/IVA activity, the pressure sensing ports on the regulator assembly and relief valves were isolated from cabin atmosphere by connecting the ports by tubing to the cabin pressure regulator discharge duct. Tubing was also added to the relief valve vent ports to preclude icing at those locations. To prevent condensation in the O_2/N_2 module, the regulator housing was insulated.

E. Cabin Pressure Regulator - Both analysis and system testing showed that excessive pressure drops through the O_2/N_2 cabin pressure regulator discharge line caused cabin pressure to fall below required limits of 5.0 $\pm$ .2 psia. Pressure drop was reduced by a plumbing configuration change which included a shorter and larger (.5") diameter discharge line. Tests of the modified system successfully demonstrated pressure control within the specification limits.

2.5.6 Conclusions and Recommendations

The ECS performed so effectively that all mission objectives were accomplished in spite of the off-nominal conditions to which the total vehicle was exposed during the first few days of the SL-1/SL-2 mission. System discrepancies during the mission were corrected by designed-in system redundancies or by real time work-around procedures. In general the Airlock Environmental Control System Design and Test Concept was an effective approach and along with system components should be considered a candidate for future programs. A more detailed discussion of ECS performance is given below:

- System design requirements were realistic and should be considered when developing a new program.
- Modular design facilitated system checkout and should be considered for new vehicle design.
- The two-gas control system was especially effective in providing cabin pressures and oxygen partial pressures well within the allowable range. A two-gas system most probably will be used on all future, long-term manned space flights and this type of a control system should be a candidate.
- Cluster O_2 and N_2 gas usage rates were well below design levels; significant quantities of both gases were available at end of mission even though unplanned purge cycles were accomplished and cabin pressure was maintained near the manned level during the orbital storage period following SL-3. The total Airlock pressure integrity design was therefore very effective and should be considered for future space usage, especially the hatch seal design and verification test program.
- The bleed orifice around the 3000 PSI solenoid shut-off valve eliminated the potential fire hazard associated with quick opening valves in oxygen systems.
- The condensing heat exchangers used in conjunction with fritted glass water separator plates effectively removed atmospheric moisture throughout the mission without the need for a changeout of separator plates. This system is a high level candidate for future usage.
- The first mission use of a molecular sieve system was outstanding. The Airlock system performed CO_2, odor, and moisture removal functions effectively with no system hardware anomalies. In fact, the system performed satisfactorily throughout the 84-day SL-4 mission without a bed

bakeout being required. This system should be considered for use on future programs.

- Lint is added to the atmosphere on long duration flights in quantities sufficient to collect on cabin heat exchangers and cause a reduced gas flow. The susceptibility of fan/heat exchanger units to contamination should be an important consideration in the choice of equipment for cabin temperature control. Future usage of these type components should include finer mesh protective screens, or increased accessibility for periodic cleaning.
- Condensation formed on cabin heat exchangers, reduced airflow, and limited cooling ability during periods of off-design operation (three coolant pump flow combined with excessive moisture addition from showers). The susceptibility of fan/heat exchanger units to condensation blockage should be an important consideration in the choice of equipment used for cabin temperature control. Future usage of these type components should either be limited to normal operating modes, or be designed for a broader range of conditions.
- The vacuum side of the condensate system had a tendency for random leaks throughout the mission. The condensate system included many quick disconnects and it was generally agreed that QD leakage was the problem. Use of QD's can be minimized on future missions as the crew has demonstrated the ability to use more positive mechanical disconnects.
- The Environmental Control System verification concept of detailed thermal analysis, plus limited testing was proven valid. No system qualification testing was performed.

2.6 EVA/IVA SUIT SYSTEM

The AM EVA/IVA system provided controlled supplies of O_2 and water via GFE interfaces for astronaut cooling and suit pressure maintenance during EVA and IVA operations.

2.6.1 Design Requirements

The basic requirement was to provide a supply of O_2, at regulated pressure and temperature, and water, at controlled temperature, flow rate, and pressure, to interface with GFE Life Support Umbilicals (LSU's) at AM EVA and IVA panels. A GFE Pressure Control Unit (PCU), attached between the LSU and pressure suit/ Liquid Cooled Garment (LCG), provided control of the O_2/water flowrates delivered to each pressure suit. The AM also was required to provide hardware for in-flight water servicing and deservicing of the LSU's and PCU's as well as controls to vent and repressurize the EVA lock compartment. The AM to LSU interface requirements are presented in ICD No. 13M07396. Additional requirements associated with the EVA/IVA Suit System design were to provide instrumentation intelligence and procedures as a basis for system operation.

2.6.1.1 Evolution

Once it was decided to provide astronaut cooling and O_2 supply during EVA and IVA via the LSU/PCU/LCG concept, the AM/LSU interface requirements were stable in principle. The AM system design was dependent upon specific LSU interface requirements as well as AM/CSM interface requirements relative to oxygen received from the CSM. Studies were conducted to define the system and interfaces. There were a number of requirement changes which, although resulting in a more flexible system with higher performance capabilities, also led to one having more inherent complexity and consequently more development problems.

The original suit cooling system proposed by MDAC-E was designed to deliver 55°F water to the LSU interface, while absorbing heat loads of 2000 Btu/hr produced by each of two EVA astronauts. Subsequent establishment of a firm requirement to limit water delivery temperatures to 45°F, made it necessary to move the EVA heat exchangers interfacing between the water and coolant loops from downstream to upstream of the coolant loop temperature control valve. This requirement, combined with the later requirement changes to provide a minimum atmospheric dew point temperature of 46°F and to supply 40°F coolant to the battery modules, had a profound impact on the coolant system design. The culmination was the design and development of the suit battery cooling module described in Section 2.4.

Other design requirement changes from the original system included the addition of a GFE liquid gas separator for removal of free gas from the circulating water loops, the addition of negative heat load to accommodate heat losses from the GFE part of the system, and utilization of the AM aft compartment in addition to the lock compartment to provide increased volume for the EVA crewmen. Implementation of the liquid gas separator assembly proved to have a significant effect on system development in that it dictated a change of additives in the suit cooling loops. This, in turn, led to materials incompatibility and pump operational problems, which were subsequently solved by using still different system additives and increasing the pump vane clearances.

The addition of the STS IVA panel, along with the requirement to supply O_2 and cooling water simultaneously to three crewmen's LSU's instead of two, resulted in increased O_2 flowrate and system heat load requirements.

2.6.1.2 Flight Configuration

The flight system functional design requirements are summarized as follows. The physical and operational requirements are described in Section 2.12.

A. Oxygen Supply - Supply 40-90°F O_2 at the following conditions to LSU's from EVA/IVA panel disconnects for suit pressurization and ventilation.

(1) With AM O_2 tank pressure greater than 400 psia, supply a total flowrate of 22.7 lb/hr to two crewmen (13.7 lb/hr to one umbilical connection and 9.0 lb/hr to the other connection) with a minimum interface pressure to the LSU of 65 psia.

(2) With AM O_2 tank pressures greater than 500 psia, supply a total flowrate of 31.7 lb/hr to three crewmen. To one umbilical connection the flowrate shall be 13.7 lb/hr at a minimum interface pressure of 55 psia, and at each of the other two connections the flowrate shall be 9 lb/hr with a minimum interface pressure of 65 psia.

(3) With AM O_2 tank pressures greater than 450 psia and loss of one crewman's umbilicals pressure integrity, provide a minimum of 9.0 lb/hr at 65 psia to the other crewman's umbilical connection.

(4) With 8 lb/hr or greater flowrates, limit maximum interface pressure to the LSU's to 132 psia for normal operation or 165 psia for a failed open AM pressure regulator. For no-flow conditions, limit maximum interface pressure to the LSU's to 176 psia.

B. Suit Cooling - Provide two systems for recirculating water at the following conditions to LSU's from EVA/IVA panel disconnects for suit cooling.

(1) Flowrate - 200 lb/hr to 325 lb/hr per SUS loop

(2) Pressure - 27.5 psia maximum @ LSU inlets during normal operation and 37.2 psia maximum during a blocked line condition.

(3) Temperature - Water delivery temperature at LSU inlets shall be as summarized below:

WATER DELIVERY TEMPERATURE AT LSU INLET ⚠1

CONDITION ⚠3		GFE H_2O HEAT LOADS (BTU/HR)		H_2O DELIVERY TEMP. (°F)	
		HEAT LOSS	HEAT INPUT	MIN.	MAX.
A	• ONE LSU CONNECTED TO EVA/IVA LOOP 1	800	2000	39	49
	• ONE LSU CONNECTED TO EVA/IVA LOOP 2	800	2000	39	49
B	• TWO LSU'S CONNECTED TO EVA/IVA LOOP 1 OR 2	800 ⚠2	3130 ⚠2	38	50
	• ONE LSU CONNECTED TO REMAINING LOOP	800	1730	39	49

⚠1 Based on nominal system performance with a solar inertial vehicle attitude and a total AM coolant system heat load (including EVA/IVA water loop loads) not exceeding 11774 Btu/hr.

⚠2 Total heat load for both LSU's.

⚠3 Under normal conditions, total duration of water flow will not exceed three hours for an EVA/IVA operation.

C. In-Flight Servicing/Deservicing - Provide in-flight servicing capability for SUS loops and servicing and deservicing of LSU's and PCU's.

D. EVA Lock Operation - Provide pressurization and depressurization of AM lock and aft compartments to accommodate EVA.

2.6.2 System Description

The system provided regulated oxygen and a flow of temperature controlled water to LSU interfaces, provisions for servicing and deservicing LSU's and PCU's, controls to vent and repressurize the EVA lock compartment and instrument intelligence.

2.6.2.1 EVA/IVA Oxygen Supply System

O_2 was supplied for pressure suit pressurization and ventilation during EVA/IVA operations from the gas system (Figure 2.5-2) to GFE LSU's interfacing with panel mounted quick disconnects. O_2 flow provisions were available at three AM control panels; i.e., IVA control panel 217 in the STS (Figure 2.6-1) and EVA control panels 317 and 323 in the lock compartment (Figures 2.6-2 and 2.5-3). Each of the EVA panels incorporated two O_2 connectors for redundancy during EVA, while the IVA panel had three in order to accommodate all three crewmen during contingency or rescue operations. Redundant sealing capability to minimize leakage from the gas system to cabin was provided by mating pressure caps on the panel connectors and upstream manual shut-off valves. The oxygen supply pressure was monitored by the EVA team on lock compartment control panel 316, Figure 2.6-4, and by the crewmen in the STS on O_2/N_2 control panel 225, Figure 2.5-6.

Flow limiting orifices were installed in each supply disconnect to prevent interruption of the AM O_2 supply to an EVA crewman if the other crewman's umbilical was broken during a normal two-man EVA operation. The orifices were designed to provide an AM/LSU interface pressure high enough to prevent O_2 flow from the crewman's Secondary Oxygen Pack (SOP), yet flow sufficient O_2 to meet the normal PCU flowrate demand range.

The system O_2 flowrate and interface supply pressure capabilities are defined in the design requirements section. These requirements were verified by tests on the assembled flight vehicle prior to shipment to KSC. Results of the tests are

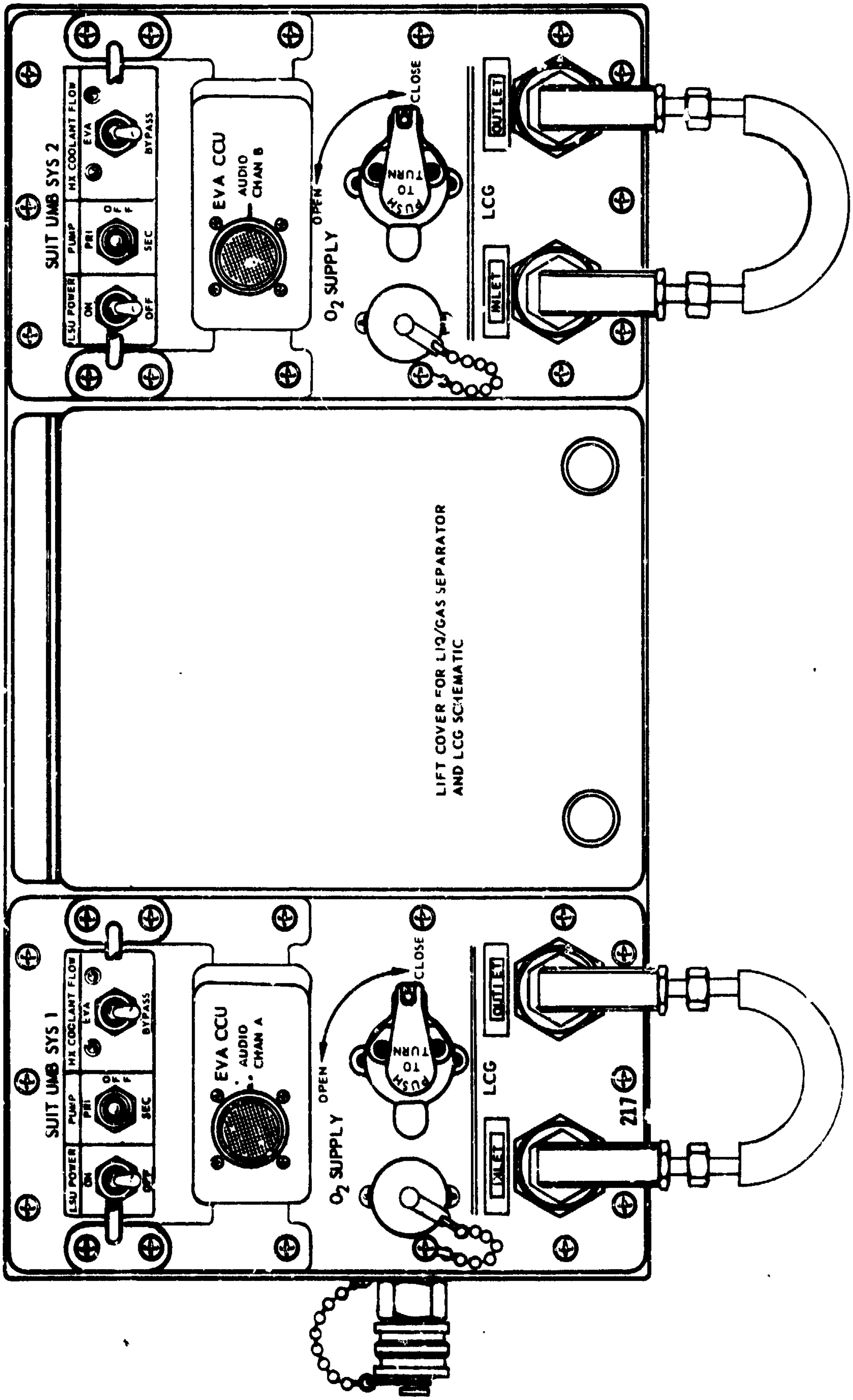


FIGURE 2.6-1 IVA CONTROL PANEL 217

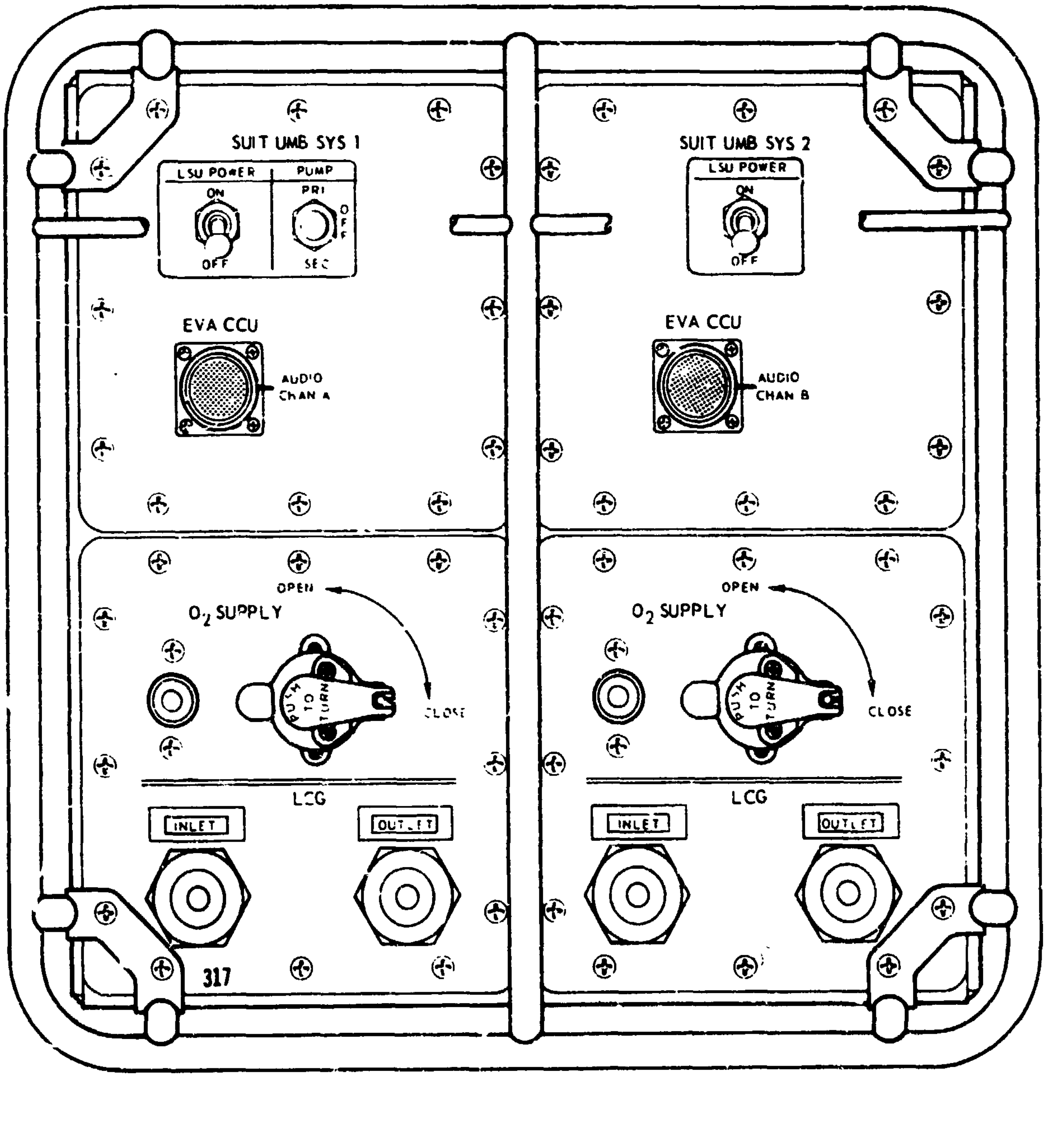


FIGURE 2.6-2 EVA NO. 1 CONTROL PANEL 317

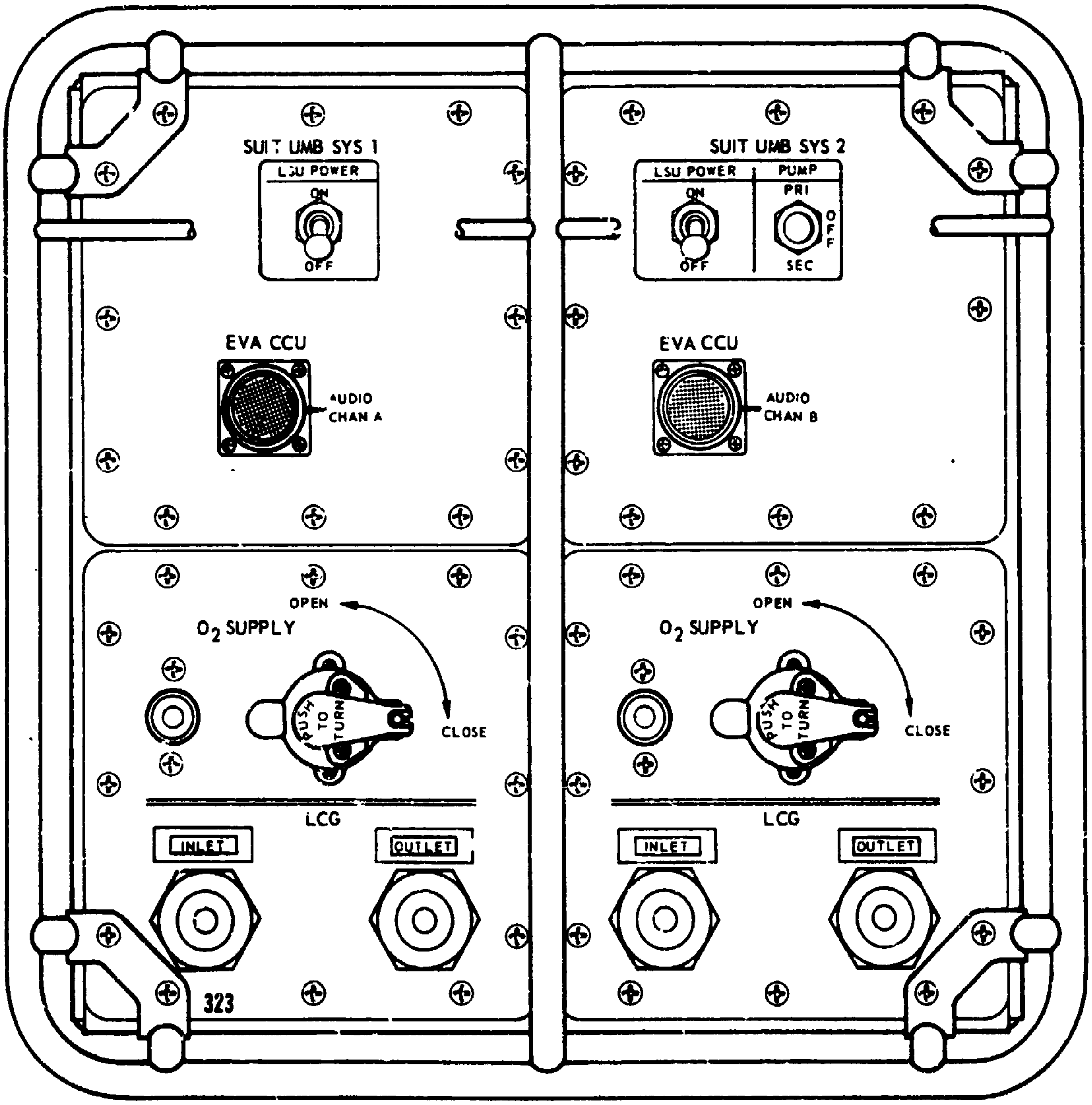


FIGURE 2.6-3 EVA NO. 2 CONTROL PANEL323

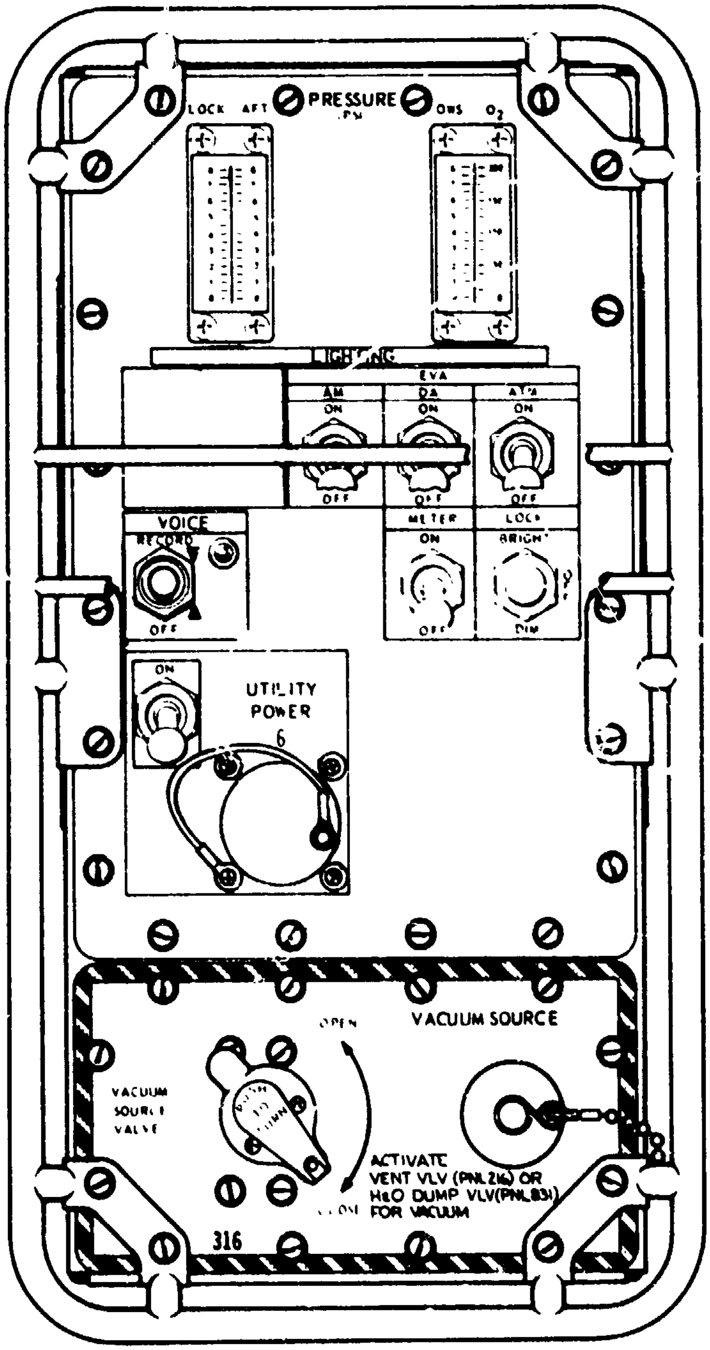


FIGURE 2.6-4 LOCK COMPARTMENT CONTROL PANEL 316

documented in SEDR D3-E76 and SEDR D3-N70. Actual O_2 flowrates delivered to the LSU's were controlled by each crewman via flow control adjustments on their PCU's. The temperature of the O_2 delivered to the LSU's was controlled between 40°F and 90°F by the O_2 heat exchanger interfacing with the coolant loop in the ATM water pump module (Figure 2.4-11).

Performance of the EVA/IVA O_2 supply system during flight was completely normal with no anomalies of any type. Mission performance is described in Section 2.6.4.

2.6.2.2 Suit Cooling System

The suit cooling system (Figure 2.6-5) provided astronaut cooling during EVA and IVA by circulating temperature controlled water through GFE liquid cooled garments (LCG's), via GFE Life Support Umbilicals (LSU's), and pressure control units (PCU's). Two identical suit umbilical systems were provided. These were designated as SUS 1 and SUS 2 and interfaced with the primary and secondary coolant loops, respectively, at heat exchangers located in the suit/battery cooling modules. The interfaces with the LSU's were at water supply and return quick disconnects, mounted on IVA control panel 217 and EVA control panels 317 and 323 (Figures 2.6-1, 2.6-2, and 2.6-3). The suit umbilical system controls were also provided at these panels. Astronaut cooling was regulated by adjusting the LCG water flowrates with the GFE PCU flow diverter valves.

A. Pumping System - To provide a relatively constant flowrate for a wide range of conditions, the pumps were of a positive displacement, rotary vane type design. Each pump was powered by a two-phase induction motor and inverter contained within the pump assembly. Pump inlet pressure maintained between 4.8 and 6.2 psia in orbit by N_2 supplied from redundant pressure regulators in the Gas System (Figure 2.5-2). Each had an internal relief valve to limit LCG pressures to 37.2 psia maximum in case of a blocked line. Each SUS loop contained redundant water pumps. In the event of a pump and/or pressurization failure, an EVA suit coolant pump differential low warning would have been indicated by illumination of EVA 1 or EVA 2 warning indicator on the lighting, caution and warning control panel 207, see Figure 2.6-6. Since the same warning indicators also could be illuminated by the temperature of water solution supplied to the umbilical being low, the distinction must be made by use of the inhibit switches. To permit

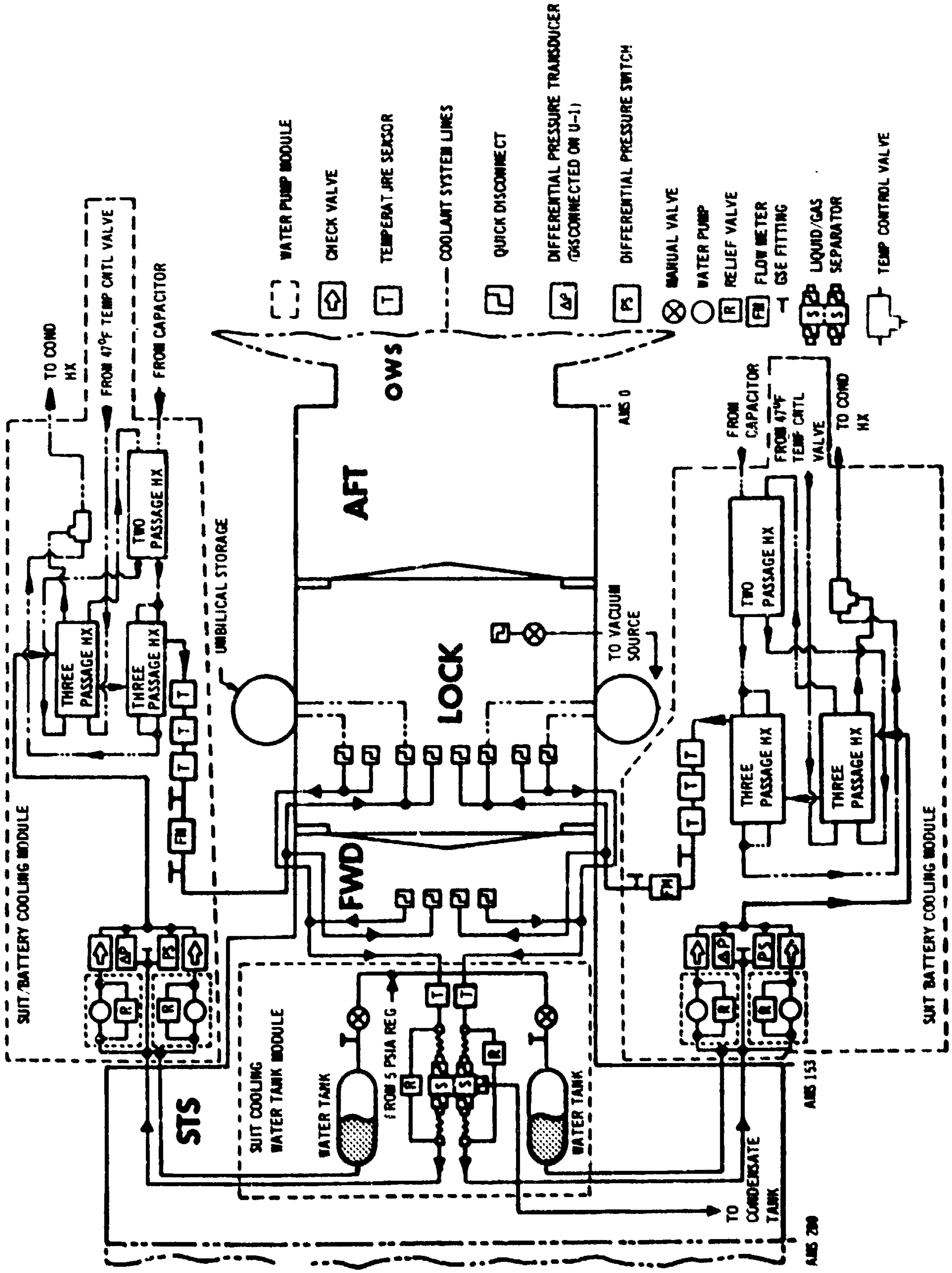


FIGURE 2.6-5 AIRLOCK SUIT COOLING SYSTEM

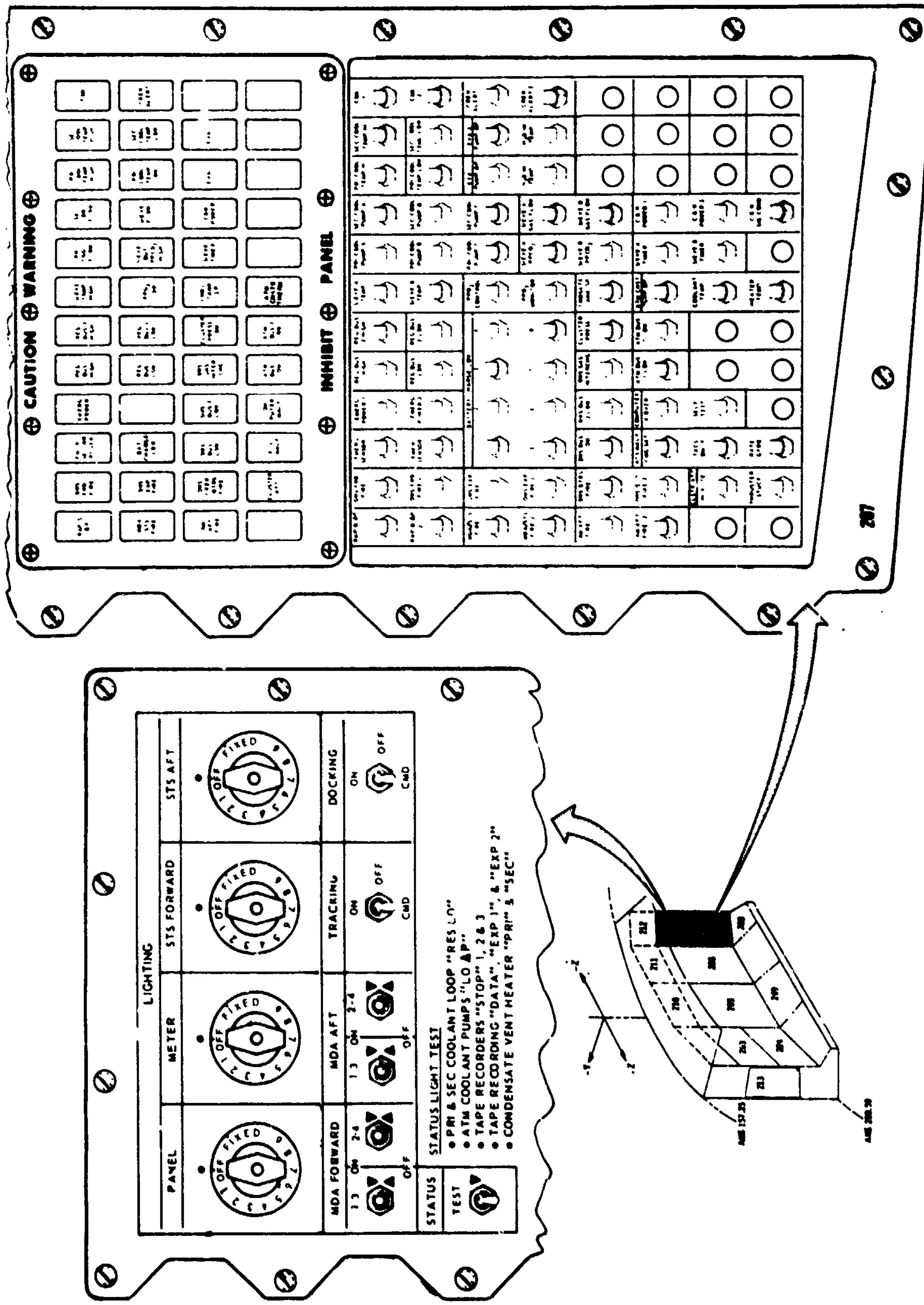


FIGURE 2.6–6 LIGHTING, CAUTION AND WARNING CONTROL PANEL 207

operation following certain failures, N_2 pressurization supply shut-off valves were provided on Panels 223 and 224, as illustrated in Figure 2.6-7.

The SUS loops were launched serviced with deionized water containing $10 \,^{+10}_{-0}$ PPM Movidyn which served as a biocide and 500 ±50 PPM sodium chromate which served as a corrosion inhibitor (PS 13240 Type VII Solution). The reservoirs contained 12 lbs of water solution at launch to replenish water lost from the loop by leakage.

Two LSU's stored in stowage containers 310 and 311 (Figure 2.6-8) were also launched serviced with water. The remainder of the GFE LSU's, PCU's, and LCG's were either stowed in the OWS and MDA, or launched onboard the CSM. Of the four additional LSU's one was serviced and three were dry. A total of eight PCU's were available through SL-4, and only one was not serviced. All nineteen LCG's available were serviced.

A GFE liquid/gas separator was provided for each loop to remove entrapped gases and trap solid contaminants. The separators for the two loops were combined into an in-flight replaceable assembly and installed as shown in Figure 2.6-9. Two spare separator assemblies were stored in stowage container 305 as shown in Figure 2.6-10.

Provisions were available for telemetering water flowrates, system inlet and outlet temperatures, and pump pressure rise. The pump ΔP transducers were disconnected at KSC because of potential adverse electrical effects on the telemetry system. Therefore, no indication of water pump pressure rise was available in-flight.

(1) Flowrate - System water flowrate performance is defined in Figure 2.6-11 for one, two, and three astronauts connected to one SUS loop. The pressure drop through the LSU, PCU, and LCG portion of the loop depended upon the position of the flow diverter valve in the PCU. When there was more than one astronaut connected to a loop, the one who directed more of the water through his LCG received less of the flow split to his LSU. Normal design EVA operations were with two crewman's LSU's connected to one SUS loop and the third crewman connected to the other. The water flowrates delivered to the crew during the mission were within the required range of

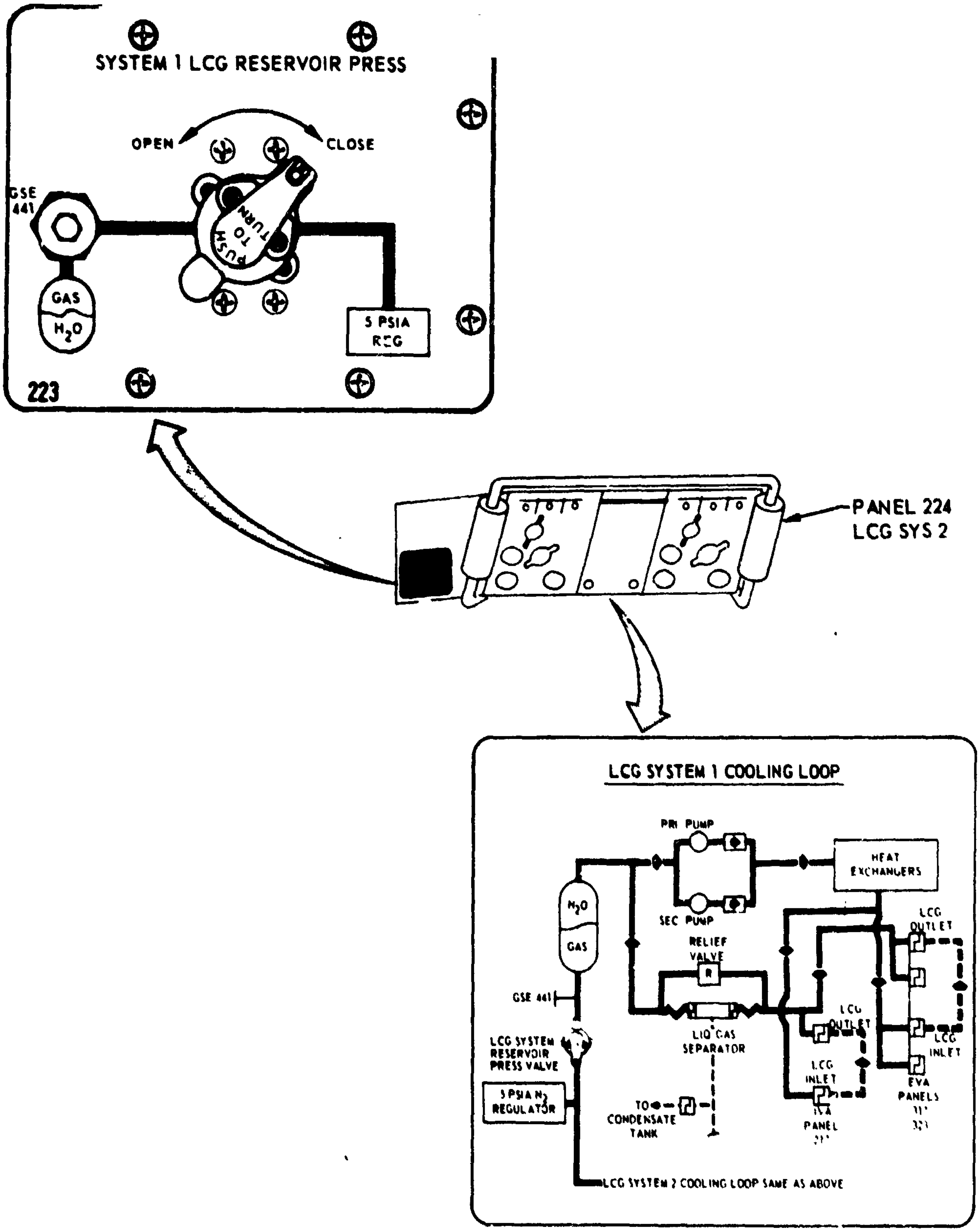


FIGURE 2.6-7 SYSTEM 1 LCG RESERVOIR PRESSURE VALVE PANEL 223

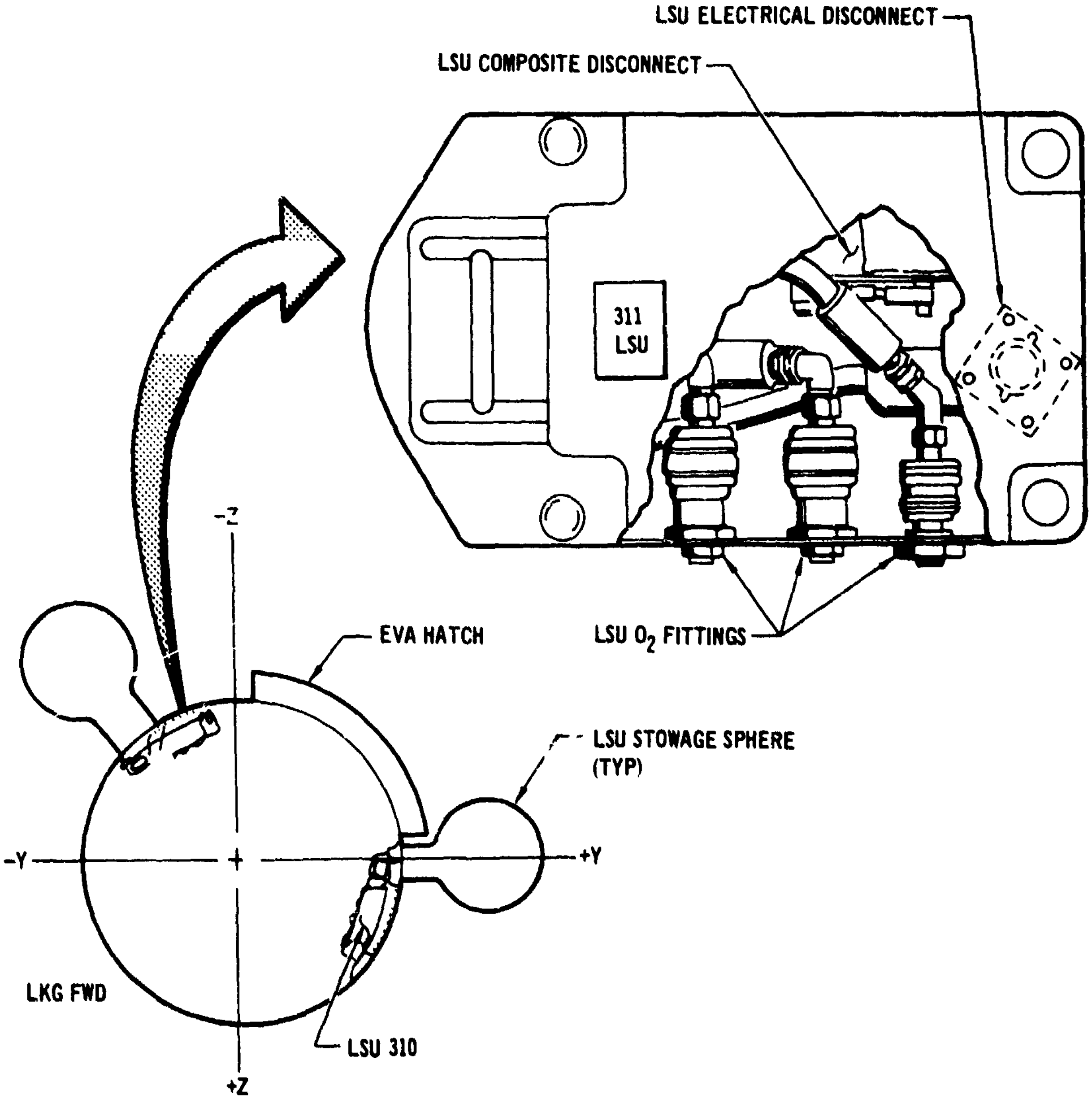


FIGURE 2.6–8 LSU STOWAGE IN AM

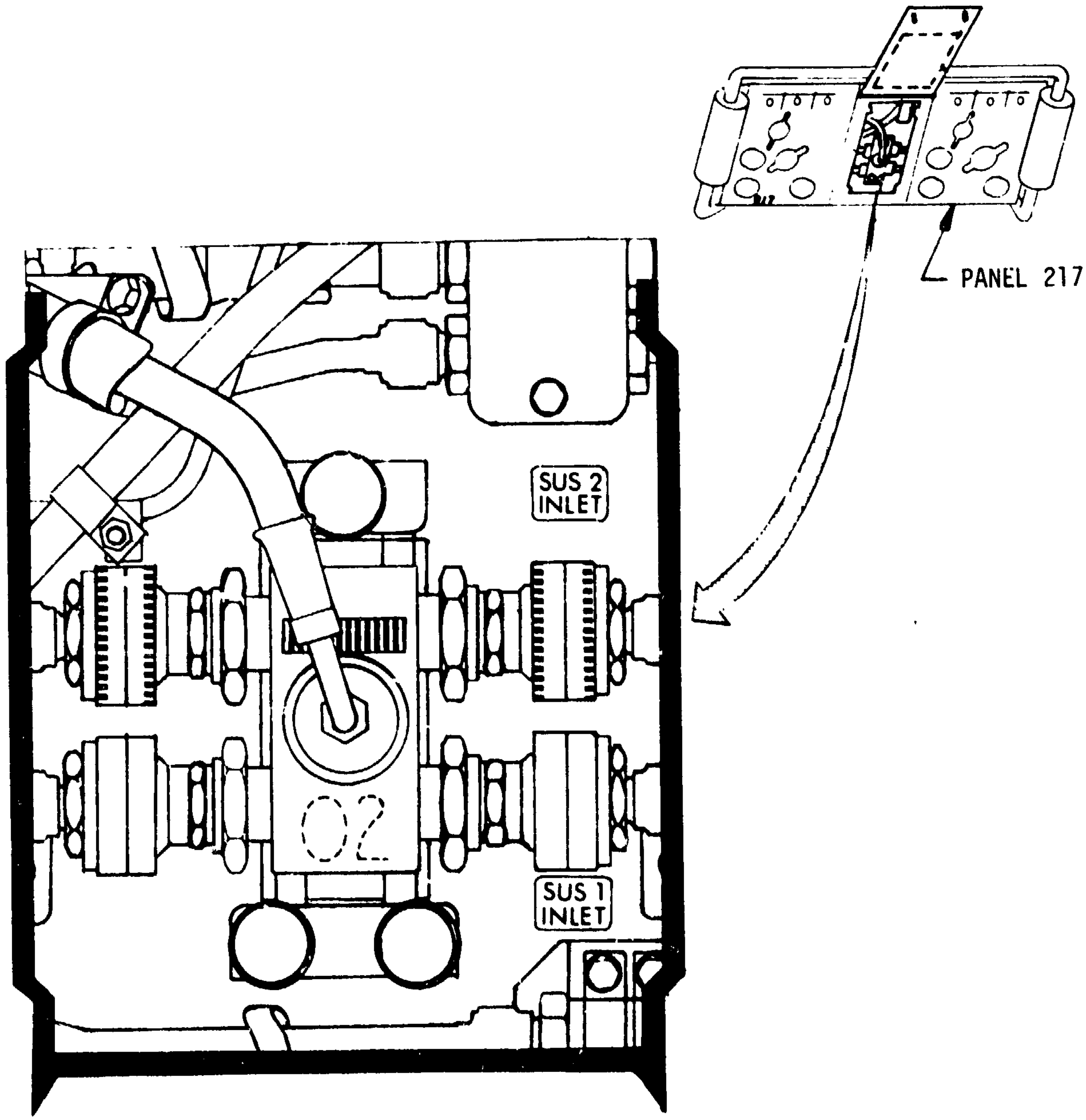


FIGURE 2.6-9 LIQUID/GAS SEPARATOR

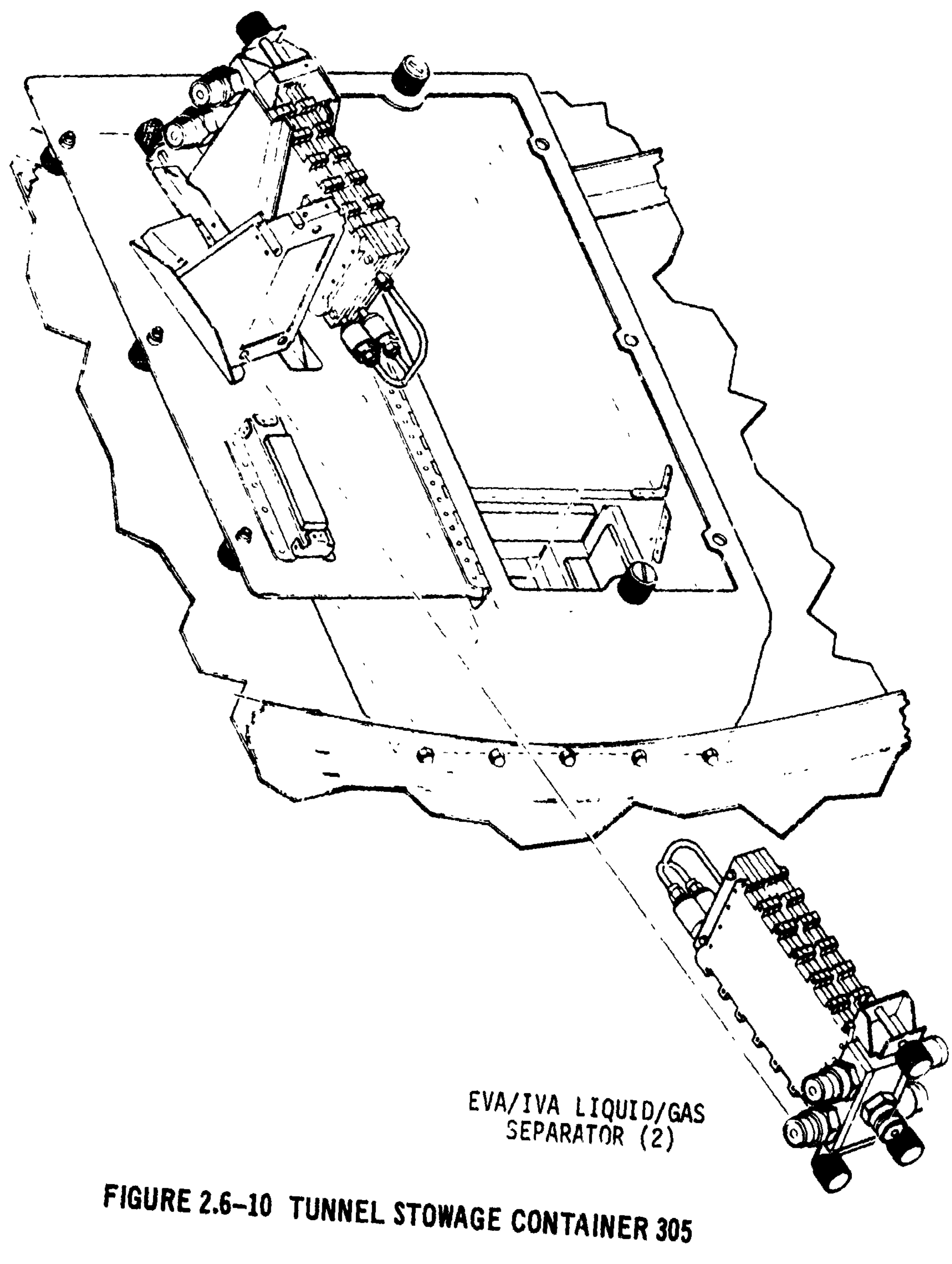
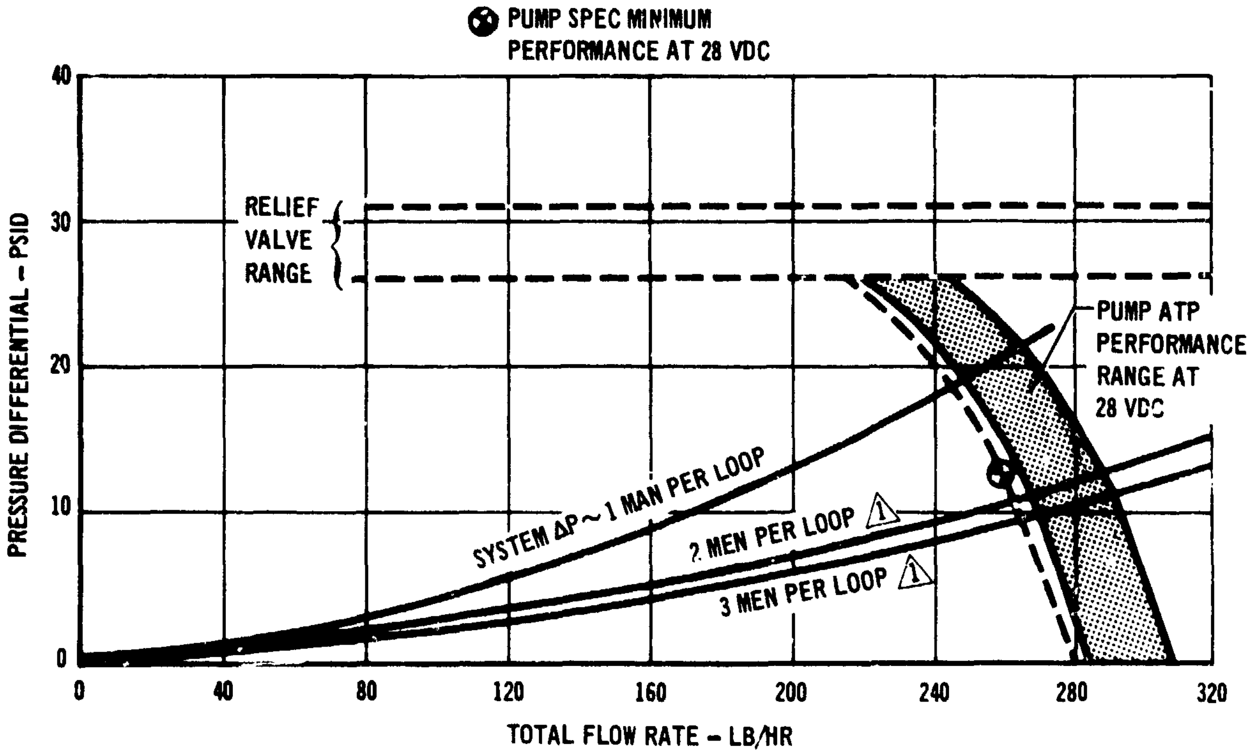


FIGURE 2.6-10 TUNNEL STOWAGE CONTAINER 305



1 TYPICAL FLOWSPLITS FOR 2 & 3 MAN PER LOOP OPERATION:

CONDITION	CREWMAN	PCU DIVERTER VLV POSITION	FLOW RATE – LB/HR	
			TO LSU	TO LCG
TWO MEN PER LOOP	NO. 1	MAXIMUM LCG FLOW	116	116
	NO. 2	4 OR BELOW	165	46 MAXIMUM
THREE MEN PER LOOP	NO. 1	MAXIMUM LCG FLOW	82	82
	NO. 2	MAXIMUM LCG FLOW	82	82
	NO. 3	4 OR BELOW	120	46 MAXIMUM

FIGURE 2.6-11 SUS WATER FLOW RATE PERFORMANCE

200 lb/hr to 325 lb/hr during all system configurations, as described in Section 2.6.4.

(2) Pressure - Pressure at the LSU inlet interface was limited to protect GFE. Pressure at the LSU inlet was calculated as the sum of the reservoir pressure and the pressure drop of GFE and AM system from the umbilical inlet to the reservoir and was measured directly during spacecraft systems testing as part of SEDR D3-E76. The maximum operating pressure was limited to 27.5 psia for normal operation and 37.2 psia for blocked line operation in accordance with the design requirements per analysis, test, and satisfactory flight experience.

B. Temperature Control - The temperature of water delivered to the LSU's was controlled by the coolant system, due to the arrangement and location of the heat exchangers with respect to the coolant loop thermal control valves and coolant flow paths. The water temperature delivered to the LSU's was a function of SUS loop heat load; with the maximum occurring at zero heat load, within the limits of the radiator and the ·al capacitor performance capability. Heat added to the water circulated through the GFE LSU's and LCG's was rejected to the coolant loop via heat exchangers in the suit battery module. For normal SUS loop design operation, the coolant loop diverter valve applicable to each SUS loop was positioned from BYPASS to EVA at Panel 217 after SUS pump activation. This resulted in coolant flow through both heat exchangers and provided maximum system heat rejection capability.

The system design capacity was based on one crewman connected to each SUS loop. The requirement was to provide a heat rejection capability of -800 to 2000 Btu/hr per loop, while delivering 200-325 lb/hr water to each LSU interface at 39°F to 49°F. The capability of each SUS loop to meet these requirements with one pump operating in each coolant loop is illustrated in Figure 2.6-12. System performance for one SUS loop with two pumps operating in a coolant loop is shown in Figure 2.6-13. The performance shown in both Figures 2.6-12 and 2.6-13 was determined primarily from the results of test data obtained during the DT-34 subsystem development test. Although this latter mode was initially intended for contingency operation

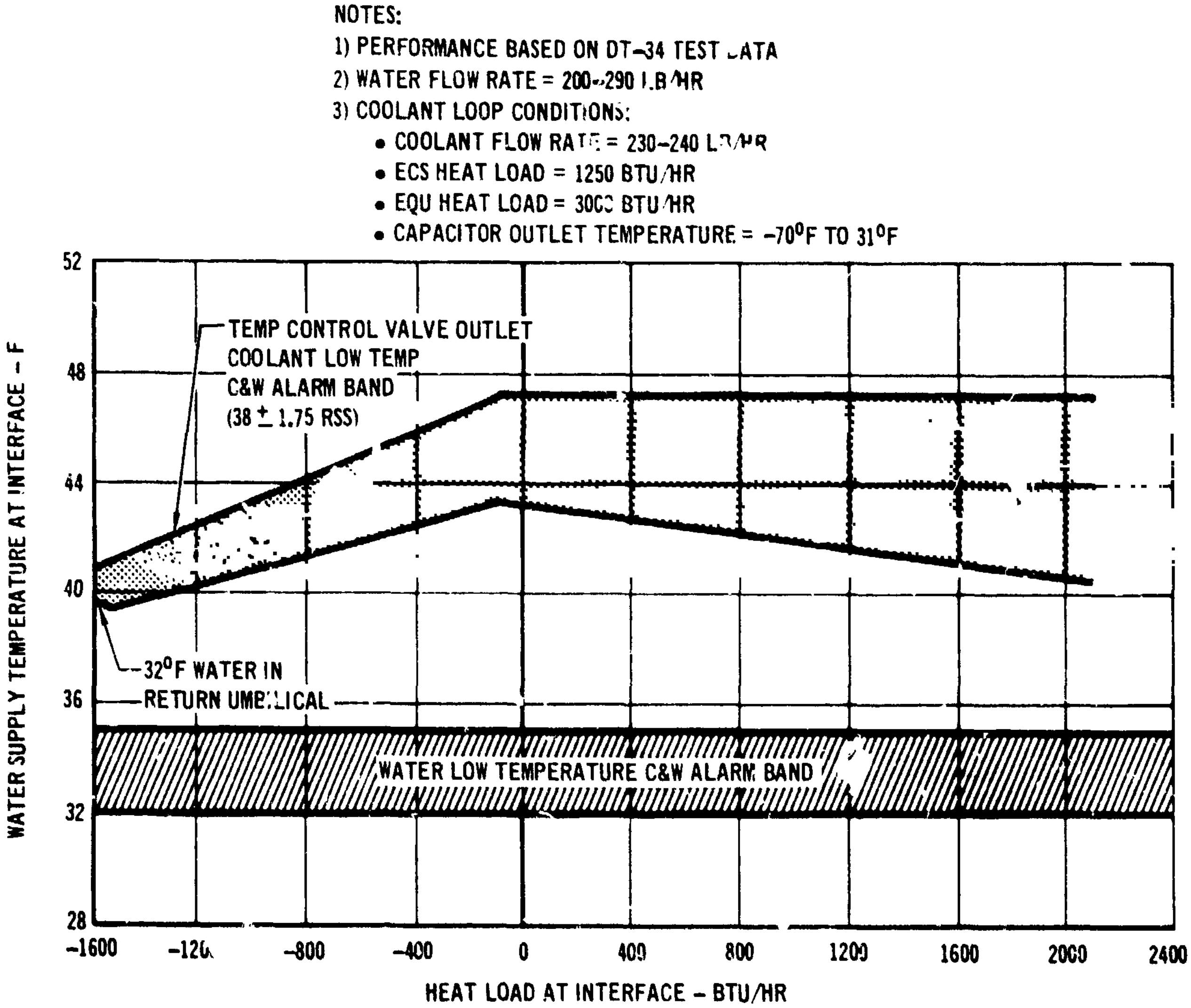


FIGURE 2.6-12 SUIT COOLING SYSTEM PERFORMANCE
(ONE COOLANT PUMP OPERATION)

NOTES:
1) PERFORMANCE CASED ON DT-34 TEST DATA
2) WATER FLOW RATE = 250-305 LB/HR
3) COOLANT LOOP CONDITIONS
- COOLANT FLOW RATE = 420-460 LB/HR
- ECS HEAT LOAD = 2500 BTU/HR
- EQU HEAT LOAD = 6000 BTU/HR
- CAPACITOR OUTLET TEMPERATURE = -70°F TO 26°F

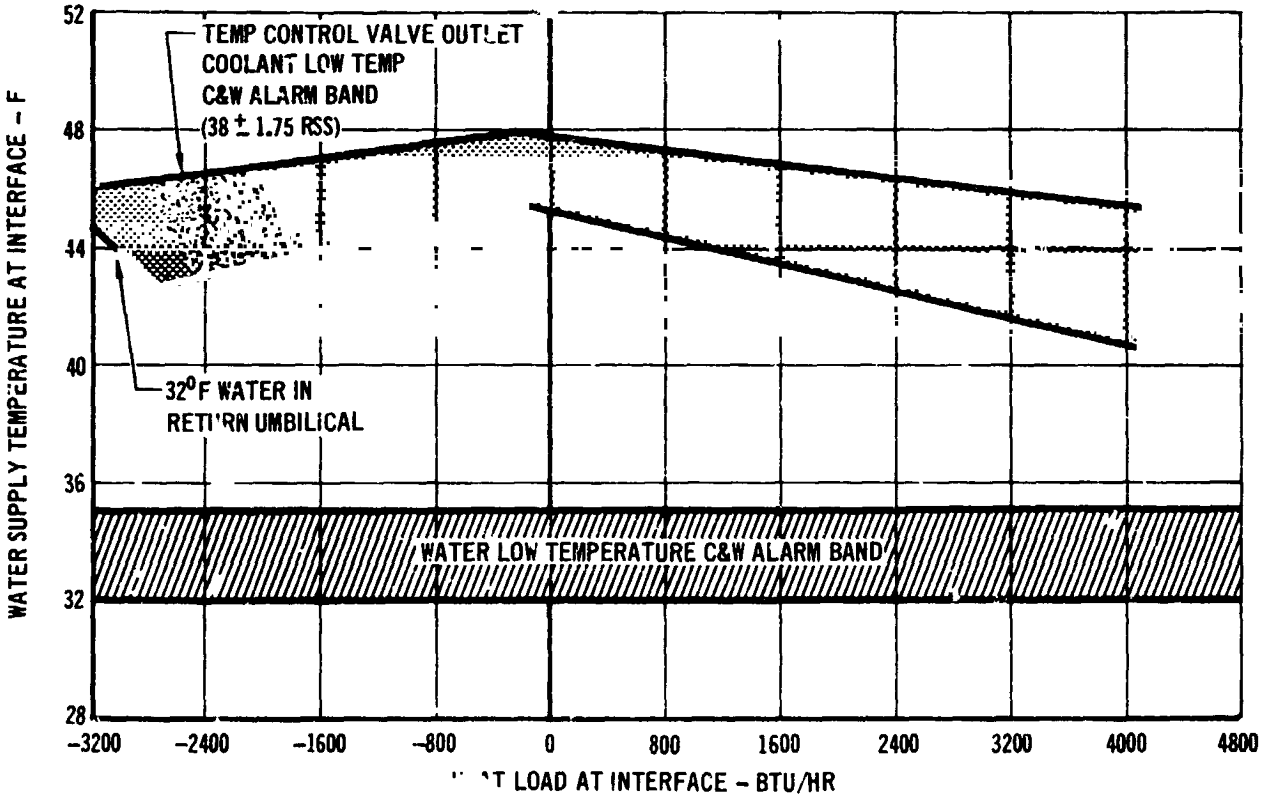


FIGURE 2.6-13 SUIT COOLING SYSTEM PERFORMANCE (TWO COOLANT PUMP OPERATION)

for cooling two EVA crewmen from one SUS loop in the event of failure of the other SUS or coolant loop, it was also applicable for cooling one or three crewmen connected to the same SUS loop. In the event of a failure which caused the temperature of water solution supplied to the umbilical to be as low as the C&W alarm band shown in Figures 2.6-12 and 2.6-13, the EVA 1 or EVA 2 warning indicators on lighting, caution and warning control panel 207, shown in Figure 2.6-6, would illuminate. Since the same warning indicators also could be illuminated by the pump pressure rise being low, the distinction must be made by use of the inhibit switches.

A later system requirement resulting from operational procedural changes late in the program after design completion was to provide cooling water to the IVA crewman as well as the two EVA crewmen. This requirement was to provide a heat rejection capability of -800 to 3130 Btu/hr for two crewmen on one SUS loop and -800 to 1730 Btu/hr for the third crewman on the other loop, while delivering 200-325 lb/hr water from each loop to the respective LSU interfaces at 38°F to 50°F and 39°F to 49°F. The capability of the system to meet these conditions with one pump operating in each coolant loop was determined to be available as long as the coolant system gross heat load was less than 11,774 Btu/hr. Higher coolant system heat loads would have caused increased radiator and thermal capacitor outlet coolant temperatures, and thus higher SUS loop water temperatures.

During EVA and IVA operations with the coolant loop diverter valve in the BYPASS position, higher water delivery temperatures and reduced cooling were available as shown in Figure 2.6-14. Due to coolant system problems experienced during the mission, this operational mode was used during all EVA and IVA operations except the first one, as described in Section 2.6.4. However, suit cooling was reported to be adequate even with three crewmen connected to the one SUS loop.

Freezing of water in lines on the suit/battery module was prevented by an additional thermal curtain over the module. Freezing of water in lines external to the suit/battery module while the system was inoperative was prevented by providing a controlled heat leak from warm coolant lines.

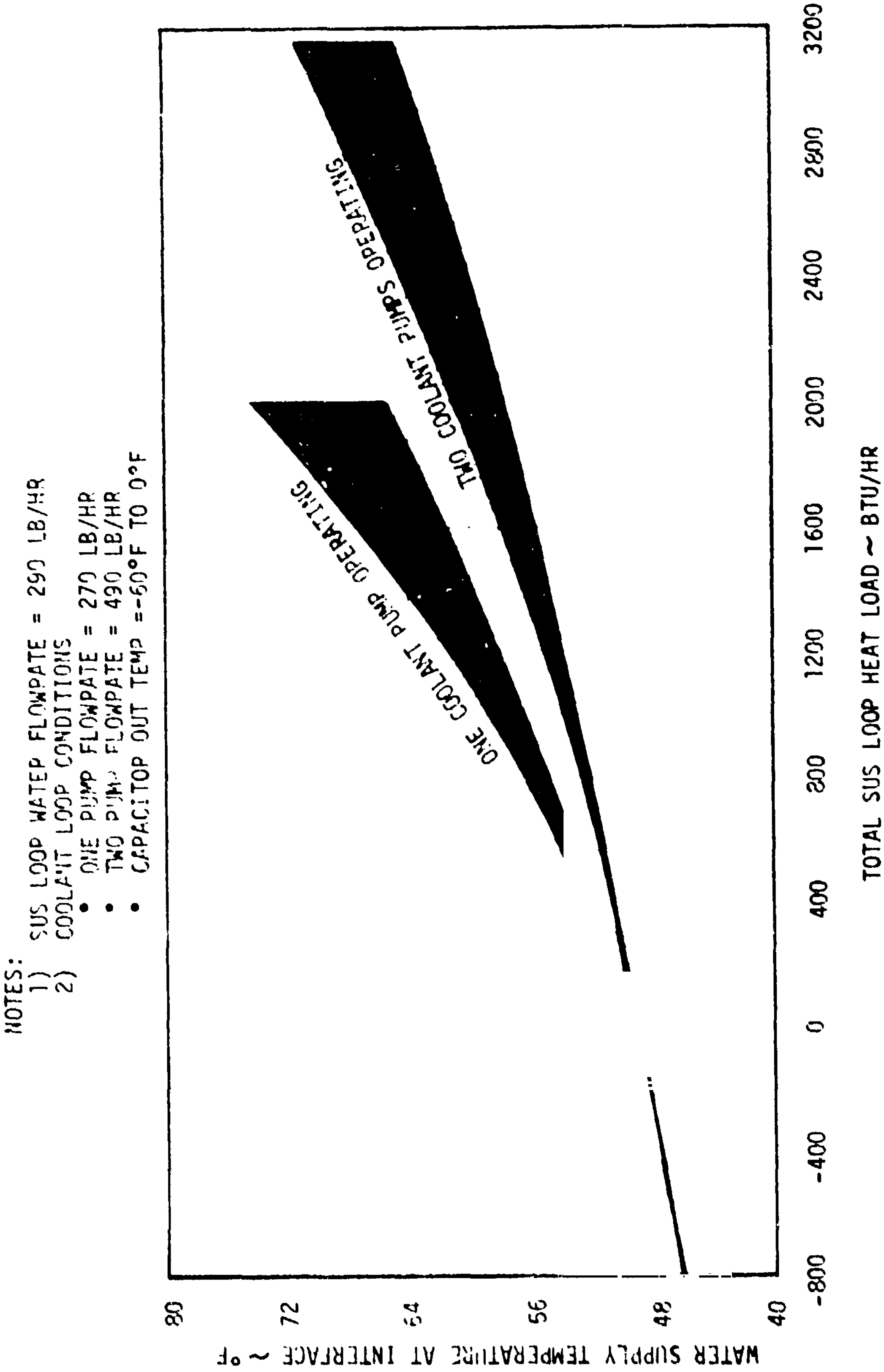


FIGURE 2.6-14 SUIT COOLING SYSTEM PERFORMANCE (COOLANT DIVERTER VALVE IN BYPASS POSITION)

C. Jumper Hose Assembly - A removable jumper hose assembly was attached to each pair of water quick disconnects on panel 217, Figure 2.6-1, to prevent excessive pressure buildup due to thermal expansion of the water between check valves at the pump outlet and the quick disconnects without connecting the GFE. In addition, they permitted circulation with pump operation between EVA/IVA operations without connecting the GFE. Each jumper hose assembly was disconnected when the suit cooling system was operating.

2.6.2.3 In-flight Water Servicing/Deservicing

Provisions were available for servicing the SUS loops, LSU's, and PCU's in-flight with water from the OWS tanks as well as deservicing the LSU's and PCU's. The servicing equipment is described in Section 2.5.2.4 and depicted in Figure 2.5-34. The detailed procedures are outlined in the SWS Systems Checklist. The additives initially in the SUS loops were sufficiently concentrated to tolerate dilution resulting from addition of the untreated OWS water.

LSU and PCU servicing was done by a flow-through purge technique; consisting of flowing water from OWS Tank #9 through the deionizer, servicing hose assembly, LSU and PCU, and through the AM condensate system into the OWS holding tank. The gas contained initially in the LSU/PCU assembly was thus purged out and replaced by deionized water. Deservicing the LSU/PCU was accomplished with a similar hardware arrangement, except the deservicing adapter was attached to the inlet of the 60 ft. servicing hose. This permitted cabin gas to flow through the assembled servicing hose and LSU/PCU and into the OWS holding tank, thus displacing the water in the LSU and PCU with gas. Servicing the AM portion of the SUS loops was provided by flowing water from OWS Tank #9, through the servicing equipment, and into the SUS loop via the liquid/gas separator inlet quick disconnect.

The in-flight servicing and deservicing operations conducted during the mission are described in Section 2.6.4. Performance of the servicing equipment and procedures was completely normal and adequate; with no hardware failures or anomalies being experienced.

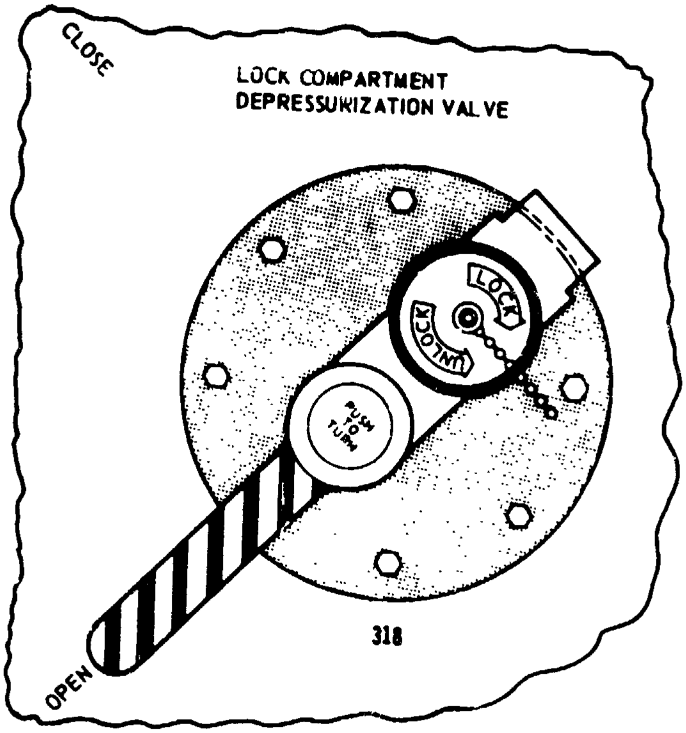
2.6.2.4 EVA Lock Operations

Lock compartment pressure control for EVA was provided by a vent valve in the EVA lock compartment wall which permitted depressurization of the AM lock compartment and equalization valves in the internal AM and OWS hatches for repressurization from the CSM/MDA/STS and OWS atmospheres. The EVA team monitored lock, aft, and OWS pressures on lock compartment control panel 316, shown in Figure 2.6-4. The crewman in the STS monitored OWS, FWD, lock, and aft pressures on O_2/N_2 control panel 225, shown in Figure 2.5-6.

A. Venting - The combined lock and aft compartments were normally depressurized for EVA using valve 318 shown in Figure 2.6-15; with the AM aft hatch maintained in a stowed position and available for contingency use. The 279 ft^3 volume could be depressurized through the vent valve to 0.20 psia in approximately 200 seconds, as shown by the depressurization profile in Figure 2.6-16. Slower depressurization was available by partially opening valve 318. The EVA hatch could be opened safely at pressures below 0.3 psid.

B. Repressurization - The primary method of repressurization utilized suit exhaust flow for an initial two-minute period to verify compartment pressure integrity. This was followed by compartment pressurization from the CSM/MDA/STS atmospheres via the AM forward hatch equalization valve, shown in Figure 2.6-15, for 30 seconds to achieve a relatively soft-suit condition. Final compartment pressure equalization was then achieved by opening the AM forward hatch handle, opening the OWS hatch equalization valve, and then opening the OWS and AM forward hatches. The resulting repressurization profile is shown in Figure 2.6-17.

   An alternate method of repressurization was available by opening the equalization valve in the OWS dome hatch. The combined lock/aft compartment could be repressurized to 4.86 psia using suit exhaust and OWS gas in approximately 160 seconds as shown in Figure 2.6-18. The main disadvantage to this method was poorer access to the OWS hatch equalization valve during a hard suit condition.



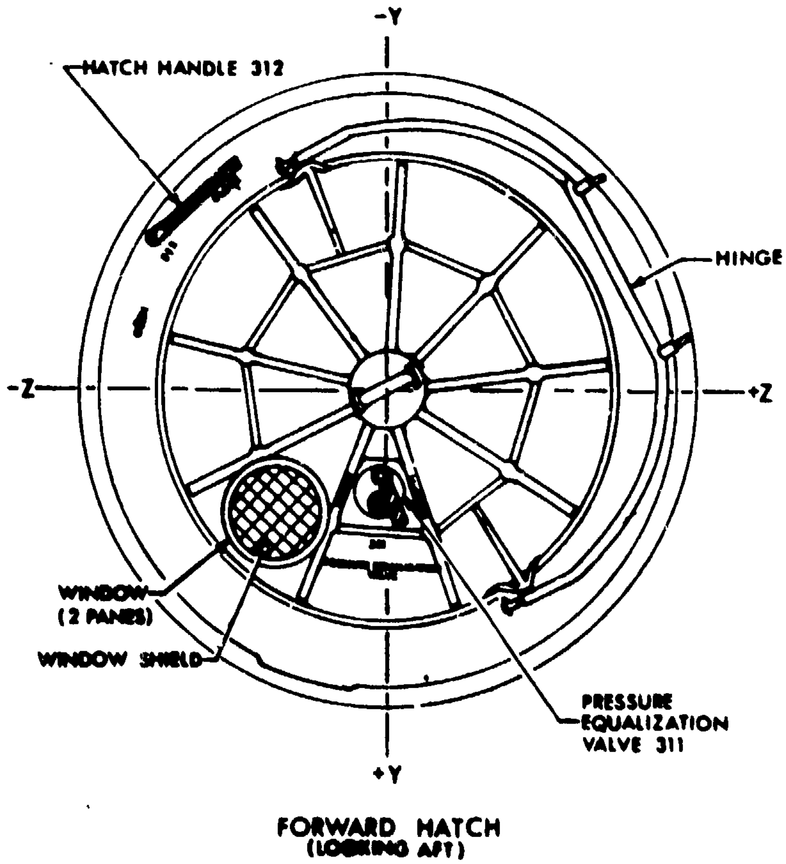


FIGURE 2.6-15 LOCK DEPRESSURIZATION VALVE AND FORWARD HATCH

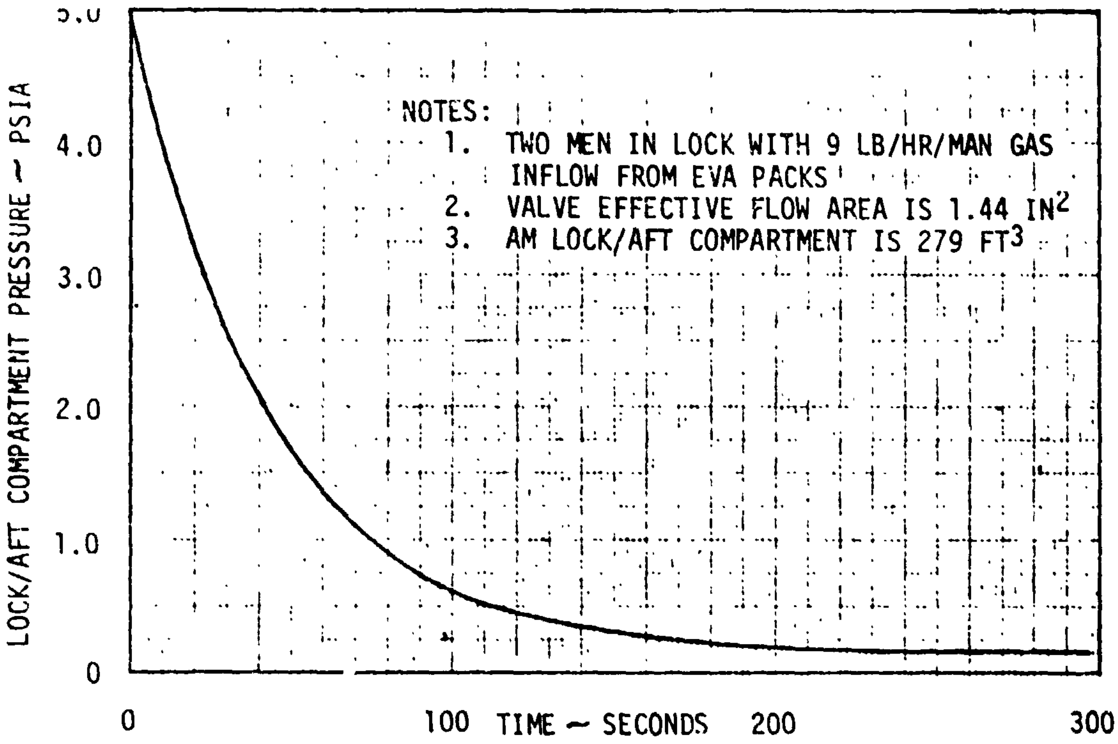


FIGURE 2.6-16 LOCK/AFT COMPARTMENT VENTING FOR EVA

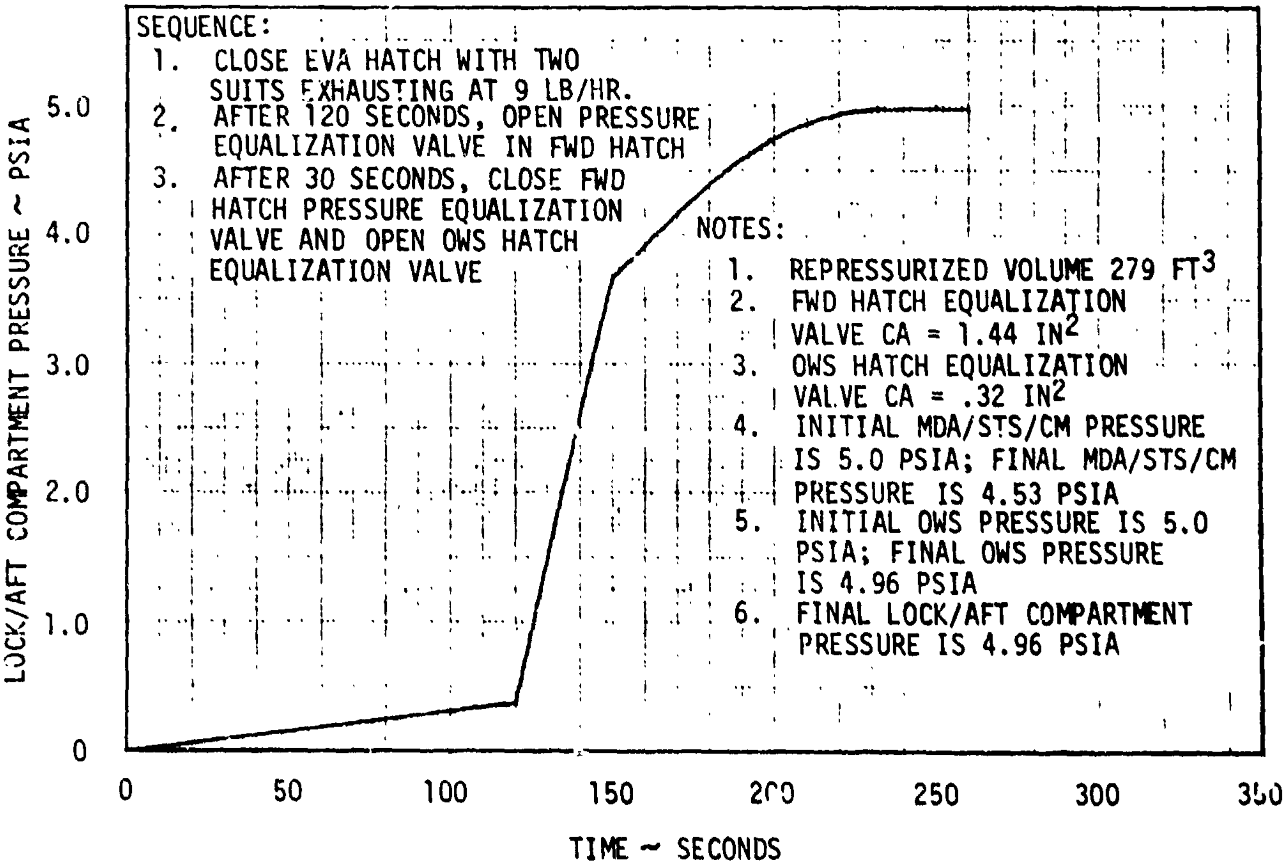


FIGURE 2.6-17 EVA LOCK/AFT COMPARTMENT REPRESSURIZATION – PRIMARY

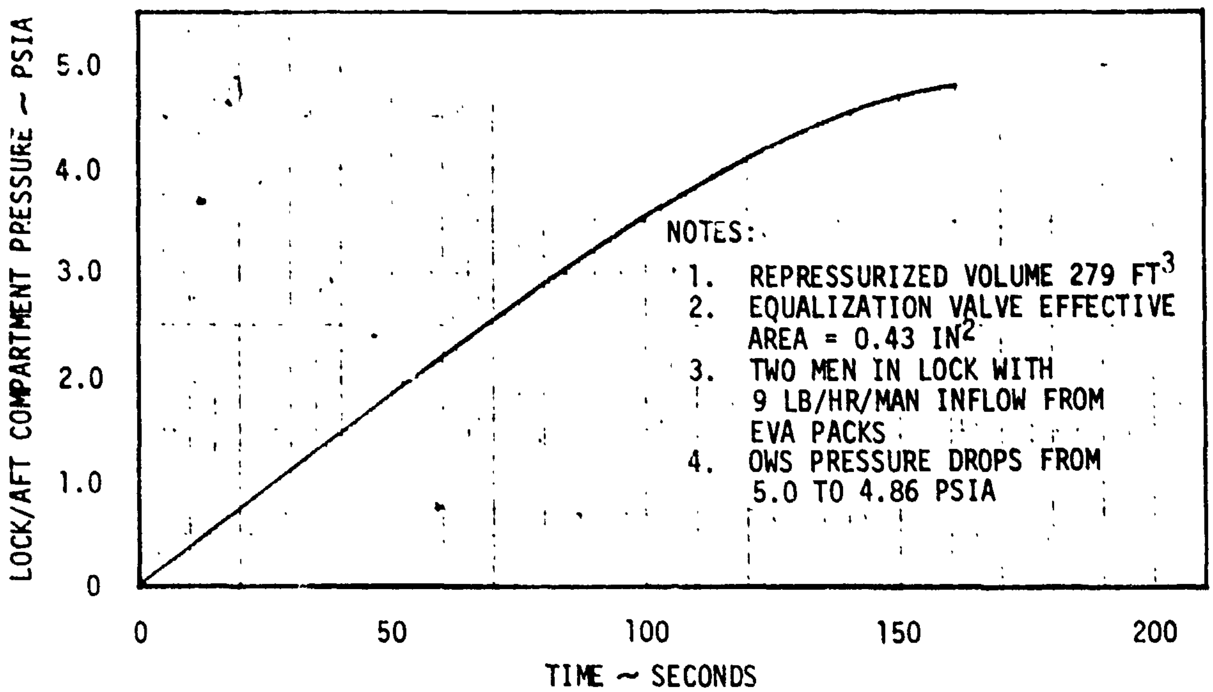


FIGURE 2.6-18 EVA LOCK/AFT REPRESSURIZATION PROFILE - ALTERNATE

EVA lock operations during the mission as well as resultant vent and repressurization profiles are described in Section 2.6.4. No problems were encountered except during the first mission when ice formed on the vent valve screen during lock venting which was removed by the crew. A removable screen cap was fabricated and launched on SL-3 for use during the second and third missions. This separate screen was installed on the vent valve prior to lock depressurization. After ice accumulation on it had resulted in a stabilized pressure during venting, it was removed; thus, allowing final venting through the ice-free screen of the vent valve.

2.6.3 Testing

Pre-launch tests were performed to provide information needed by engineering for design, to qualify a particular part numbered component, to verify that the particular part and serial numbered components operated properly, to verify that U-1 and U-2 modules and systems functioned properly, and to support verification that the vehicle was ready for flight. Post launch tests were conducted to provide information needed for real time mission planning.

2.6.3.1 Development Tests

Development tests were performed on components and systems to obtain data on the functional characteristics needed to support the design process. Test requirements were specified by Test Request (TR).

A. Performance Tests - Performance tests were conducted to establish the performance of new components and systems. Some were conducted by vendors to satisfy requirements identified in Specification Control Drawings (SCD). Those tests conducted by MDAC-E are summarized below.

- TITLE — EVA/IVA Water Cooling Subsystem Development Test.

  BACKGROUND — The Water Cooling Module rejected metabolic heat which had been absorbed from an Astronaut by water flowing through the LCG.

  OBJECTIVE — Evaluate the performance of the EVA/IVA Water Cooling Subsystem.

  RESULTS — The performance of the EVA/IVA Water Cooling Subsystem was satisfactory for all test conditions. Reference TR 061-068.22.

- TITLE — Water Pump Gas Tolerance Development Test.

  BACKGROUND — Free gas might be present in Water Cooling Systems.

  OBJECTIVE — Determine the effects of dissolved and free gas on water pump performance.

  RESULTS — The water flowrate decreased as the air injection rates were increased but the pump did not stall under any of the conditions tested. Reference TR 061-068.77.

BACKGROUND The LSU and PCU were to be serviced with water prior to use during EVA/IVA.

OBJECTIVE Develop the in-flight procedures for servicing and deservicing the LSU/PCU and for servicing the AM SUS loops.

RESULTS Procedures for servicing and for deservicing the LSU/PCU and servicing the AM SUS loops were developed. Reference TR 061-068.81.

- TITLE Water Pump Flush And Dry Confidence Test: Iodine Water Injection and Absorption Test.

BACKGROUND Water flush and vacuum drying procedures must be developed for water pumps.

OBJECTIVE Determine an acceptable water flushing and vacuum drying procedure for the water pumps.

RESULTS No procedure was developed which consistently flushed the water solutions from the pump. No apparent damage occurred to the pump when water containing iodine was mixed with the Airlock water solutions. Reference TR 061-068.85.

- TITLE 61C830069 Water Pump/Suit Cooling Loop Additive Test.

BACKGROUND Restart failures had occurred with the water pumps, used in the EVA/IVA suit cooling loops.

OBJECTIVE Evaluate the compatibility of the water pump with PS 13240 Type II, IV, V, AND VI fluids. Determine if pump restart could be achieved with PS 13240 Type VII solution. Evaluate procedures for servicing the suit cooling system with Type VII fluid. Verify proper operation of the Suit Cooling System and the ALSA when serviced with TYPE VII fluid.

RESULTS Pumps serviced with Type II, IV, and VI fluids showed deposits of materials on pump vanes and rotor after restart failures. Pump serviced with Type V fluid operated satisfactorily and showed no evidence of deposits. Reference TR 061-168.10, TR 061-168.10.01. Pumps serviced with Type VII operated satisfactorily and showed no evidence of deposites. Reference TR 061-168.10.02.

Procedures for servicing the Suit Cooling System with Type VII fluid were satisfactory. Reference 061-168.10.05.

Suit Cooling System and ALSA operated satisfactorily when serviced with Type VII fluid. Reference TR 061-168.10.06.

- TITLE — Apollo CSM Pump Performance Test.

  BACKGROUND — The Apollo CSM pumps should be evaluated as backup to the 61C830069 water pumps.

  OBJECTIVE — Evaluate operation of Apollo CSM water pump assembly when serviced with PS 13240 Type IV fluid.

  RESULTS — Pumps operated satisfactorily throughout the test but were found to be susceptible to stalling when small air bubbles were at pump inlet. Reference TR 061-168.11.

- TITLE — Apollo PLSS Pump Performance Test.

  BACKGROUND — The Apollo PLSS pumps should be evaluated as backup for the 61C830069 water pumps.

  OBJECTIVE — Determine the performance characteristics of two series-connected Apollo PLSS pumps.

  RESULTS — The series operated pumps did not meet the minimum performance requirements of the Suit Coolant Loop. Reference TR 061-168.12.

- TITLE — Compatibility of deionized OWS water with 61C830069 Water Pump/SUS Fluid, Test.

  BACKGROUND — There was a requirement to show compatibility of the OWS servicing fluid with the SUS fluid and SUS loop components.

  OBJECTIVE — Verify that OWS potable water, when deionized, is compatible with PS 13240 Type VII solution, and will allow normal pump operation.

  RESULTS — The system operated satisfactorily. Reference TR 061-168.14.

B. Endurance Test - An endurance test, designated ET-1 and documented by report TR 061-068.35, was conducted to verify that system components would function properly during a complete mission.

The test hardware included more than 70 Airlock flight configuration components assembled into functional systems. The test was designed to load the components and make them perform under conditions expected during flight. The test followed the proposed Skylab mission plan which consisted of 3 Active Phases and 2 Orbital Storage Phases covering a real time period of 8 months.

During the formative stages of ET-1, the Suit Cooling System and ATM C&D/EREP cooling system used essentially the same hardware (pumps, check valves, filters, heat exchangers, and instrumentation), and the same cooling fluid additive (500 ppm Roccal, 2% by weight dipotassium hydrogen phosphate, and 0.2% by weight sodium borate). A single water loop was, therefore, used during ET-1 for assessing the endurance of both cooling water loops. The GFE liquid-gas separator was not included, nor were the GFE LSU, PCU, AND LCG. The aluminum internal surfaces of the GFE portion of the C&D loop were simulated, however, based on information available at that time.

All components initially assembled into the ET-1 EVA/IVA System functioned adequately except one of the cooling water pumps (P/N 61C830069-303, S/N 103), which failed to start during initial system checkout after servicing. Failure analysis disclosed that the problem was due to binding of the vane and rotor, caused by contamination introduced to the system during installation. After removal of the failed pump, testing was conducted with the alternate pump (S/N 117), which performed satisfactorily throughout ET-1. The alternate pump was operated 6966 hours out of a total exposure time of 10,368 hours and successfully underwent 14 on-off cycles. Post test tear-down and analysis of the pump showed negligible evidence of wear or corrosion. Germicidal effectiveness testing and analysis for products of corrosion were also conducted on cooling water samples and provided satisfactory results. Small amounts of nickel were noted however to be in solution in the cooling water, evidently originating from the nickel in the fins of the heat exchangers.

A coolant loop instability problem was experienced during ET-1, which resulted in coolant flowrate oscillations through the hot and cold ports of the 40°F temperature control valve and coolant temperature oscillations at the valve outlet. This problem was due to interaction of hot and cold flow in the EVA/IVA heat exchangers immediately upstream of the temperature control valve and is discussed further in Section 2.4.

2.6.3.2 Qualification Tests

Qualification tests and documentation are available for all Airlock components and systems. Test results are summarized in MDAC-E Report G499, Volume V.

2.6.3.3 Acceptance Tests

Acceptance tests were conducted to prove the delivered components and systems functioned properly.

A. Vendor Acceptance Test - An acceptance test had to be passed at the vendor's plant before shipment to the contractor. Acceptance test requirements were specified in the Acceptance Test Procedures (ATP's). These procedures were prepared by the vendor and approved by the contractor.
B. Pre-Installation Acceptance Test - A Pre-Installation Acceptance (PIA) Test had to be passed at the contractor's plant to prove that the hardware arrived in good condition prior to going into the crib which supplied parts for U-1, U-2 and spares. PIA test procedures were written by MDAC-E Servicing Engineering Department Report (SEDR) and basically included the same requirement as the ATP documents.

2.6.3.4 System Tests

System tests were conducted to verify that modules and systems operated properly. System test requirements were specified by SEDR.

A. Major Subassemblies - Major sub-assemblies were tested prior to installation during vehicle buildup. A tabulation of subassembly tests prior to installation is shown below:

SEDR NO.	SEDR TITLE	TEST MEDIA
D3-G61	Misc. Fluid Systems Functional Tests	O_2, N_2, Coolant
D3-M51	Misc. AM Fluid System Mfg. Tests	Miscellaneous
D3-G62	O_2 Supply Subassembly Funct. Test	O_2
D3-G63	O_2/N_2 Control Subassembly Test	O_2, N_2
D3-M48	Suit/Battery Cooling Module Funct.	Coolant, Water
D3-M48	Suit/Battery Cooling Module Mfg.	Coolant, Water

B. Systems - Final system and integrated acceptance test flow at the MDAC-E facility are depicted in Figure 2.6-19 for the Suit Cooling System and Figure 2.5-36 for the EVA/IVA O_2 supply system as part of the Gas System. Highlights of the testing as well as some of the problems and their solutions are presented below:

(1) O_2 Supply System

- During SEDR D3-N70, the O_2 and N_2 systems were leak checked, components were functionally checked, and system flow rates were verified. System instrumentation and caution and warning parameters were validated along with verification of DCS controlled functions.
- The O_2/N_2 system was revalidated during SEDR D3-E72 following rework of some of the module plumbing. A functional check of the EVA/IVA O_2 system using a suited crewman at sea level ambient conditions was also performed. Following SEDR D3-E72-1, the O_2/N_2 module was removed from the vehicle and the high pressure O_2 and N_2 regulators were replaced because of a configuration change to eliminate low temperature problems which had resulted in sense port icing and leakage during high flowrate conditions. Retest of the reworked module was accomplished by MPS AVE 117 prior to re-installation in the spacecraft.
- The O_2/N_2 module was reinstalled in the vehicle and the O_2/N_2 retest was accomplished by SEDR D3-E76, including verification of O_2 flowrate and AM/LSU pressure requirements. During retest the water tank N_2 pressurization system was inadvertently over-pressurized. The O_2/N_2 module was removed from the vehicle and

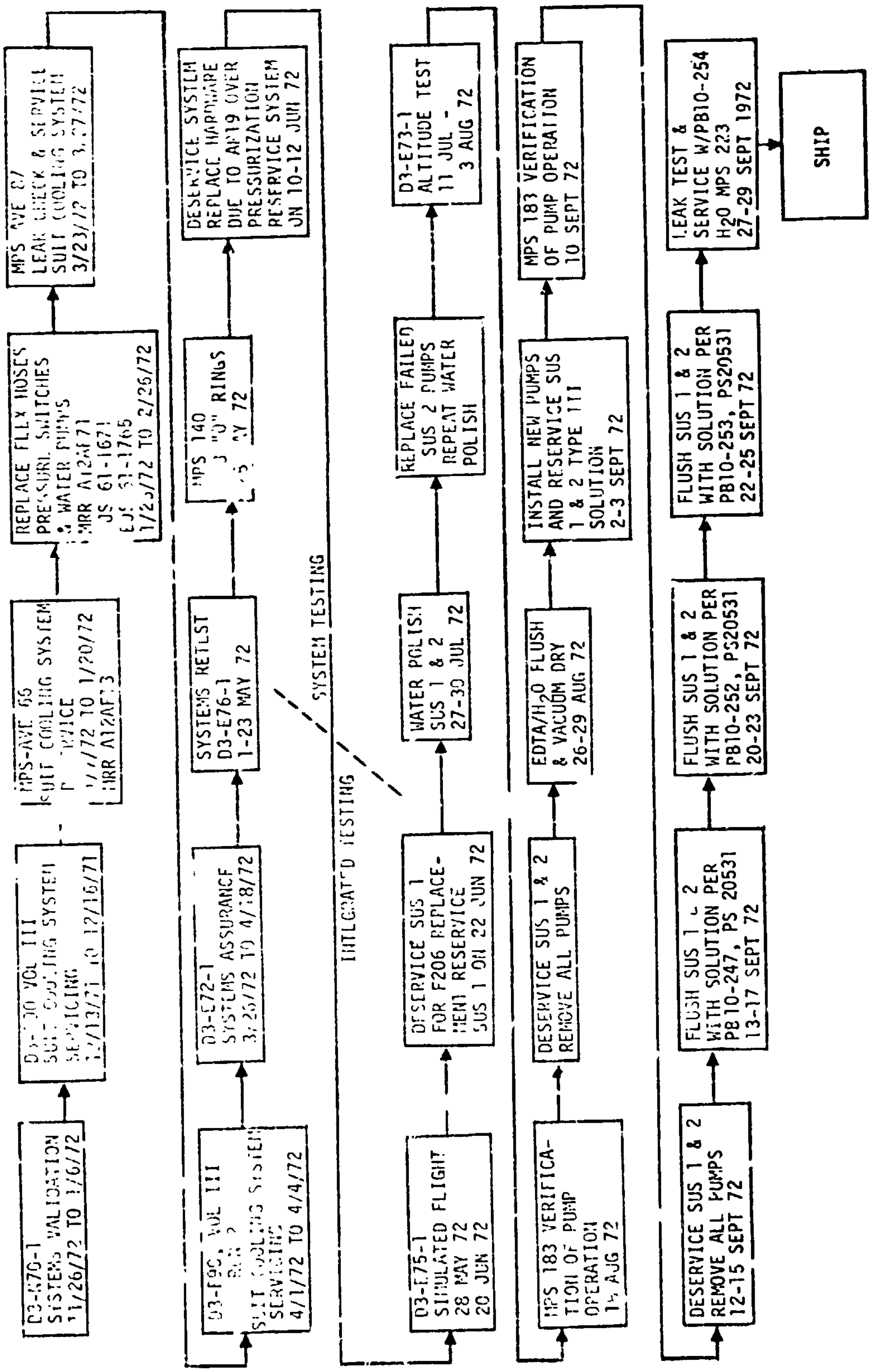


FIGURE 2.6-19 SUIT COOLING SYSTEM TEST HISTORY – MDAC-E

the damaged components replaced. Retest of the module after component replacement but prior to reinstallation was accomplished by MPS AVE 137. Retest after module re-installation was accomplished by AR 19 versus SEDR D3-E76.

(2) Suit Cooling System

- The Suit Umbilical Systems (SUS #1 and #2) were leak checked during Systems Validation (D3-N70) and then serviced per SEDR D3-F90-1, Vol. III. After servicing both systems were functionally checked to verify flowrates, system ΔP's and interface pressures.
- The systems were also operated during the thermal control systems checkout. During this period a water pump failure occurred. The problem was documented on MRR A12AFG1 and dispositioned for later removal. The fluid analysis of the system indicated an out of specification condition with respect to particulate contamination which was recorded on MRR A12AF13. At the conclusion of SEDR D3-N70 the systems were deserviced per MPS 66 and flushed and vacuum dried per MRR A12AF13. While the systems were deserviced, flexible hoses and pressure switches were changed out per EJS 61-1671 and EJS 16-1765. The suit systems were leak checked per MPS 87 after completion of all rework and were then reserviced by SEDR D3-F90-1, Vol. II.
- The systems were functionally checked during Systems Assurance Test (SEDR D3-E72-1) with a suited subject in the loop to verify temperature and flow capabilities of both Suit Umbilical Systems. The systems were operated and verified during a systems retest, SEDR D3-E76-1. During the course of testing, it was discovered that some of the quick disconnects had been cleaned with freon after assembly which could wash lubrication from the "O" ring seals. To prevent any leakage problems, the "O" rings in the quick disconnects on the vehicle were lubricated per MPS 140. During D3-E76 an inadvertent overpressurization occurred on the SUS loops. The systems were deserviced and any hardware that could have seen pressures in excess of proof were replaced. Leak tests and reservicing of the systems were then performed in preparation for SEDR D3-E75-1, simulated flight testing.

2.6.3.5 Integrated Tests

Integrated tests were conducted to verify the vehicle was ready for flight both before it left the factory for the launch site and after it arrived at the Kennedy Space Center (KSC).

A. Factory Tests - The tests conducted by MDAC-E in St. Louis are summarized below. Integrated test requirements were specified by SEDR.

(1) O_2 Supply System - The O_2 supply system was not functionally operated nor any special testing conducted on it during Simulated Flight, SEDR D3-E75. The system was operated during the Manned Altitude Chamber Test, SEDR D3-E73, to supply O_2 to the crewmen via the AM/LSU interfaces for simulated EVA/IVA operations.

(2) Suit Cooling System

- The SUS #1 water flowmeter (F206) ceased operation during SEDR D3-E75 simulated flight testing. SUS #1 was deserviced to replace the flowmeter. The loop was then reserviced and operated successfully. A requirement was initiated to filter the SUS loops to preclude clogging of the liquid/gas separator. During this filtration process (water polishing) the pumps in the SUS #2 loop failed to operate. SUS-2 was deserviced, pumps replaced, re-serviced and the system polished in preparation for Altitude Chamber testing, SEDR D3-E73-1.
- SUS loops were utilized during D3-E73-1 and monitored for proper operation with the Astronauts on the loop in liquid cooled garments. No problems occurred in the loops during chamber de-pressurization or EVA/IVA activities.
- After the altitude chamber test, MPS 183 was initiated to operate the water pump daily to verify operation. The pumps failed to operate. A review of the failure indicated potential problems associated with the additives used in the water solution. The systems were flushed with an EDTA/H_2O solution. The pumps again failed to operate during the MPS 183 verification. Further investigation revealed that the nickel in the systems stainless steel heat exchangers was being attacked by the additives in the solution causing a hard precipitate to form which caused the pumps to be inoperative. The solution to this problem

consisted of installing new pumps with greater vane to rotor clearance, and new additives for the water solutions (solution per PS 13240, Type VII). The new pumps could not be ready for installation prior to U-1 delivery to KSC, however, the systems were treated to remove the precipitate and passivated to prevent recurrence. The flushing agent was an EDTA solution per PB 10-247 versus PS 20531, and an ammonium hydroxide solution per PB 10-252 versus PS 20531. The heat exchangers were changed out for units that had never seen additives. The entire system was passivated with a sodium chromate solution per PB 10-253 vs. PS 20531.

- The systems were then serviced with low conductivity MMS 606 water per PB 10-254 vs. PS 20531 and prepared for delivery to the launch site.

B. KSC Tests - Results of testing at KSC together with a comparison with Factory test results are presented in Figures 2.6-20 through 2.6-23. Launch site test requirements were specified in MDC Report E0122, Specification and Criteria at KSC for AM/MDA Test and Checkout Requirements; and KSC Report KS2001, Test and Checkout Plan.

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
1. System Leak Test						
• Leak Test of SUS 1	• 1.6 SCCM N_2 @ 31 psig	D3-N70-1	0.18 SCCM	Tank replacement MOD KIT 65	Verified	Connections of all components replaced were verified by mass spec using He. No detectable leakage allowed above normal background.
• Leak Test of SUS 2	• 1.6 SCCM N_2 @ 31 psig	D3-N70-1	1.45 SCCM			
2. System water flush and Servicing	• Fill system with water (PS 13240 TYPE VII). Withdraw 4.0 ± 0.2 lbs. prior to mechanical closeout.	D3-F90-1 Vol. III	N/A	KS-0016 TPS'S AM 108-220 -222 -226	Verified	System was flushed prior to KS-0045. Reflush also after tank replacement.
	• Polish system to meet following requirements/ 100 ml of fluid:		Verified	AM 108-227	Verified	
	• Below 2 Microns By Wt* 2-10 Microns 800 10-25 Microns 712 25-50 Microns 126 50-100 Microns 22 100-500 Microns 4 Above 500 Microns 0 *Total filterable solids (0.45 micron disk only) not to exceed 0.5 mg/100 ml.					
3. Water/Gas Separator Fit Check	• Acceptable installation of water/gas separator to suit cooling loop (three flight units).	D3-N70 Seq 14	Verified*	KS-0016	Verified	*S/N 1005, 1006, 1007 DR AM1-07-0030. H_2O/Gas sys outlet wouldn't mate. Mod Kit 4 replaced coupler and caps.
4. Pump Operation		D3-N70-1		KM-0003		
• Operation of SUS 1 primary and secondary pumps from EVA #1 panel (317).	• Audible ON/OFF response to manual switching.		Verified		Verified	SUS 2 pumps failed to start. MOD KIT 46 changed pumps and fluid in SUS 1 and SUS 2
	• EVA #1 light remains OUT during system operation.		Verified		Verified	See KS-0045 results after installation of new pumps. (DR AM1-07-0062 and AM1-07-0121)
	• Pump flowrate = 200 lb/hr min and pump ΔP = 16.0 $^{+10.2}_{-6.2}$ psid.		(P)261 lb/hr @ 18.5 psid (S)261 lb/hr @ 18.5 psid		273 lb/hr max. 268 lb/hr min. 20.6 psid max. 18.9 psid min.	Pump operation conducted with simulated GFE ΔP of 12 $^{+0.1}_{-0}$ @ 250 lb/hr.
• Operation of SUS 1 primary and secondary pumps from IVA panel (217).	• Audible ON/OFF response to manual switching.		Verified		Verified	
	• EVA #1 light remains OUT during system operation.		Verified		Verified	
	• Pump flowrate = 200 lb/hr min. and pump ΔP = 16.0 $^{+10.2}_{-6.2}$ psid.		(P)261 lb/hr @ 18.4 psid (S)261 lb/hr @ 18.7 psid		274 lb/hr max. 268 lb/hr min.	

FIGURE 2.6-20 SUIT COOLING SYSTEM REQUIREMENT VERIFICATION (SHEET 1 OF 2)

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
4. Pump Operation (Cont'd)						
• Operation of SUS 2 primary and secondary pumps from EVA #2 panel (323)	• Audible ON/OFF response to manual switching.	03-N70-1	Verified	KM-0003	Verified	
	• EVA #2 light remains OUT during system operation.	↓	Verified	↓	Verified	
	• Pump flowrate = 200 lb/hr min and pump ΔP = 16.0 $^{+10.2}_{-6.2}$ psid.	↓	(P)255.8 lb/hr @ 19.2 psid. (S)249.8 lb/hr @ 18.5 psid	↓	277 lb/hr max. 0 lb/hr min. 39.9 psid max. 18.3 psid min.	IDR 016 (upgraded to DR AM 1-07-0062) reads zero when should read flow. IDR 024 - ΔP xducer (D202) read 18.1 psid w/o pump operation. Replaced and retested OK per DR AM 1-07-0090.
• Operation of SUS 2 primary and secondary pumps from IVA panel (217).	• Audible ON/OFF response to manual switching.	↓	Verified	↓	Verified	
	• EVA #2 light remains OUT during system operation.	↓	Verified	↓	Verified	
	• Pump flowrate = 200 lb/hr min, and Pump ΔP = 16.0 $^{+10.2}_{-6.2}$ psid.	↓	(P)259 lb/hr 18.5 psid (S)254 lb/hr 17.9 psid	↓	285 lb/hr max. 272 lb/hr min. ΔP (off scale)	IDR 024 (upgraded as stated above).

RETEST @ KSC AFTER PUMP AND FLUID CHANGE

DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT			REMARKS
4. Pump Operation (Cont'd)		KS-0045				
• Operation of SUS 1 primary and secondary pumps from EVA #1 panel (317).	Pump flowrate = 200 lb/hr min. and Pump ΔP = 16.0 $^{+10.2}_{-6.2}$ psid.			Run 1	Run 2	Run 1: LCG SIM connected to Pnl 317 SUS 1 QD's. Run 2: LCG SIM on Pnl 323 SUS 1.
		32-013/024	Pri ΔP	20.7	19.9 psid	
		013/024	Pri Flow	281	272 lb/hr	
		32-017/028	Sec ΔP	18.3	19.0 psid	
		017/028	Sec Flow	270	266 lb/hr	
		013/028	Inlet Temp	68.2	67.9 Deg F	
		013/028	Outlet Temp	67.7	67.9 Deg F	
• Operation of SUS 1 primary and secondary pumps from IVA panel (217).	Pump flowrate = 200 lb/hr min. and pump ΔP = 16.0 $^{+10.2}_{-6.2}$ psid.	32-033	Pri ΔP	19.4 psid		LCG SIM on Pnl 217 SUS 1
		033	Pri Flow	273 lb/hr		
		034	Sec ΔP	18.6 psid		
		034	Sec Flow	267 lb/hr		
		033	Inlet Temp	67.5 Deg F		
		033	Outlet Temp	67.7 Deg F		
• Operation of SUS 2 primary and secondary pumps from EVA #2 panel (323).	Pump flowrate = 200 lb/hr min. and pump ΔP = 16.0 $^{+10.2}_{-6.2}$ psid.			Run 1	Run 2	Run 1: LCG SIM on Pnl 317 SUS 2 Run 2: LCG SIM on Pnl 323 SUS 2 IDR 097 (KS-0009): Flowmeter failed to operate one time. Xferred to DR AM 1-08-0571 and replaced.
		33-012/023	Pri ΔP	20.3	20.3 psid	
		012/023	Pri Flow	280	287 lb/hr	
		016/027	Sec ΔP	21.1	19.1 psid	
		016/027	Sec Flow	287	280 lb/hr	
		012/027	Inlet Temp	63.1	60.3 Deg F	
		012/027	Outlet Temp	60.2	59.5 Deg F	
• Operation of SUS 2 primary and secondary pumps from IVA panel (217).	Pump flowrate = 200 lb/hr min. and pump ΔP = 16.0 $^{+10.2}_{-6.2}$ psid.	33-031	Pri ΔP	21.0 psid		LCG SIM on Pnl 217 SUS 2
		031	Pri Flow	279 lb/hr		
		032	Sec ΔP	20.3 psid		
		032	Sec Flow	284 lb/hr		
		031	Inlet Temp	58.8 Deg F		
		031	Outlet Temp	59.5 Deg F		

FIGURE 2.6-20 SUIT COOLING SYSTEM REQUIREMENT VERIFICATION (SHEET 2 OF 2)

REQUIREMENTS		VERIFICATION				COMMENTS/REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
1. EVA outlet flow test. Apply 400 $^{+5}_{-0}$ psia O_2 at GSE 504 and provide a reference pressure of 5 ± 0.2 psia at GSE 547. Perform the following at each EVA panel:		D3-E76-1 Seq 7	See EVA/IVA Flow test data	KM0003	GSE 504: 400.7 to 401.7 psia GSE 547: 4.8 psia	DR #AM 1-07-0086 tests used GN_2 instead of O_2. Equivalent flowrates were 8.34 + 0.46, -0 lb/hr. and 12.69 + 0.46, -0 lb. hr.
• With 9.0 $^{+0.5}_{-0}$ lb/hr O_2 from QD #2 and 13.7 $\pm$ 0.5 lb/hr O_2 from QD #1, measure pressure at QD #1 and at GSE 502.	• 65 psia minimum at QD #1 • 125 ±10 psia at GSE 502. • 65 psia min. @ QD #2 @ Factory			40-1072 40-116	@ Pnl 317: QD 1-99.7 psia GSE 502: 126.7 @ Pnl 323: QD 1-98.7 GSE 502: 127.2 psia	@ Pnl 317: 13.1 lb/hr @ QD #1 8.56 lb/hr @ QD #2 @ Pnl 323: 12.9 lb/hr @ QD #1 8.65 lb/hr @ QD #2
• With 9.0 $^{+0.5}_{-0}$ lb/hr O_2 from QD #1 and 13.7 $^{+0.5}_{-0}$ lb/hr O_2 from QD #2, measure pressure at QD #2 and at GSE 502.	• 65 psia minimum at QD #2 • 125 ± 10 psia at GSE 502. • 65 psia min @ QD #1 @ Factory			40-112 40-120	@ Pnl 317: QD 2-100.8 psia GSE 502: 128.4 psia @ Pnl 323: QD 2-101.45 GSE 502 - 128.2 psia	@ Pnl 317: 13.1 lb/hr @ QD #2 8.65 lb/hr @ QD #1 @ Pnl 323: 12.9 lb/hr @ QD #2 8.5 lb/hr @ QD #1
2. IVA outlet flow • Apply 500 $^{+5}_{-0}$ psia O_2 at GSE 504 and provide a reference pressure of 5±0.2 psia at GSE 547. Perform the following:		D3-E76-1 Seq 7	See EVA/IVA Flow Test Data	KM-0003 Seq 40	GSE 504: 500-501 psia GSE 547: 4.8-5.1 psia	
• With 13.7 $^{+0.5}_{-0}$ lb/hr O_2 from QD #1, 9.0 $^{+0.5}_{-0}$ lb/hr from QD #2, and 9.0 $\pm$ 0.5 lb/hr from QD #3, measure pressure at QD #1 and at GSE 502.	•55 psia minimum at Pnl 217 QD #1 •125 ±10 psia at GSE 502 •65 psia min @ Pnl 217 QD #2 and QD #3 @ Factory			40-102	@ Pnl 217: QD 1 - 84.76 psia Flow - 12.9 lb/hr. GSE 502: 125.7 psia	@ Pnl 217: 8.5 lb/hr @ QD #2 8.53 lb/hr @ QD #3
• With 9.0 $^{+0.5}_{-0}$ lb/hr O_2 from QD #1, 13.7 $^{+0.5}_{-0}$ lb/hr from QD #2, 9.0 $^{+0.5}_{-0}$ lb/hr from QD #3, measure pressure at QD #2 and at GSE 502.	•55 psia minimum at Pnl 217 QD #2 •125 ±10 psia at GSE 502. •65 psia min @ Pnl 217 QD #1 and QD #3 @ Factory			40-098	@ Pnl 217: QD 2 - 87.7 psia Flow - 12.9 lb/hr GSE 502: 126.2 psia	@ Pnl 217: 8.65 lb/hr @ QD #1 8.53 lb/hr @ QD #3
• With 9.0 $^{+0.5}_{-0}$ lb/hr O_2 from QD #1, 9.0 $^{+0.5}_{-0}$ lb/hr from QD #2, and 13.7 $^{+0.5}_{-0}$ lb/hr from QD #3, measure pressure at QD #3 and at GSE 502.	• 55 psia minimum at QD #3 • 125 ± 10 psia at GSE 502. • 65 psia min @ Pnl 217 QD #1 and QD #2 @ Factory.			40-106	@ Pnl 217: QD 3: 84.76 psia Flow: 12.9 lb/hr GSE 502: 125.2 psia	@ Pnl 217: 8.5 lb/hr @ QD #1 8.53 lb/hr @ QD #2

FIGURE 2.6-21 EVA/IVA O_2 SUPPLY SYSTEM REQUIREMENT VERIFICATION (SHEET 1 OF 2)

SEDR D3-E76 EVA/IVA O_2 (N_2) TEST RESULTS

SEE FIGURE 2.6-22 SCHEMATIC FOR REFERENCE

SEQ.	TEST NO.	REGULATOR	N_2 FLOWRATES ~ LB/HR				PRESSURES ~ PSIG (1)					
			FM1	FM2	FM3	FM4	G2	G16	D204	G4	G15	G18
			Sus1/217	Sus1/217	Sus2/217	OWS FILL	GSE504	GSE502	TM 1	Sus1/217	Sus1/217	Sus2/217
07-343	1	A	7.5	6.65	7.0	0	285	49.5	-	7.5	3.5	7.3
	2	A	12.7	8.34	8.34		420	97	-	55.2	61.5	63.8
	3	A	8.34	12.7	8.34		420	97	-	61.5	52.8	63
	4	A	8.34	8.34	12.7		419.5	97	-	63	62.9	54.2
07-344	5	A&B	12.7	8.34	8.34		285	107	-	71.7	76	78
	6	A&B	8.34	12.7	8.34		285	101	114.1	67	59	69
	7	A&B	8.34	8.34	12.7		285	112	125.2	84	83.5	78.5
	7A	A&B	10.4	9.4	10.5		285	91	103.9	50.5	50.5	50.5
	7B	A&B	13.5	12.4	13.8		485	114	126	50.5	50.5	50.5
	8	A&B	8.2	7.4	7.5	7.3	285	70	82.6	9	5	9.5
07-347	9	N/A	12.7	8.34	8.34	0	N/A	95.5	107.8	50.5	57.5	60
	10	N/A	8.34	12.7	8.34			95.5	105.5	56	44.5	58.5
	11	N/A	8.34	8.34	12.7			95.5	106.3	58.5	57.5	48.5
			Sus1/317	Sus2/323	Sus2/217		G2	G16	D024	Sus1/317	Sus2/323	Sus2/217
07-351	12	N/A	12.7	8.34	8.34	0	N/A	95.5	106.3	47.5	56.5	59.5
	13	N/A	8.34	12.7	8.34			95.5	106.3	57.5	43.8	59
	14	N/A	8.34	8.34	12.7			95.5	107.1	59.0	58	50
	15	N/A	8.34	8.34	14.0			95.5	104.7	53.5	52.5	21
	16	N/A	8.34	12.7	12.1			95.5	104.7	47.5	10.5	30
07-355	17	A	7.1	6.6	7.1		285	48.5	-	7	3	7
	18	A	12.7	8.34	8.34		450	113	126	79	84.[illegible]	86
	19	A	8.34	12.7	8.34		442	116.5	129.9	90	82.[illegible]	91
	20	A	8.34	8.34	12.7		442	112.5	126	83.5	84.5	80
	21	A	8.34	8.34	14.1		441	92	104.7	53	52.5	25
	22	A	8.34	12.7	9.8		442	89	101.5	48.5	10	46.5
07-356	23	A&B	12.7	8.34	8.34		285	105	118.1	67.5	74.2	76
	24	A&B	12.7	8.34	8.34		280	96	-	54.5	63	64
	25	A&B	8.34	12.7	8.34		285	104	-	73.5	64.5	75
	26	A&B	8.34	8.34	12.7		285	110	123.6	83	82	77
07-358	27	N/A	0	7.41	0		N/A	120.5	134.6	118	115	N/A
	28		0	12.51				100.5	113.4	94.5	83	
	29		0	18.6				100.5	-	87	18	
	30		8.34	8.34				100.5	-	87	86.5	
	31		8.34	12.98				100.5	-	80.5	72	
	32		8.34	17.2				100.5	-	72.5	19.5	
	33		12.7	12.7				100.5	-	67	66	
	34		12.97	15.8				100.5	-	54.5	14	
	35		15.2	15.1				100.5	-	23.5	13	

(1) Ambient Pressure = 14.5 psia; D204 Reads psia, 5 psia ref. press. @ 120 psig Reg.

FIGURE 2.6-21 EVA/IVA O_2 SUPPLY SYSTEM REQUIREMENT VERIFICATION (SHEET 2 OF 2)

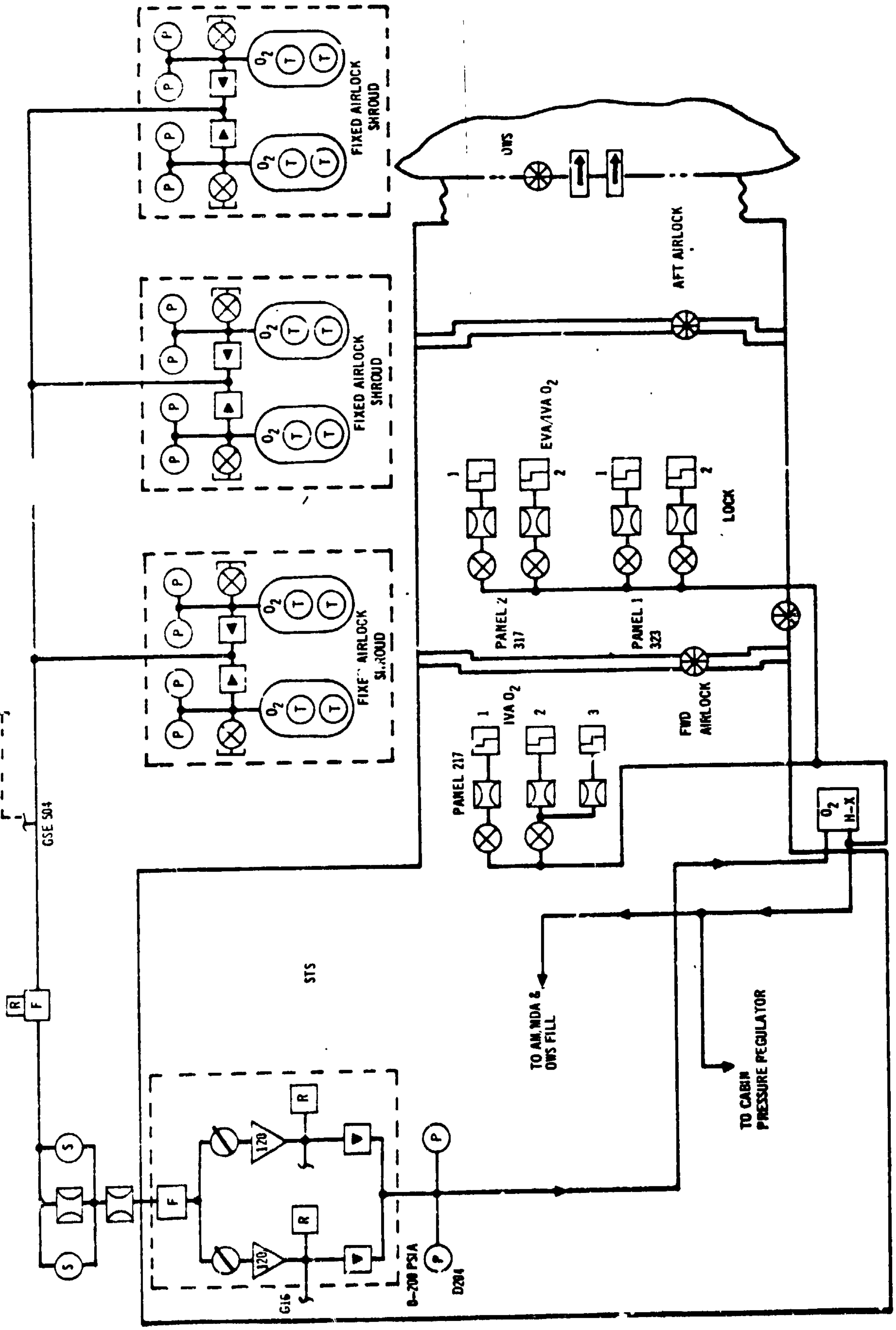


FIGURE 2.6-22 EVA/IVA GAS DELIVERY SYSTEM

REQUIREMENTS		VERIFICATION				COMMENTS/ REMARKS
		FACTORY		KSC		
DESCRIPTION	SPECIFICATION	PROCEDURE	MEASUREMENT	PROCEDURE	MEASUREMENT	
1. EVA Valve (1)	Valve operates manually (visual indication).	D3-N70-1	Verified	45-153	No. 318	Verified
2. AM Internal Hatch Valves (2)	Each valve operates manually (visual indication)	D3-N70-1	Verified	45-097, 100 45-041, 049	No. 311 No. 325	Verified

FIGURE 2.6-23 EVA LOCK PRESSURE CONTROL VALVE REQUIREMENT VERIFICATION

2.6.3.6 Mission Support Tests

Mission Support Tests were conducted during the flight of U-1 using the Static Test Unit (STU). A summary of special tests associated with EVA/IVA operations which were performed with STU is presented below. Test details as well as descriptions of STU are presented in the ECS/TCS Skylab Test Unit Report No. TR 061-068.99.

- TITLE — SUS Loop Operation Using Two 60 Foot LSU's In Series.

  BACKGROUND — U-1 crew might have used two 60-foot umbilicals connected in series to permit additional distance for repair work during EVA.

  OBJECTIVE — Determine SUS loop operation when an additional Life Support Umbilical (LSU) is installed in the SUS loop. Reference TR 061-015-600.01.

  RESULTS — SUS water pump performance was satisfactory when the flow restriction of another LSU was added to the SUS loop. Reference TM 252:653.

- TITLE — SUS Loop Cooling Rates With BY-PASS/EVA Valve In BY-PASS Position.

  BACKGROUND — Coolant loop temperature control valve "B" stuck when U-1 crew positioned the BY-PASS/EVA valve in the EVA position. Remaining EVA periods were planned with the valve in BY-PASS.

  OBJECTIVE — Determine cooling effectiveness of SUS loop No. 1 heat exchanger. Reference TR 061-015-600.08.

  RESULTS — Cooling effectiveness of the SUS loop was determined for several imposed heat loads with the BY-PASS/EVA valve in the BY-PASS position. Reference TM 252:714.

- TITLE — Life Support Umbilical (LSU) Pressure Test

  BACKGROUND — The U-1 LSU might have remained serviced for long periods.

  OBJECTIVE — Determine if a LSU can withstand proof pressure following a long term service condition. Reference TR 061-015-600.32.

  RESULTS — An LSU had been serviced with fluid for four months. The unit sustained proof pressure of 74 psig without damage. Reference TM 252:703.

- TITLE SUS Water Pump Temperature Test

BACKGROUND U-1 SUS loop operation might have been required without cooling from the coolant loop.

OBJECTIVE Determine if the SUS water pump will operate with no coolant flow through the SUS heat exchanger for a 90-minute period without exceeding temperature limits. Reference TR 061-015-600.34.

RESULTS The SUS water pump was operated for 90 minutes while pump body temperature stabilized at 81°F. Reference TM 252:701.

2.6.4 Mission Performance

Airlock Module systems provided support for twelve EVA/IVA operations ranging up to a record duration of seven hours (exclusive of EVA preps or post-EVA activity) on DOY 359. Oxygen flow to suited crewmen was completely normal on each occasion and was utilized for cooling of the crewmen during the final EVA on SL-3 (DOY 265) due to shut-down of the primary coolant loop. Satisfactory water cooling was supplied for all other EVA operations with up to three crewmen on one suit cooling system. Operation of the lock compartment was accomplished normally although depressurization rates were decreased as ice formed from moisture in the gas collected on the protective screen over the depress valve vent port.

2.6.4.1 Oxygen Supply

Oxygen was supplied to the LSU at normal pressures and temperatures during all suited operations. Regulated O_2 pressures during these periods ranged between 122 and 127 psia while O_2 temperatures were normally controlled between 50 and 60°F by the heat exchanger interfacing with AM coolant loops. Although not a planned operation, high O_2 flowrates were utilized for cooling of EVA crewmen during the DOY 265 EVA with satisfactory results for existing heat loads. Average metabolic rates during this period were reported to be 790 and 1060 Btu/hr for the two EVA crewmen. No problems were identified with the EVA/IVA O_2 supply system.

2.6.4.2 Suit Cooling System

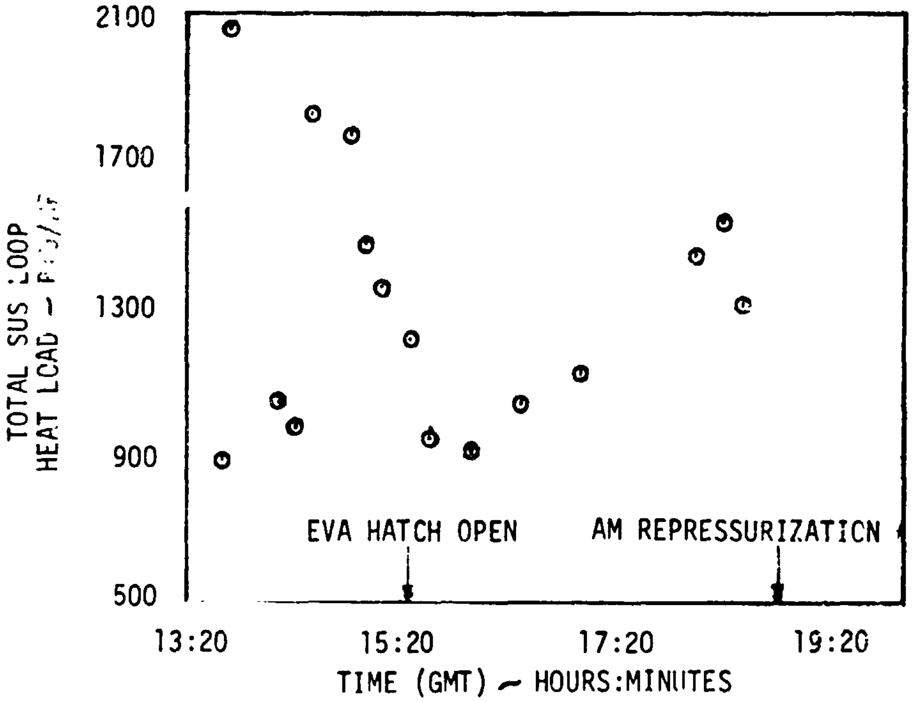
Suit cooling systems were successfully activated on 29 separate occasions as shown in Figure 2.6-24 and performed in a normal manner at all times. Of this total, 11 operations were in direct support of EVA/IVA, two were to provide heat into the secondary coolant loop following a temperature control valve discrepancy, and the remainder were for normal systems checkout. Water flowrates of 225 to 296 lb/hr were obtained with SUS 1 while SUS 2 provided flowrates between 265 and 300 lb/hr depending on system configuration. Water temperatures and flow durations are also shown in Figure 2.6-24.

Suit cooling system performance, in terms of water delivery temperature and system heat loads as a function of time, is shown in Figures 2.6-25 and 2.6-26 for two typical EVA operations. The EVA on DOY 158 was conducted with the coolant loop diverter valve in the EVA position and resulted in water delivery temperatures

DOY	EVENT	SUS NO.	HX COOLANT FLOW	COOLANT PUMPS	WATER SUPPLY TEMP (°F)	FLOW DURATION (HR:MIN)	REMARKS
157	PUMP CYCLING	1 2	BYPASS BYPASS	0 1	48.7 - 60.0	1:30 1:30	PRI LOOP OFF
158	EVA	1	EVA	1	33.0 - 48.7	:50 +	SUS 1 TURNED OFF WHEN PRI LOOP TCV-B STUCK COLD
		2	EVA	1	43.6 - 47.2	6:10	SEC LOOP TCV-B STUCK IN COLD POSITION DURING EVA
159-162	SEC LOOP WARMUP	2	BYPASS	1	56.0 - 59.0	84:15	
162	TCV-B MODULATION TEST	1	BYPASS	1	46.5 - 58.0	:30	
163	SEC LOOP WARMUP	2	BYPASS	0	58.0 - 92.0	2:20	SEC LOOP OFF
165	TCV-B MODULATION TEST	2	BYPASS	1	52.1 - 54.0	:30	
170	EVA	1	BYPASS	2	51.5 - 59.6	3:35	
217	PUMP CYCLING	1	BYPASS	1	49.0 - 52.5	:30	
218	EVA	1	BYPASS	2	NO DATA	9:00	
229	IVA (M509-3)	1	BYPASS	1	NO DATA	3:35	
230	PUMP CYCLING	1	BYPASS	1	49.5 - 52.5	1:30	
233	PUMP CYCLING	2	BYPASS	1	49.0 - 50.5	1:40	
236	EVA	2	BYPASS	2	52.5 - 57	7:10	
240	PUMP CYCLING	1	BYPASS	0	62.5 - 64.0	:17	PRI LOOP OFF
258	PUMP CYCLING	1	BYPASS	0	NO DATA	:10	PRI LOOP OFF
325	PUMP CYCLING	1	BYPASS	1	51.1 - 57.2	:30	
326	EVA	1	BYPASS	2	51.5 - 58.2	9:10	
341	PUMP CYCLING	1	BYPASS	0	53.5 - 63.0	:10	PRI LOOP OFF
350	PUMP CYCLING	1	BYPASS	0	55.2 - 65.5	:10	PRI LOOP OFF
359	EVA	1	BYPASS	2	51.5 - 55.0	9:20	
362	PUMP CYCLING	1	BYPASS	1	NO DATA	:30	OPERATION FOLLOWING SUS 1 AND LSU/PCU SERVICING
363	EVA	1	BYPASS	2	52 - 55	5:55	
008	PUMP CYCLING	1	BYPASS	1	50 - 54	:30	
018	PUMP CYCLING	1	BYPASS	2	50 - 55	:20	
028	PUMP CYCLING	1	BYPASS	1	50 - 53	:15	
034	EVA	1 2	BYPASS BYPASS	2 1	51 - 54 47 - 49	8:35 1:25	STANDBY OPERATION

FIGURE 2.6-24 SUMMARY OF SUIT COOLING SYSTEM OPERATION

1. TWO CREWMEN CONNECTED TO SUS NO. 2
2. COOLANT DIVERTER VALVE IN EVA POSITION
3. ONE COOLANT PUMP OPERATING



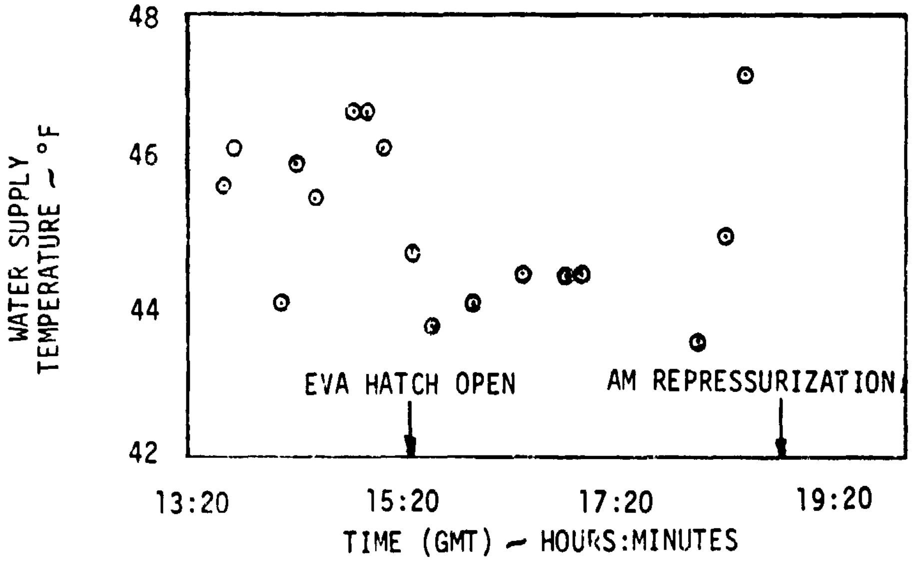
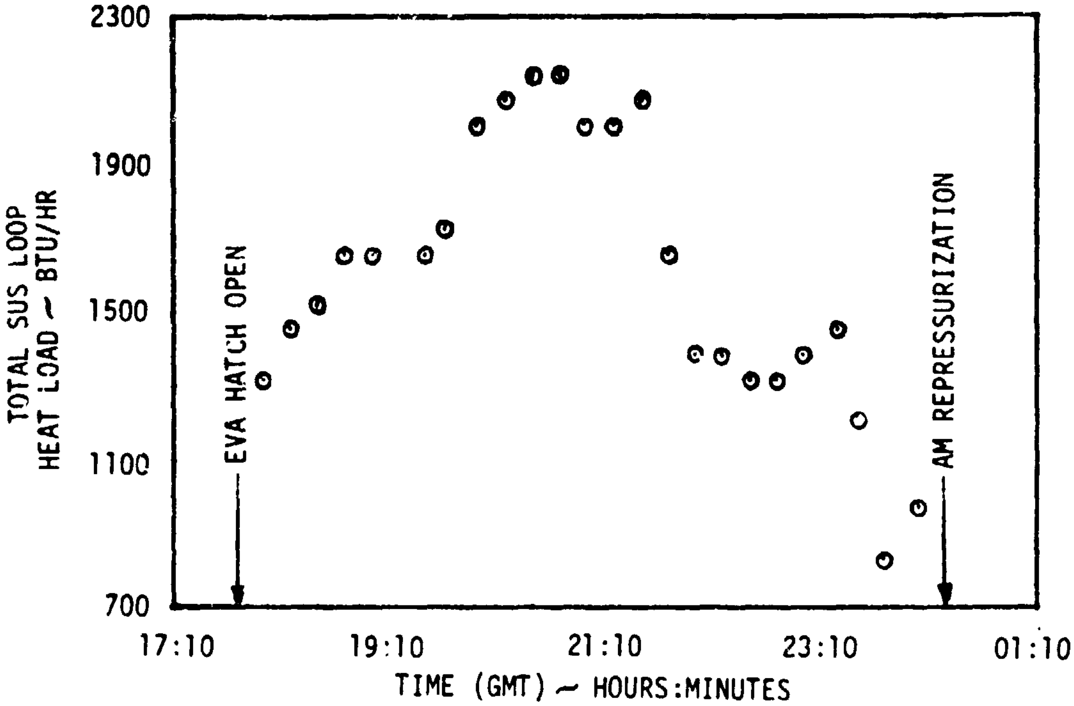


FIGURE 2.6-25 SUIT COOLING SYSTEM PERFORMANCE – DOY 158 EVA

1. THREE CREWMEN CONNECTED TO SUS NO. 1
2. COOLANT DIVERTER VALVE IN BYPASS POSITION
3. TWO COOLANT PUMPS OPERATING



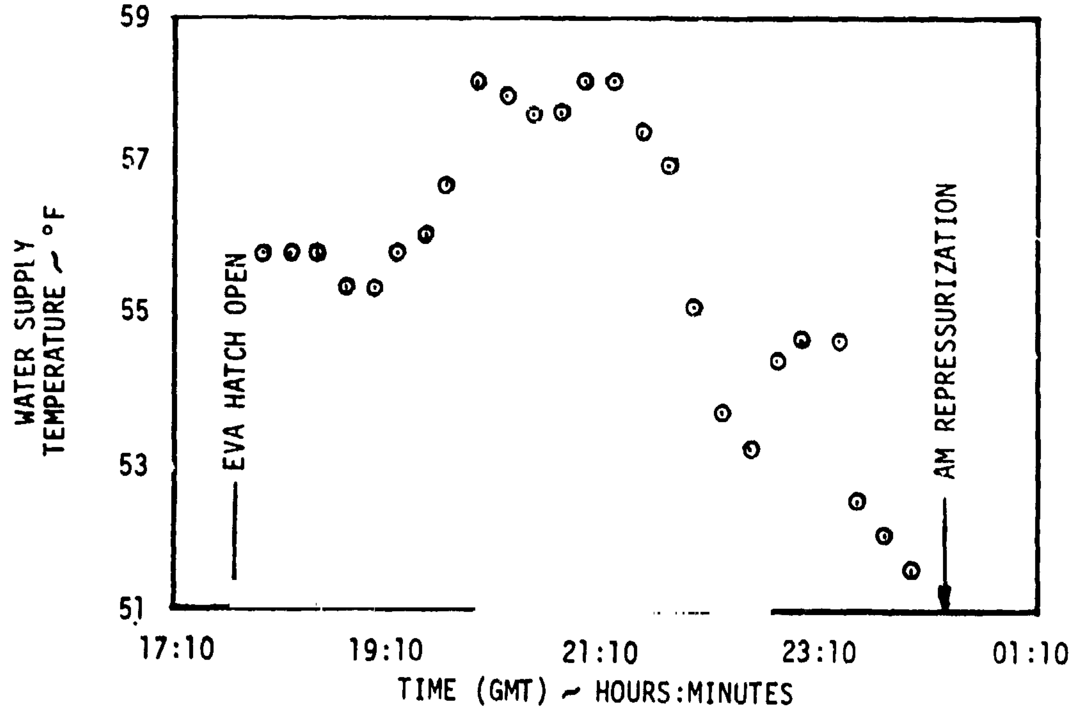


FIGURE 2.6-26 SUIT COOLING SYSTEM PERFORMANCE - DOY 326 EVA

within the required range. The EVA on DOY 326 was conducted with the diverter valve in the BYPASS position, resulting in somewhat higher water delivery temperatures but sufficient for effective cooling for all three astronauts on the same SUS loop. A survey of the Skylab EVA's showed that the astronaut heat loads were considerably below the maximum values required for design. Also, no situation was encountered which resulted in a negative heat load on the system.

During EVA on DOY 359, a water leak at the GFE LSU/PCU composite connector resulted in depletion of water in the SUS 1 reservoir. However, the EVA was completed without difficulty.

During EVA on DOY 034, a water leak occurred for the second time at the LSU/PCU composite connector. Crew action was taken to minimize the leakage rate and the EVA was completed with SUS 1. However, SUS 2 was activated and operated normally in a standby mode during the latter portion of EVA.

Gas removal from water in the suit cooling systems was apparently normal throughout the mission and the liquid-gas separator assembly installed at launch was never replaced.

During EVA on DOY 158, water in SUS 1 was exposed to subfreezing temperatures in the heat exchanger due to AM primary coolant flow at temperatures below 0°F when the downstream temperature control valve (TCV-B) stuck in a cold position. Although water was initially flowing, measured water temperatures did approach the freezing point and all indications were that system water flow was lost. It is therefore suspected that freezing in the affected heat exchanger was encountered. Similarly, during SL-1 operation in the abnormally cold vehicle attitude, water line temperatures as low as 33.7°F were recorded on DOY 145; however, freezing temperatures could have existed in other areas where instrumentation was not available.

2.6.4.3 In-Flight Water Servicing/Deservicing

LSU/PCU's were successfully deserviced on SL-3 just prior to the DOY 265 EVA since water flow would not be provided through these components and such action was desired by flight controllers to prevent the possibility of localized freezing. Required components were again serviced prior to the first EVA on SL-4. Following loss of water in the SUS 1 reservoir on DOY 359 as discussed in Section 2.6.4.2, the SUS 1 loop and an alternate LSU/PCU were serviced on DOY 361 in preparation for EVA on DOY 363. Servicing of SUS 1 after leakage on DOY 034 was not required since additional system usage was not planned. No problems were encountered during any of the servicing/deservicing operations.

2.6.4.4 Lock Operation

The Airlock Module lock and aft compartments were successfully depressurized and repressurized during the performance of EVA on nine occasions. A typical vent and pressurization profile is shown in Figure 2.6-27. The only discrepancy reported involved the formation of ice from moisture in the lock compartment atmosphere on the screen over the depress valve opening during venting. Reports indicate that SL-2 crewmen removed ice from the screen to speed up the venting process. A second removable screen was supplied on SL-3 which permitted ready removal of the ice buildup on the second screen when pressure dropped below 1 psia thereby exposing the clean original screen for completion of venting. No further difficulties were encountered after use of the second screen was initiated.

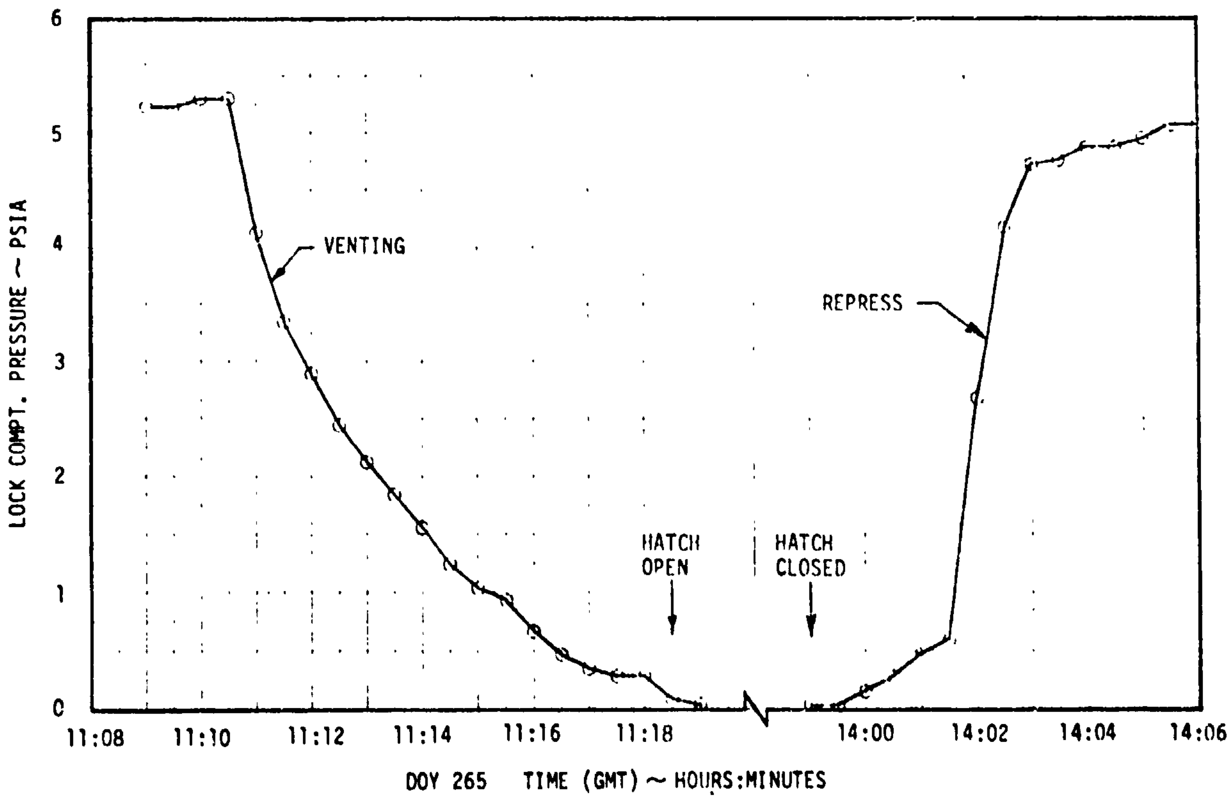


FIGURE 2.6-27 TYPICAL EVA VENT AND PRESSURIZATION PROFILE

2.6.5 Development Problems

2.6.5.1 EVA O_2 Supply -

Problems were encountered in developing a 120 psi pressure regulator assembly for the ECS which would meet the requirements imposed by the EVA/IVA system. Basically the high EVA/IVA flow rates created low temperatures which caused relief valve leakage and combined with cabin humidity to cause sense port blockage with ice. These problems are discussed in Section 2.5.5.

2.6.5.2 Suit Cooling System

The major problem encountered was the occurrence of corrosion in the loop which caused formation of deposits in the pump. This phenomenon resulted in the inability of the pumps to start after having been dormant for a period of time. The problem was ultimately solved by addition of a suitable corrosion inhibitor to the fluid and increasing the pump vane/rotor clearances.

The early design of the SUS loops utilized untreated MMS-606 water as the circulating medium. Vendor pump tests using this fluid disclosed starting problems, caused by corrosion on the pump internal parts. These problems forced a change of the pump vanes, rotor, and liner materials from the more wear resistant tungsten carbide to a more corrosion resistant Colmony alloy. The materials changes combined with the addition of additives to the water for corrosion and bacteria control resulted in satisfactory pump performance. These additives were 2% by weight of dipotassium hydrogen phosphate and 0.2% by weight of sodium borate for corrosion control and 500 PPM Roccal for bacteria control. This fluid, PB 3-302 (Rev. E), and pump design also was used in the ATM C&D/EREP coolant system.

After installation of the liquid gas separator in the SUS loops, NASA materials testing indicated that the Roccal additive was incompatible with the separator performance, causing water carry-over through the gas discharge port. A concern was also expressed about the presence of the Roccal reducing the strength properties of the tygon tubing in the LCG's. The Roccal was therefore replaced by 20 PPM movidyn, another biocide consisting of a colloidal silver solution. Subsequent SUS loop operation with this new fluid resulted again in problems with pump starting. Failure analysis determined that the pump locked up after dormancy due to deposits formed by interaction of the dipotassium hydrogen phosphate and silver in the movidyn with nickel from the fins of the SUS loop heat exchangers. These deposits formed between the pump vanes and rotor interfaces, preventing one or more of the vanes from moving freely in the rotor slots.

At that point in the program, the flight vehicle was undergoing final tests in preparation for shipment to KSC, so a crash effort was undertaken to determine a solution to the problem. The basic approach was to find a suitable replacement for the water solution. Simultaneously, additional design analyses and tests were conducted on alternate pumps and heat exchangers in the event of failure to find a suitable replacement fluid. An alternate pump module, utilizing a modified CSM coolant pump, powered by a transformer and compressor inverter, was designed and tested as a backup to the existing pump module. Also, a design feasibility study was initiated to modify the SUS loop heat exchangers to an all stainless steel configuration.

Neither of the above design changes was required. The final solution was arrived at by beaker-type materials testing and end-to-end systems testing on a variety of candidate fluid compositions. These tests established SUS loop compatibility with a fluid consisting of MMS 606 water containing additives of 20 PPM movidyn and 500 PPM sodium chromate. The SUS pumps were also modified by increasing the vane/rotor clearance to further minimize start up problems.

The flight vehicle SUS loops were drained, cleaned, and reserviced at KSC with the new fluid (PS 13240, Type VII). The modified, increased vane clearance, pumps were also installed. The final system configuration proved to be satisfactory as evidenced by the fact that no problems with SUS pumps were experienced at any time during the mission.

2.6.6 Conclusion and Recommendations

The EVA/IVA system performed well enough to include some lengthy and strenuous workshop repair tasks, resulting in expansion of original mission objectives. All mission objectives were accomplished and at no time was crew safety compromised. It is recommended that the Airlock EVA/IVA system - design concept, verification procedure, and operational hardware - be used on future missions with an EVA requirement.

EVA/IVA performance details are discussed below:

- Some design requirements were inconsistent with Skylab EVA experience and should be changed accordingly:
  - (1) Waste heat load range requirement of -800 to +2000 BTU/HR./MAN was too severe and should be changed to be compatible with LCG heat transfer capability at the operating temperature level. Maximum heat load for all three crewmen was approximately 2200 BTU/HR. and a negative heat load was not experienced.
  - (2) The maximum allowable water delivery temperature of 50°F was too severe. Temperatures of 58°F provided adequate cooling.
  - (3) Total duration of EVA exceeded seven hours, with cooling water flow exceeding eight hours - system requirements were three and four hours, respectively.
  - (4) The system was designed to support two EVA crewmen on one loop with the other crewman (STS) on second loop. During the mission, a single loop effectively supported all three crewmen.
- Modular design facilitated system checkout.
- Oxygen flow and suit cooling system support was provided as required for 12 EVA/IVA operations including, on DOY 359, a record EVA hatch open time exceeding seven hours.
- Loss of SUS #1 cooling fluid occurred due to leakage of LSU/PCU during an EVA. Reservice, as planned and provided for, was accomplished. Provisions to allow inflight reservicing of fluid systems should be included in all future missions.
- During SST a pump/fluid compatibility problem was discovered. It was caused by a late change in system additives and insufficient all-up material/fluid testing prior to U-1 system activation for test. To insure

no repetition on future programs, material/fluid testing must be conducted early and all-up system testing, with all materials and components, should be conducted as early as possible for all fluid systems, especially water systems.

- Differential pressure instrumentation was deactivated prior to launch due to a potential of shorting out the 5V bus and eliminating all instrumentation connected to that bus. Loss of ΔP information complicated the determination of loop performance and the isolation of flow problems. Differential pressure transducer design improvement should be made prior to next program usage.
- Airlock EVA vent formed ice when venting moist gas overboard. The problem was solved by providing a screen, on SL-3, that could be removed (with ice formation) late in venting operation. Future overboard vents should include means to prevent excessive ice build-up, i.e., vent heaters or removable screens.

2.7 ELECTRICAL POWER SYSTEM

2.7.1 Design Requirements

2.7.1.1 Introduction

The Airlock Electrical Power System (EPS) design evolved from a simple primary battery system to a complex solar array/secondary battery system. This evolution was prompted by changes in mission objectives and design requirements.

2.7.1.2 Design Evolution

Initially, all system power after docking was to be derived from the CSM; therefore, the AM was required to provide only a minimal amount of power during the initial mission phase. The AM EPS consisted of a number of silver-zinc primary batteries and a distribution system.

As the mission duration was extended and the sophistication of the OWS increased to accommodate a more ambitious experiment program, the AM EPS design concept was changed to a solar array/secondary battery system for orbital operations, with primary silver-zinc batteries used for preactivation power requirements. The first of these designs had solar arrays mounted on the Airlock in various configurations. As the requirements increased, the solar arrays were moved to the OWS where there was more room to accommodate the increased array size. Also, in the early design stages, the batteries and power conditioning equipment design were evolved through a series of tradeoff studies. Although both silver-cadmium and nickel-cadmium batteries were considered, the nickel-cadmium type was selected based on the availability of considerably more test and flight data, implying less development risk. A number of solar array/secondary battery system designs were evaluated, with the primary goal of increasing the overall efficiency and reliability of the system. This necessitated departures from some normally accepted conservative practices and considerable effort in state-of-the-art advancement. To increase EPS efficiency, buck regulation was selected for both the battery charger and voltage regulator, and a peak power tracker was incorporated in the charger to extract maximum array power when demanded by the system. The modular regulator design was selected for both the battery charger and voltage regulator for maximum reliability and high efficiency, in addition to redundant control circuitry. When the initial design approach was firmed up, the AM EPS consisted of four power conditioning groups (PCG's), each consisting of a battery charger, a voltage regulator and a battery. Input power for the PCG's was derived from solar arrays mounted on the OWS, the solar array being an adaptation of an existing Agena design.

At this time, the ATM was a free flying vehicle which was to dock with the Skylab during the final manned mission. In the earlier missions, it was planned to fly the cluster in a gravity gradient attitude with the vehicle X-axis along the local vertical. After the ATM had docked, the attitude was to be solar inertial. In order to provide more power to the buses in the gravity gradient attitude, it was planned to have an articulated solar array for improved solar pointing.

Power requirements continued to increase in the early design stages, resulting in greater solar array area and expansion of the number of AM PCG's first to six and finally to eight. Reduction of preactivation load requirements coupled with the increased nickel-cadmium battery energy for eight units, led to the elimination of AM primary silver-zinc batteries.

2.7.1.3 Design Changes

It was initially planned to use the ATM solar modules for both the ATM and OWS solar arrays to achieve standardization. However, since the input voltage requirement for the two power systems was different, it would have been necessary to wire the ATM solar modules such that one-half of the series string of one module was wired in series with a second module. Thermal analyses of the solar array predicted that the maximum array output voltage would be higher than the 110 volts used for AM PCG design. Design requirements for the AM charger and voltage regulator were changed at this time to accept input voltages of 125 volts maximum which provided some margin above the maximum predicted voltage. Shortly after this, the so-called "dry launch" design was adopted which made the ATM an integral part of the cluster and made the OWS S-IVB a true space laboratory rather than a propulsive stage. Since the ATM attitude system could hold the cluster in the solar inertial attitude, there was no longer any need to orient the OWS solar array and the articulation mechanisms were removed from the workshop design.

An optimized solar array was later conceived for the OWS which was designed specifically to be used with the AM PCG's as an integrated power system. Maximum and minimum voltage and power requirements were specified the same as the 1-1/2 ATM module design, and, therefore, did not necessitate any PCG redesign.

In the process of design evolution, a second amp-hour meter (AHM) was added to the battery charger to improve its reliability. Also, a discharge limit feature was added to provide a signal to the voltage regulator when the AHM computed battery SOC equaled 30%. The voltage regulator reduced its output voltage by 2 volts in response to this signal and effectively removed the associated battery from the bus. This feature was added to prevent inadvertant overloading of any one battery, although intentional deep discharges were still possible by use of over-ride logic circuitry. An on-board display of AHM status was added in addition to ground telemetry. A feature was also added to permit manual override of the 100% state-of-charge (SOC) signal from the AHM and to continue battery charging at the voltage limit.

Battery cell evaluations during initial testing prompted internal cell changes to reduce the probability of cell internal shorts. To further improve cyclic life, battery operating temperature was reduced. This lowering of operating temperature was accomplished by changing the battery case material, lowering the coolant loop battery module inlet temperature, and reducing the AHM return factor and battery trickle charge rate. The latter necessitated battery charger design changes.

2.7.2 System Description

2.7.2.1 Introduction

The AM EPS was one of three electrical power systems which provided power for the entire Orbital Assembly. The performance requirements of the AM EPS included compatibility with those of the other two power systems and of the consuming elements. This description is basically limited in scope to the AM EPS, but also takes cognizance of the characteristics of other module systems where they are pertinent to the design of the AM EPS. Performance values and tolerances used in this description are intended for illustrative purposes only. The individual equipment specifications, listed in Appendix H, should be consulted for any in-depth evaluation of individual hardware units.

The AM EPS was designed to accept power from a solar array system mounted on the OWS and to condition this power for application to the AM EPS buses and to the AM EPS batteries. The OWS solar array system was divided into eight electrically identical parts called solar array groups (SAG's). Each group was composed of thirty solar modules (Figure 2.7-1) which provided input power to either one of two selectable individual PCG's. Each PCG was composed of a battery, a battery charger, a voltage regulator, and the associated power distribution and control circuitry. The function of each PCG was to provide conditioned power to using equipment, and to recharge the nickel-cadmium batteries during the orbital daylight periods. Various control functions were designed into the AM EPS to effectively manage each PCG and to apply the PCG outputs to the various AM EPS buses. Appropriate control switching was provided on the STS instrument panels or by ground control via the AM Digital Command System.

The AM EPS also included the wiring and controls for power transfer between all of the various power systems and for power distribution to the electrical power loads in all of the OA modules. The AM EPS was designed to operate in parallel with the ATM EPS or CSM EPS to supply power to the AM, ATM, OWS, MDA, and CSM. The distribution system was controlled by switches on the STS instrument panels or via the AM DCS. Appropriate monitoring displays for the PCG's and the distribution system were provided on the STS instrument panels and appropriate EPS parameters were instrumented for ground monitoring by the AM telemetry system.

The major equipments comprising the AM EPS consisted of eight power conditioning groups, several control panel assemblies, a dual bus distribution system, a number of relay panels, and two shunt regulators. The battery chargers, batteries, voltage regulators and relay panels for four PCG's were mounted on each of two battery modules. The location of the battery modules is shown in Figure 2.7-2 and the equipment mounted on a battery module is shown on Figure 2.7-3. The shunt regulators were mounted on the -Y axis under truss No. 1 as shown in Figure 2.7-2. The control panel assemblies were located in the Structural Transition Section and included the on-board controls, displays, and circuit breakers. The electrical power and distribution system included 188 relays, 90 switches, 110 circuit breakers, 166 status lights and miscellaneous other related equipment.

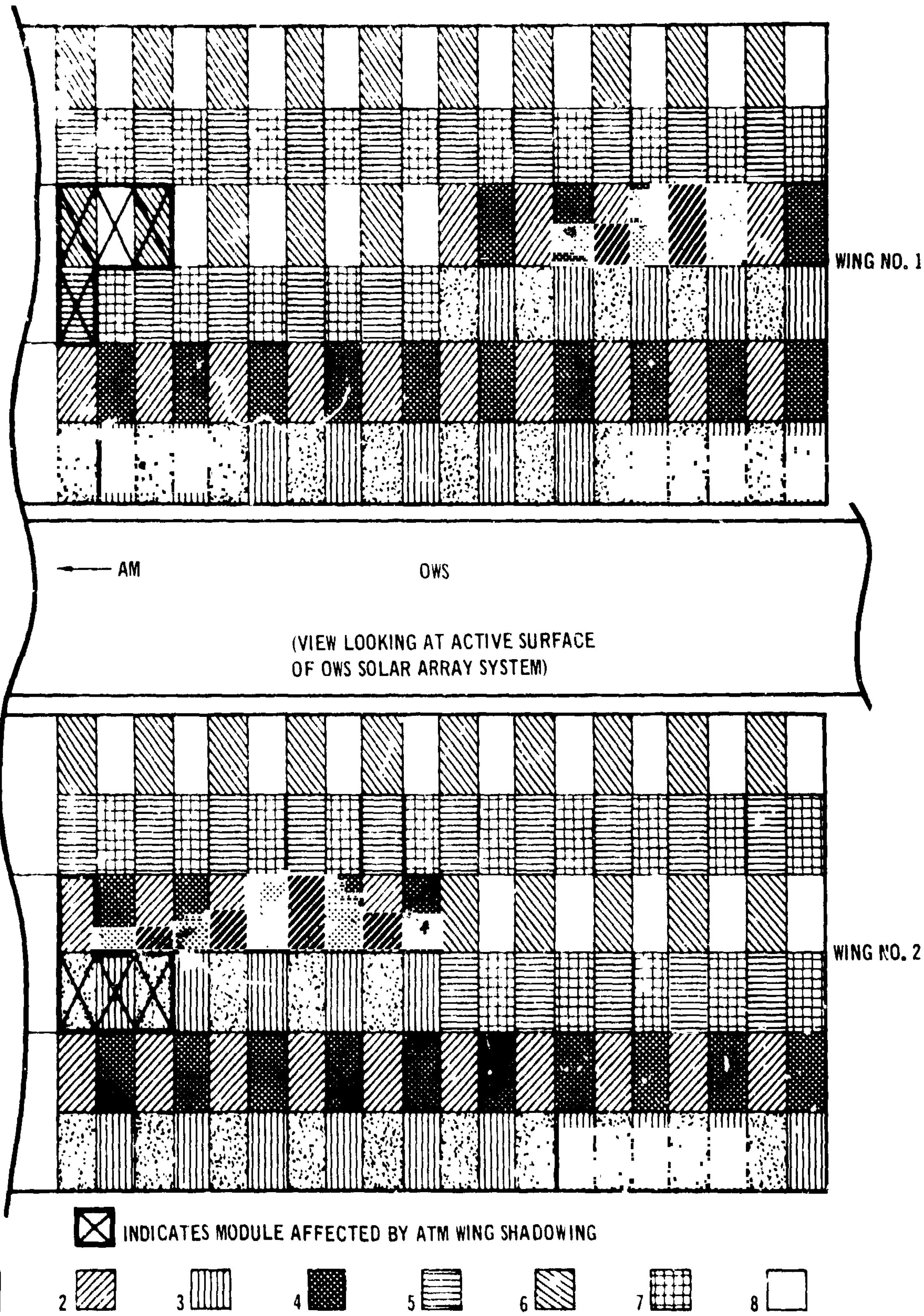


FIGURE 2.7-1 MODULE LAYOUT – SOLAR ARRAY GROUP

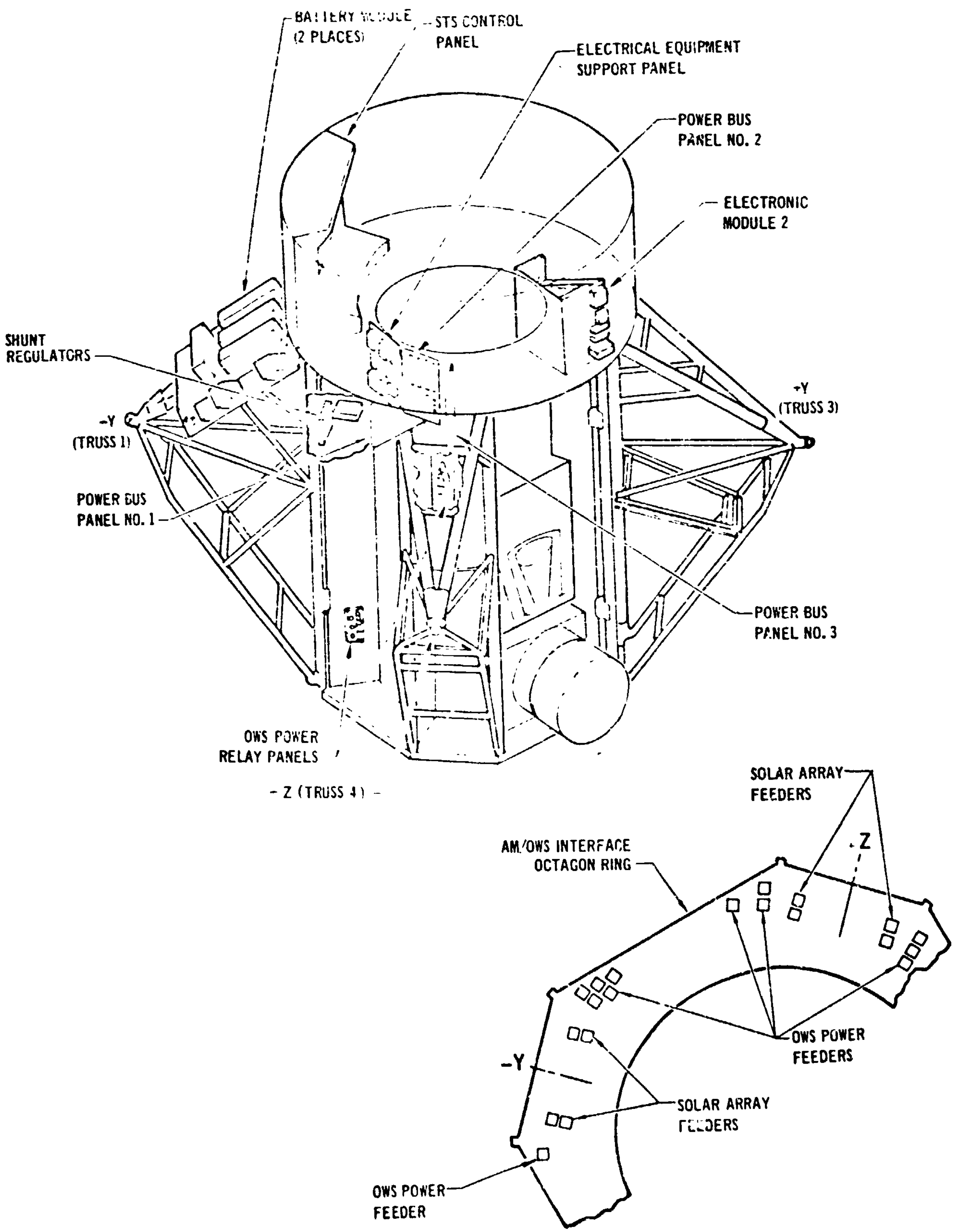


FIGURE 2.7-2 AIRLOCK MODULE EQUIPMENT LOCATION

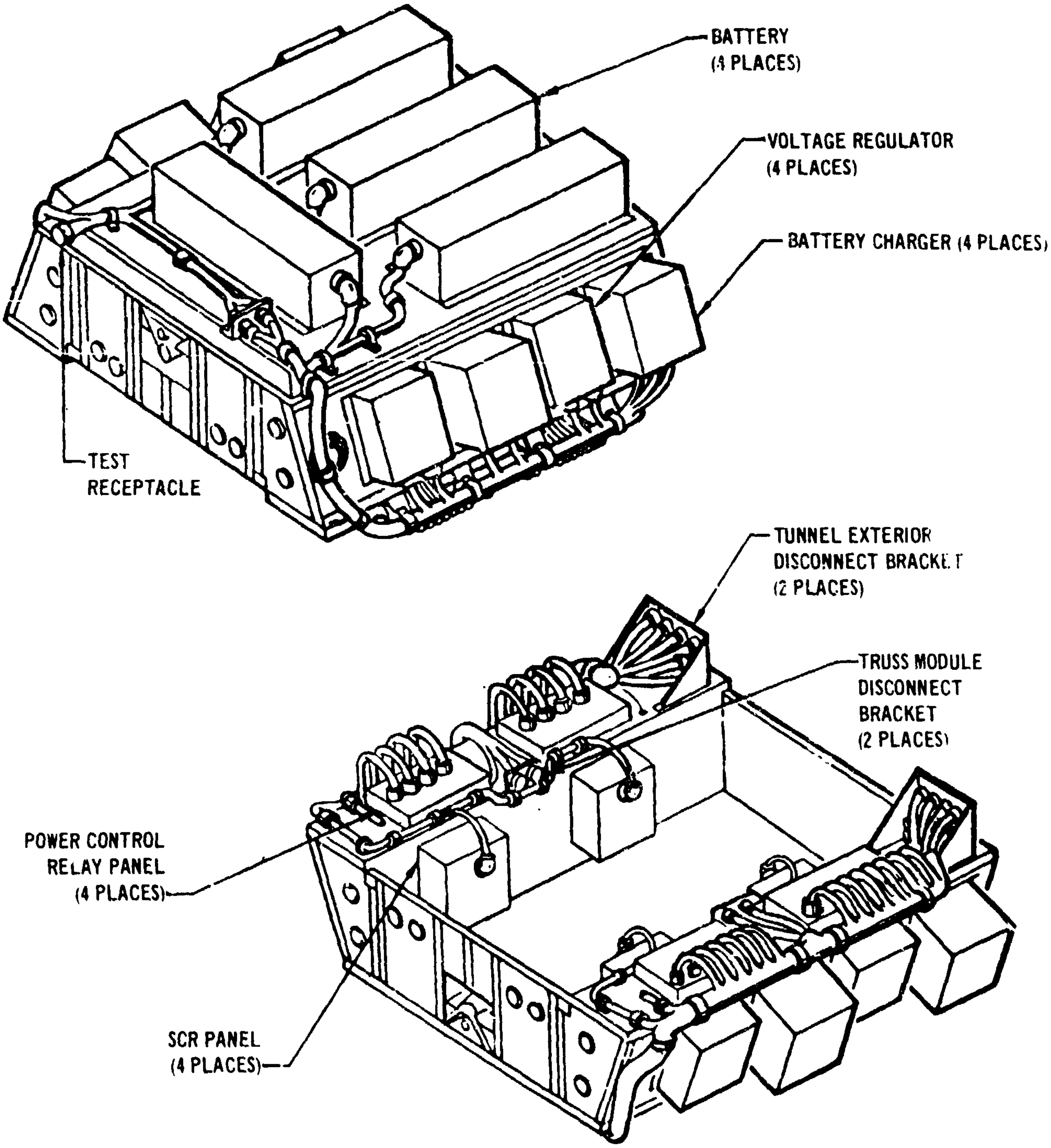


FIGURE 2.7–3 PCG COMPONENT LOCATION – BATTERY MODULE

2.7.2.2 Power Conditioning Groups

A total of eight PCG's were required in the AM EPS to efficiently utilize the total array energy received from the OWS solar array system. The multiple number of PCG's also provided redundancy to meet mission reliability requirements. A typical PCG circuit configuration, including controls and instrumentation is shown on Figure 2.7-4. The control functions will be discussed as they relate to the operation of the major PCG components; the battery, battery charger, and voltage regulator. The PCG equipments interfacing with the OWS solar array were designed to operate compatibly with the solar array group design characteristics.

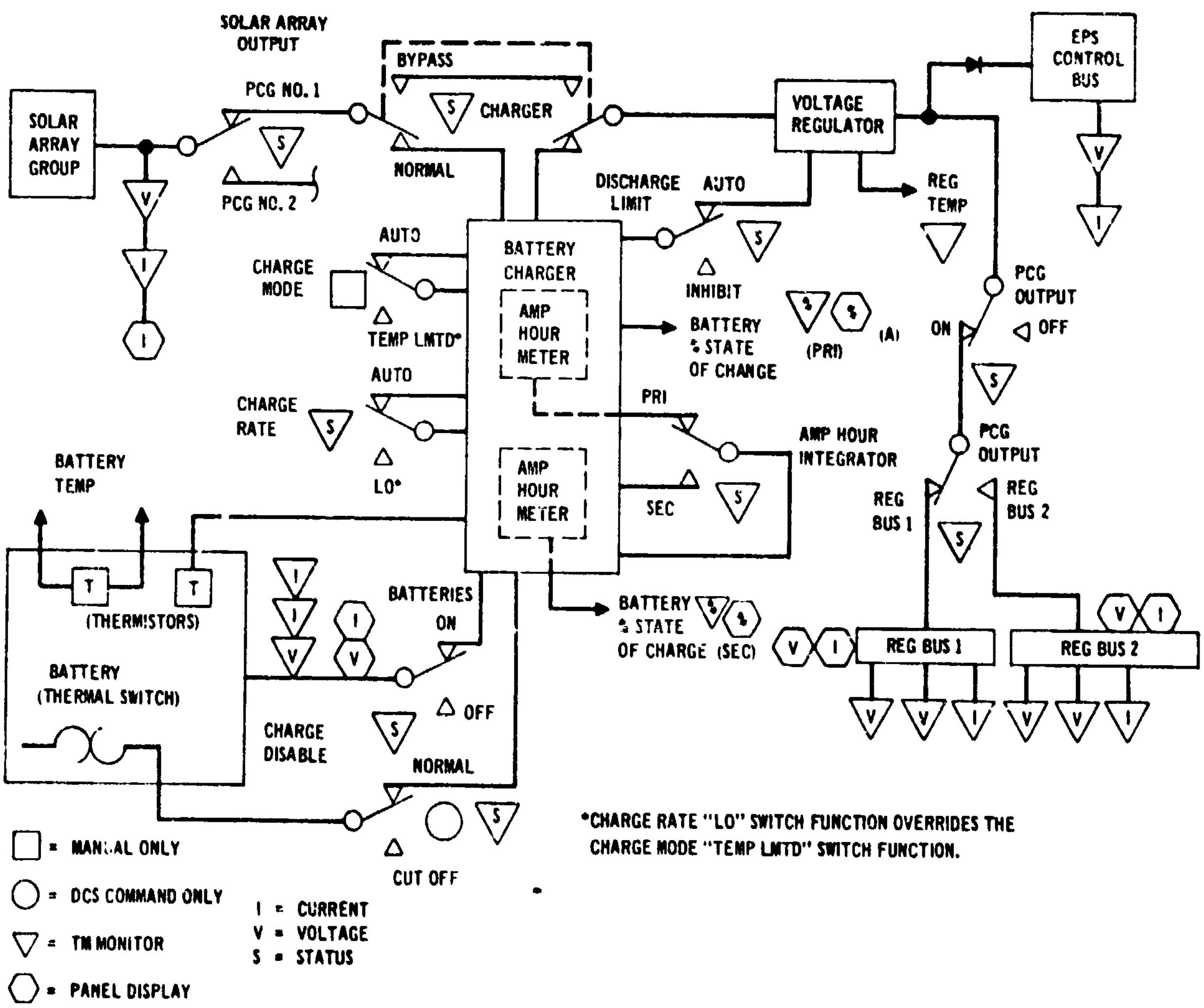


FIGURE 2.7-4 TYPICAL PCG CIRCUIT - CONTROLS AND INSTRUMENTATION

A. Battery Charger - The complete detailed characteristics of an individual battery charger were specified in McDonnell Procurement Specification 61B769006. Physically, each charger weighed 27 lbs, with dimensions of 7.25" x 10" x 11.55", and required coldplate mounting. Each battery charger conditioned the power obtained from an associated OWS solar array group, controlled the charging of its associated nickel-cadmium battery, and fed solar array conditioned power or battery power to its associated voltage regulator to satisfy system load requirements. The battery charger was designed to provide a maximum instantaneous output power of 2300 watts and a maximum continuous output power of 1500 watts. Maximum output voltage was 52 VDC.

The acceptable AM/OWS interface voltage range for battery charger operation was from 125 volts maximum at open circuit to 51 volts minimum at the peak power point of the solar array group V-I characteristics. Maximum input power was 2580 watts.

The battery charger consisted functionally of three major circuits; the switching regulator circuit, the peak power tracker circuit, and the ampere-hour meter circuit. These circuits are shown on the battery charger block diagram, Figure 2.7-5. The switching regulator was the actual power conversion circuit which conditioned the solar array power and provided the regulated output. The peak power tracker restricted the load demand on the solar array group to the peak power available from the group. The ampere hour meter controlled the charging modes for the battery.

(1) Peak Power Tracker - The function of the peak power tracker circuit was to automatically adjust the battery charger output voltage such that the power demand on the associated solar array group was limited to its available peak power. Without the peak power tracker, a load demand in excess of the available peak power would cause a sharp drop in the solar array output voltage, and, therefore, a sharp drop in its output power.

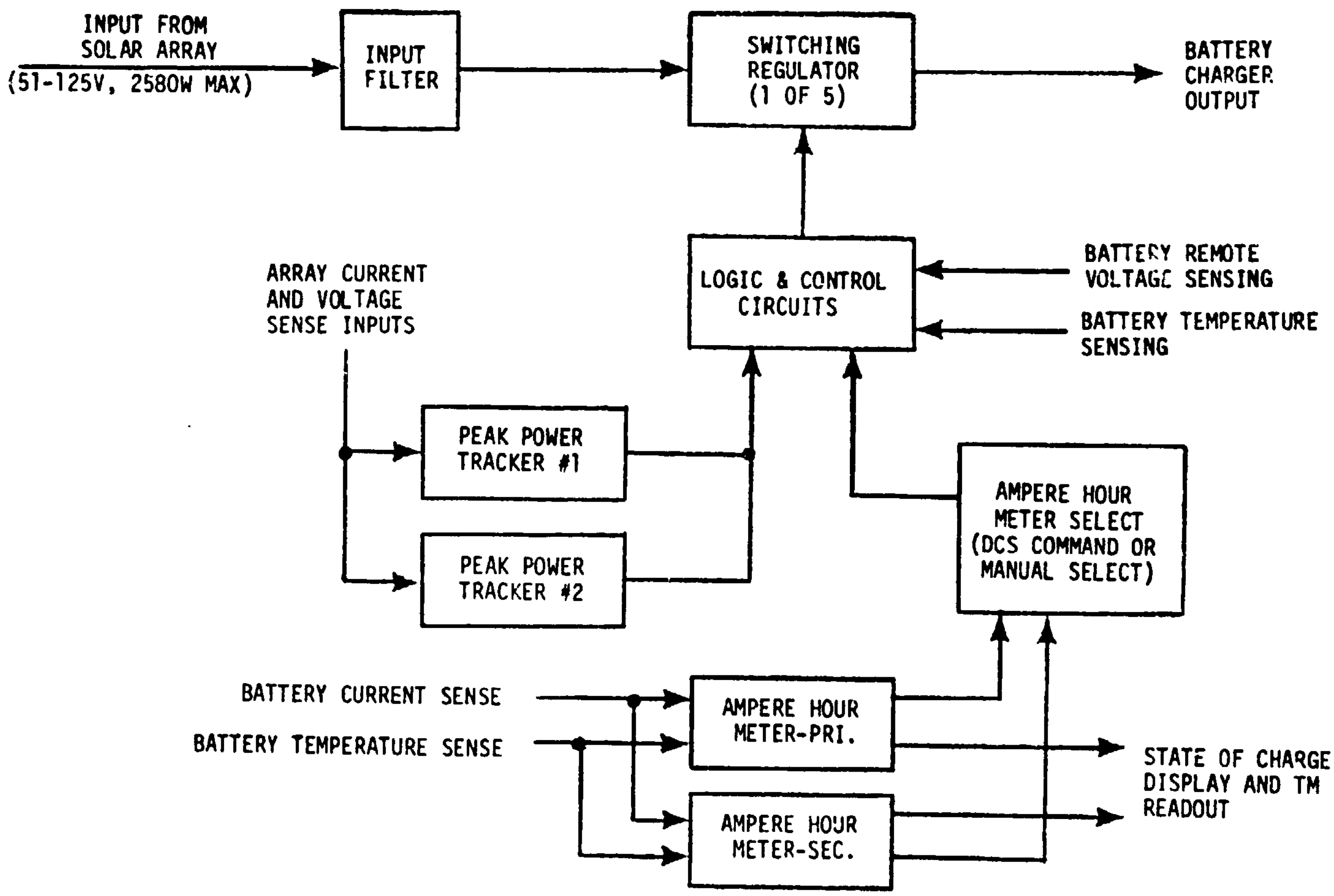


FIGURE 2.7-5 BATTERY CHARGER FUNCTIONAL BLOCK DIAGRAM

The circuit sensed the array output parameters of voltage and current, determined the relationship of the operating power point to the peak power point, and generated an appropriate signal to the battery charger regulator circuit to control the charger output voltage and output power. The peak power tracker circuit was designed to limit the load on the solar array to its available power. Under limiting conditions it caused operation at or within 5% of the solar array peak power point. Redundant active peak power tracker circuits were provided in each battery charger, as shown in Figure 2.7-5, for improved system reliability.

In addition, the peak power tracker circuit was designed such that any failure within the circuits affected only its peak power tracking function and did not affect any other function of the battery charger.

(2) Ampere Hour Meter - The function of the ampere hour meter circuit was to continuously compute the state-of-charge (SOC) of the associated battery and to provide charge control signals based on the computed SOC. This was accomplished by monitoring the battery discharge in ampere-hours during dark periods and the battery recharge in ampere-hours (including the return factor) during daylight periods. The battery status at any time was then computed in % SOC based on starting at 100% with a fully charged battery. The 100% SOC was based on a battery capacity of 33 ampere hours. The primary control signal, generated when the computed SOC reached 100%, terminated the voltage limited charge mode and initiated the current limited charge mode. An analog signal indicating the computed SOC was also generated in the ampere-hour meter for telemetry and display usage. Two identical ampere hour meter circuits were provided in each battery charger, as shown on Figure 2.7-5, for improved system reliability. Both of these circuits computed the battery SOC at all times and provided a continuous analog signal indicating computed battery SOC for telemetry and display. However, only one of these circuits provided battery charge control signals at any one time. Selection of either the primary or secondary circuit for control purposes was made by a DCS command or by a crew manual switch.

Temperature compensation was provided during charge cycles to account for the interrelationship between charging efficiency and battery temperature. Three thermistors in the associated battery provided temperature sense signals to the compensating network of the ampere-hour meter. The battery was considered fully recharged when the ampere-hours delivered to the battery were equal to the ampere-hours removed, multiplied by the "return factor" shown in Figure 2.7-6. At that point, the ampere-hour meter output indicated a battery SOC of 100%, and a signal was provided to the battery charger regulator circuit to cause operation in the constant current battery charging mode rather than the voltage limited battery charging mode used at computed SOC values less than 100%. When the computed battery SOC value dropped to 30%, a signal was provided to the associated AM EPS

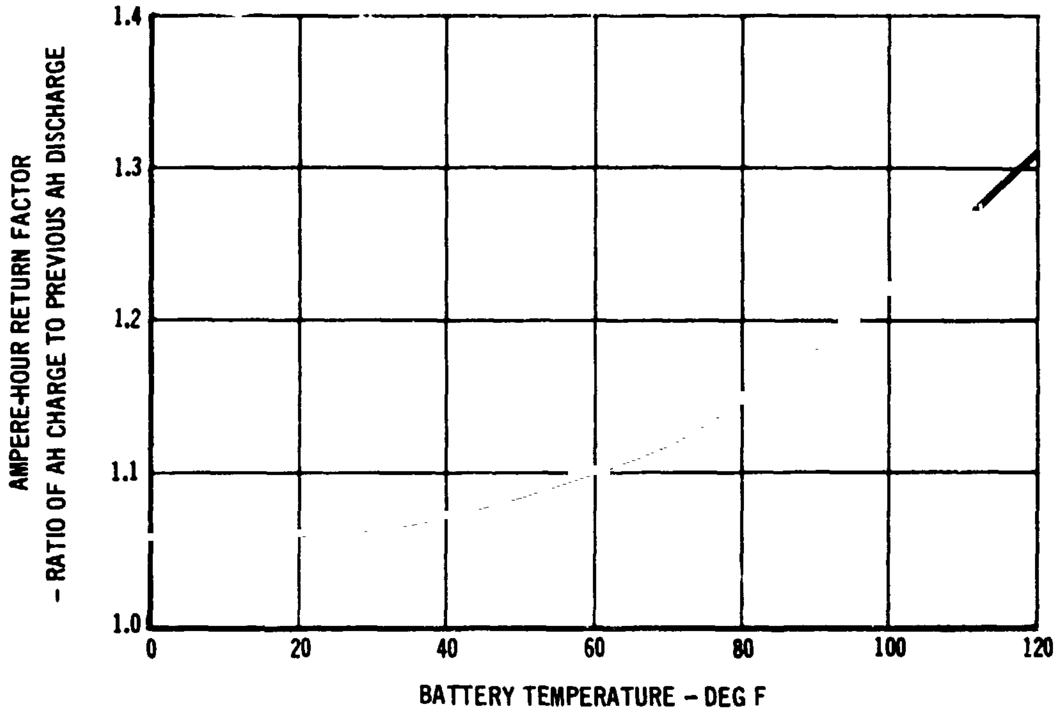


FIGURE 2.7-6 AMPERE-HOUR RETURN FACTOR VERSUS BATTERY TEMPERATURE

voltage regulator which caused the voltage regulator to reduce its output voltage by approximately two volts. This effectively removed all load from the PCG and permitted all available power from the associated solar array group to be utilized for the recharging of the battery. When the battery had required a 50% state-of-charge, the two-volt reduction mode was discontinued, and normal operation resumed. The initiating control signal could be inhibited by a DCS command or by astronaut control. Each ampere-hour meter circuit provided a signal to a battery SOC meter provided on the STS instrument panel and to the Instrumentation System for continuous TM display.

(3) Regulator - The battery charger regulator was a pulse width modulated type voltage regulator where the regulated DC output voltage was less than the unregulated DC input voltage. The regulator consisted of an input filter and five individual power modules. A multiple number of regulator modules were used for both increased system reliability and minimum parasitic losses at low load conditions, resulting in high overall efficiency of operation.

A battery charging cycle typically included three modes of battery charger operation; a peak power tracking mode as explained above, a voltage limited mode, and a constant current mode. A charging cycle would start with the battery SOC at some value less than 100%, and operate in the peak power tracking mode until the battery terminal voltage increased to the temperature dependent voltage limit; then operate in the voltage limited mode and provide an output voltage determined by the battery temperature curve of Figure 2.7-7.

When the battery SOC reached 100%, the battery charger switched from the voltage limited mode to the constant current mode, maintaining the battery charging current at 0.75 $\pm$0.5 amp.

Battery charging in either mode was terminated and battery current was reduced to zero $^{+0.5}_{-2.0}$ amps if the battery temperature exceeded a high temperature limit of approximately 120°F as measured by thermistors in the battery. A thermal switch in the battery acted as a backup to the thermistors and provided the same results at a maximum temperature of 125°F.

Several manual and DCS controls could be used to modify the charging cycle. The Charge Mode control (manual only), when set to its Temperature Limited position, inhibited the 100% SOC signal. This prevented the automatic changeover from the voltage limited mode to the constant current mode when 100% SOC was reached. Battery charging would then proceed in the voltage limited mode as long as the battery temperature limit was not reached. The Charge Rate control (manual or DCS) when set to the Lo position restricted

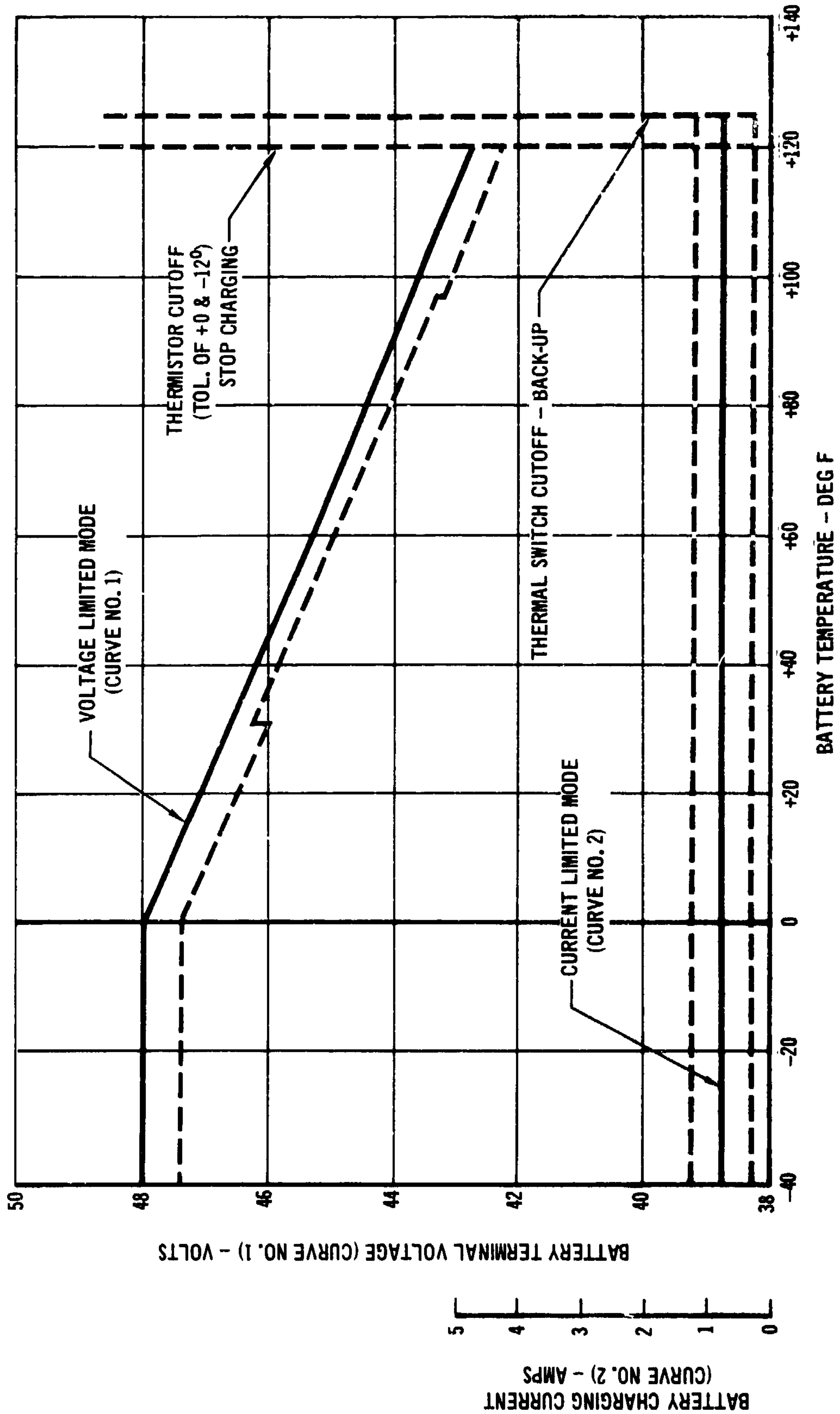


FIGURE 2.7-7 BATTERY CHARGING MODE CURVES

charging to the constant current mode. This included overriding of the Charge Mode control. The Charge Disable Control (DCS only), when set to the cut off position caused the battery charger to maintain the battery charging current at zero $^{+0.5}_{-2.0}$ amperes. The positive power connection from the battery to the battery charger could be opened by positioning the "Batteries" switch (manual or DCS) to the "Off" position. This condition was sensed by the battery charger as a complete loss of the battery voltage signal. The battery charger voltage, then applied only to the voltage regulator, was controlled at 52 ± 1 volt for this condition.

There were four different operational conditions arising from varying levels of available solar array power. The condition where solar array power was sufficient to supply both the equipment load and the battery load has previously been described. Under the condition where the available power was sufficient to supply equipment loads, but not sufficient to supply the total of equipment and battery loads, the charger output voltage was reduced such that the equipment load was satisfied and the remaining available power was utilized for charging the battery. When the available array power was not sufficient to supply the equipment load alone, the charger output voltage was reduced further until the battery and battery charger in parallel could supply the equipment load. When the solar array voltage became less than approximately 51 volts at the AM/OWS interface, the battery charger was switched off and equipment loads were totally supplied from the battery. The latter condition included the normal operation during orbital dark periods.

B. Voltage Regulator - The detailed performance characteristics of an individual voltage regulator were specified in McDonnell Procurement Specification 61B769005. Physically, each regulator weighed 14 lbs with dimensions of 4.3" x 10" x 10.85" and required coldplate mounting. Eight voltage regulators were included in the AM EPS, one in each PCG. The function of the voltage regulator was to furnish regulated DC power, within specified voltage limits, to the AM REG buses and the EPS Control buses.

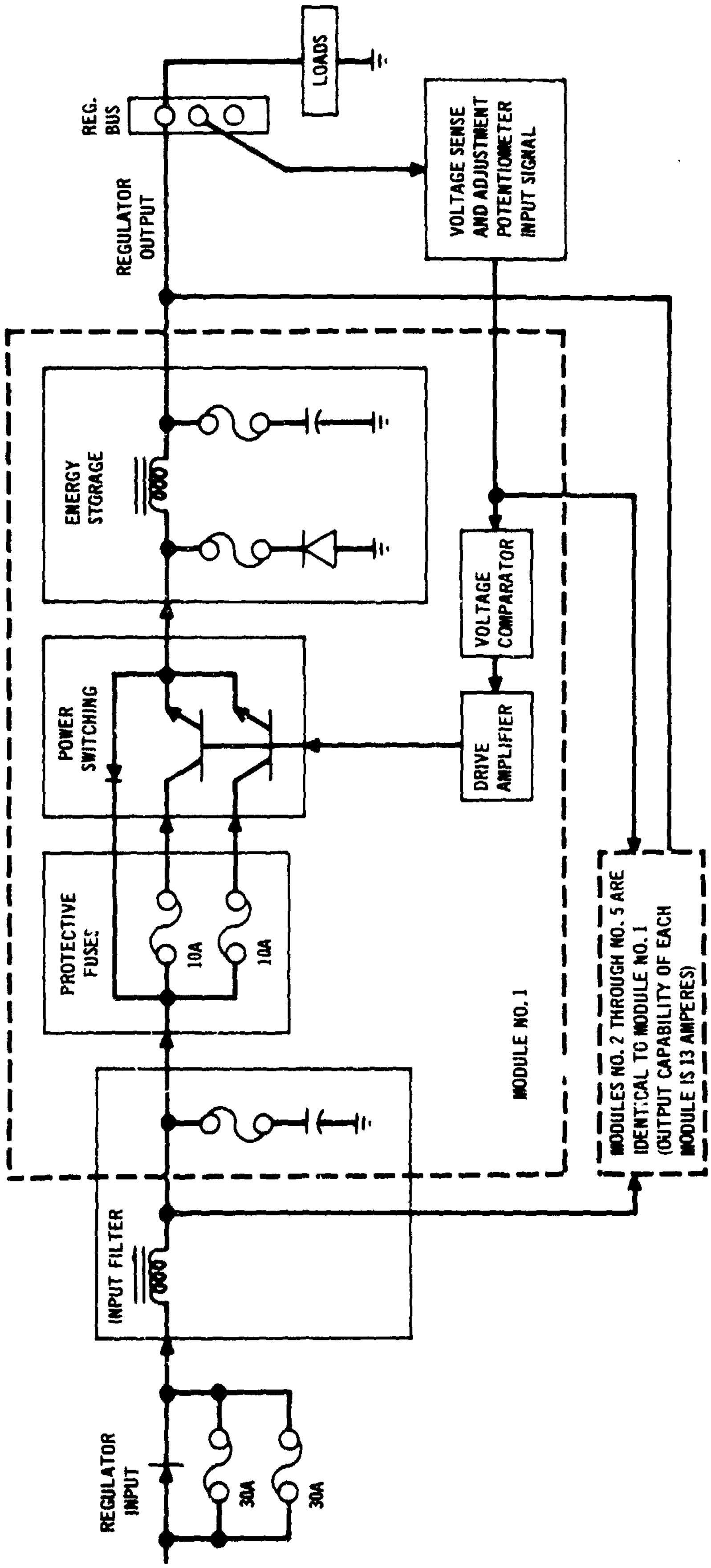
Each voltage regulator received input power from one of four sources, the nickel-cadmium battery, the battery charger, the battery and battery charger operating in parallel, or the associated solar array group. The input voltage level varied according to the output characteristics of these sources. The battery supplied power within an approximate voltage range of 30 to 40 volts, depending on battery SOC and battery temperature. For the parallel battery and battery charger operation, the voltage varied from approximately 35 volts to 46 volts, depending on the amount of sharing and on battery SOC. The above conditions are discussed in detail in the descriptions of the battery charger and battery. In a contingency mode of operation power could be supplied directly from the solar array group output to the voltage regulator input by positioning the Charger switch to its Bypass position. For this case, the input voltage to the regulator would be approximately 51 volts minimum to 125 volts maximum.

The voltage regulator provided specified voltage levels at the AM REG bus for input voltages from 32 to 125 volts. For input voltages less than 32 volts, the regulator provided the specified bus voltage level or the input voltage level minus approximately two volts, whichever was lower.

Each voltage regulator basically consisted of five power modules and an input filter, as shown on Figure 2.7-8. The multiple number of power modules was included in the design for improved system reliability. Each power module was a pulse width modulated type regulator where the output voltage was less than the input voltage at all times. The power module was designed to provide high efficiency operation even at low load conditions.

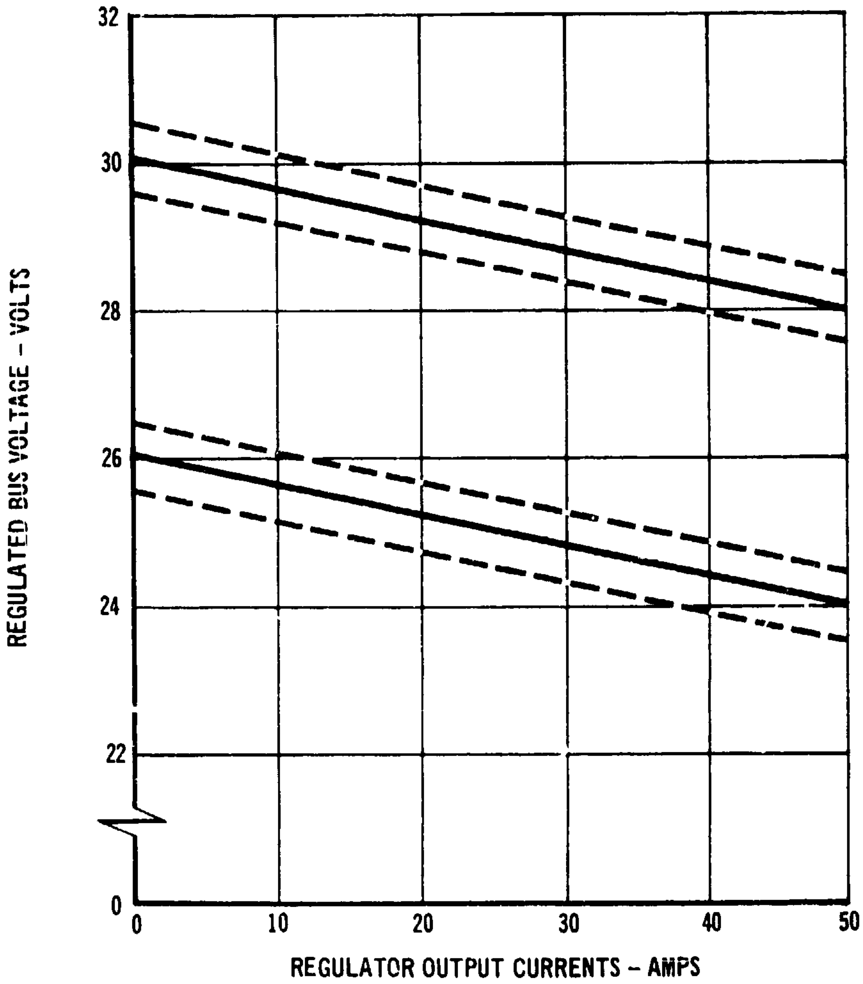
Figure 2.7-9 shows the output characteristic or V-I curve for an individual voltage regulator using nominal values for the defining parameters. For the AM EPS voltage regulators, the slope factor has a value of 0.04 $\pm$ 0.002 volt per ampere.

The no-load voltage for each regulator was crew adjustable only by means of two EPS manual control potentiometers: a Reg Bus, and a fine adjust potentiometer. Each of these potentiometers was connected directly



NOTE: MODULE OUTPUT VOLTAGES ARE SET AT DIFFERENT LEVELS SO THAT MODULES ARE AUTOMATICALLY ADDED TO SATISFY INCREASING LOAD REQUIREMENTS.

FIGURE 2.7-8 VOLTAGE REGULATOR BLOCK DIAGRAM



NOTES:

1. NO-LOAD BUS ADJUSTMENT RANGE IS 26 TO 30 VOLTS BY USE OF REG. BUS ADJUSTMENT POTENTIOMETER.
2. DOTTED LINES SHOW INDIVIDUAL REGULATOR FINE ADJUSTMENT RANGE OF ± 0.45 VOLTS ABOUT THE REG. BUS ADJUSTMENT BY USE OF FINE ADJ. POT.
3. INDIVIDUAL REGULATOR SLOPE FACTOR IS –0.04 VOLTS/AMP. (TOLERANCE OF ± 0.002 VOLTS/AMP IS NOT SHOWN).
4. ACCURACY (DRIFT FROM OPERATING POINT) OF ± 0.05 VOLTS IS ALSO NOT SHOWN.

FIGURE 2.7–9 VOLTAGE REGULATOR VOLTAGE AND CURRENT CHARACTERISTICS across the Reg bus and furnished a signal to the voltage regulator. There were two Reg bus potentiometers, one for each of the two Reg buses in the AM EPS. Each Reg bus potentiometer was hard wired to its Reg bus, but its control signal was switched to each of the voltage regulators supplying power to that Reg bus. It, therefore, simultaneously adjusted the outputs of a group of regulators in order to adjust the Reg bus voltage level. The no-load adjustment voltage range provided by the Reg bus potentiometer was from 26 to 30 volts.

There were eight Fine Adjust potentiometers in the AM EPS, one for each of the eight voltage regulators. The adjustment range associated with a Fine Adjust pot. was ± 0.45 volts with respect to the voltage level set by the appropriate Reg bus pot. The purpose of the Fine Adjust potentiometers was to provide an individual regulator adjustment to allow control of load sharing among regulators connected to a common Reg Bus.

In addition to the above parameters which affect the output V-I curve, the regulator output had an allowable drift of ± 0.05 volt under conditions of constant loading within the allowable output current range.

The output current range for voltage regulator specification performance was from 0 to 50 amperes. The voltage regulator automatically limited its output current to a maximum of 65 ± 3 amperes, regardless of loading conditions. A typical curve for the voltage regulator characteristic from no-load to short circuit is shown on Figure 2.7-10. Figure 2.7-10 also shows the allowable V_{OC}, I max, and I_{SC} tolerance bands.
For current loads in excess of 50 amperes the regulator was not required to maintain specified voltage performance. The regulator was, however, capable of operating continuously under any load condition without sustaining damage and was capable of providing specified performance upon removal of any excess current loading condition.

In a special mode of operation, the output of the voltage regulator was reduced by 2 volts upon receipt of a signal from its associated battery charger. This signal was the 30% battery SOC signal previously described in the battery charger description. The effect of the 2 volt reduction

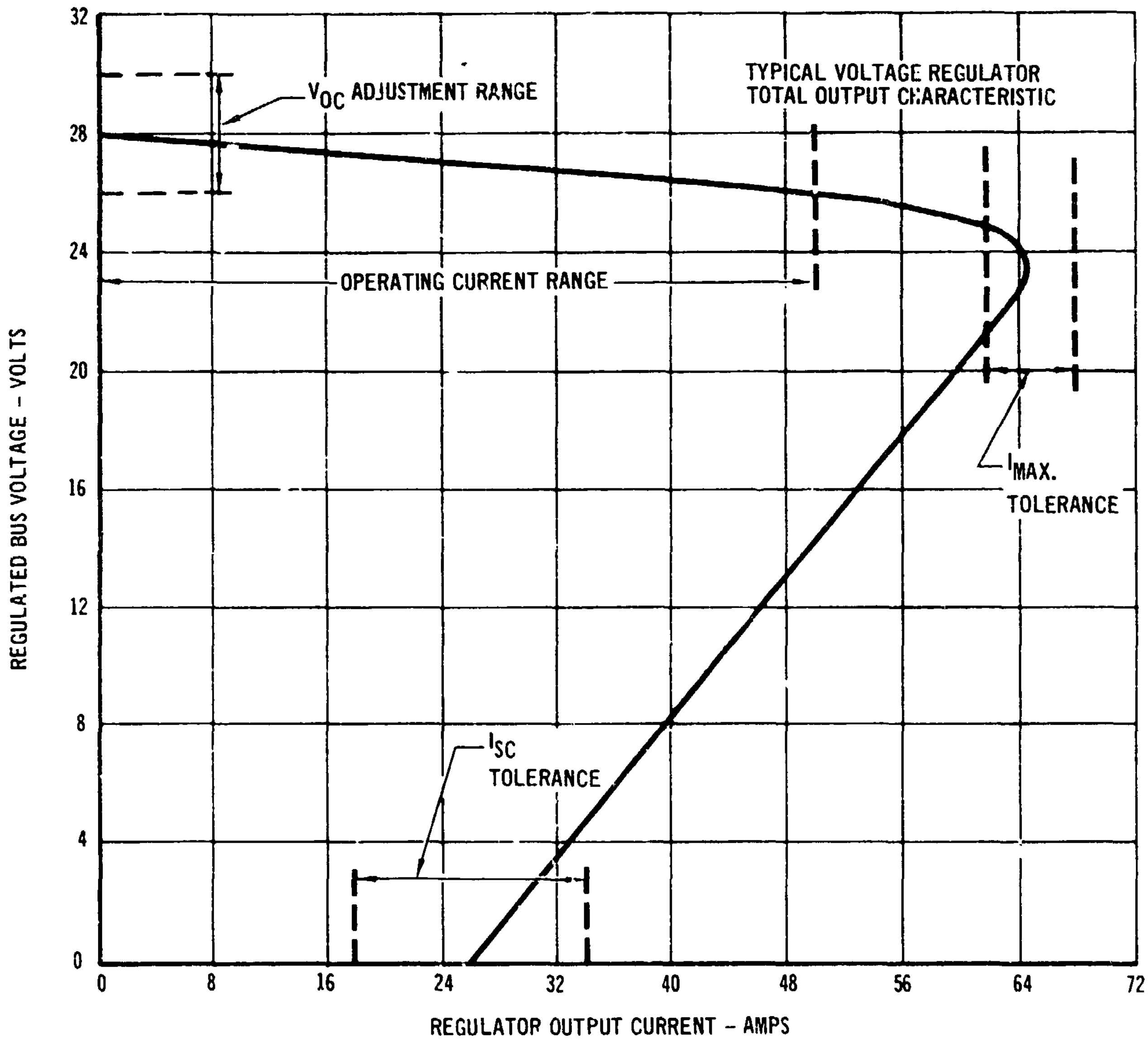


FIGURE 2.7-10 TYPICAL VOLTAGE REGULATOR TOTAL OUTPUT CHARACTERISTIC

was to totally unload the regulator as previously discussed. Upon removal of the signal from the battery charger, the voltage regulator output rose 2 volts to its original voltage level of operation.

C. Battery - The AM EPS batteries were nickel-cadmium batteries designed for active cooling. Detail requirements and characteristics were specified in McDonnell Procurement Specification 61B769004.

(1) Function - The function of the batteries was to furnish power to equipment loads through the AM EPS voltage regulators during orbital dark periods when there was no power available from the solar array system and to furnish supplemental power during periods when the power available from the solar array was insufficient to satisfy the total equipment load requirement. Battery capacity was 33 amp-hrs based on 120°F temp and 18 amp discharge rate to 30V. Average flight discharge voltage was 38V. Specified cycle life was 4000 cycles at approximately 25% depth of discharge.

(2) Recharge - The batteries were recharged whenever array power greater than the bus load requirement was available. The charge potential applied to the battery during the initial phase of recharging was limited to a level consistent with maintaining peak solar array power utilization. The recharge potential necessary to maintain peak solar array power utilization increased as the batteries approached completion of recharge. This phase of recharge was terminated when a potential limit consistent with battery temperature was imposed by the battery charger. Full utilization of the array power was no longer accomplished during the constant potential charge mode which continued until such time as the ampere-hour meter within the charger indicated sufficient recharge had been accomplished. Upon generation of such an indication the charger switched to a low level (0.75 $\pm$0.5 ampere), constant current charge mode for the remainder of the charging period.

(3) Construction - Each of the eight AM EPS batteries consisted of 30 series connected cells and associated temperature sensing devices packaged in a 7" x 8.25" x 27.25" aluminum container, and weighed 123 lbs.

Each cell consisted of a parallel connected group of positive and negative plates packaged in a stainless steel can and sealed with a cell header assembly. All plates were fabricated using a nickel wire - sintered nickel structure into which either active nickel or cadmium material was impregnated to produce a positive or a negative plate, respectively. Seventeen nickel (+) plates and eighteen cadmium (-) plates, alternately arranged and separated by nonwoven nylon made up a cell pack. The header assembly was welded to the cell can to complete the cell assembly. Each cell was fitted with a self-reseating pressure relief valve. Cell leakage criteria was the same as that imposed on hermetically sealed assemblies.

Each of the 30 cells was taped and then epoxy potted into one of the 30 individual compartments in the battery containers. Each battery also contained three nichrome wire temperature sensors; two temperature sensor assemblies containing three thermistor elements, and a normally closed thermal switch. Figure 2.7-11 details the function of each of these units. The temperature sensing devices were placed

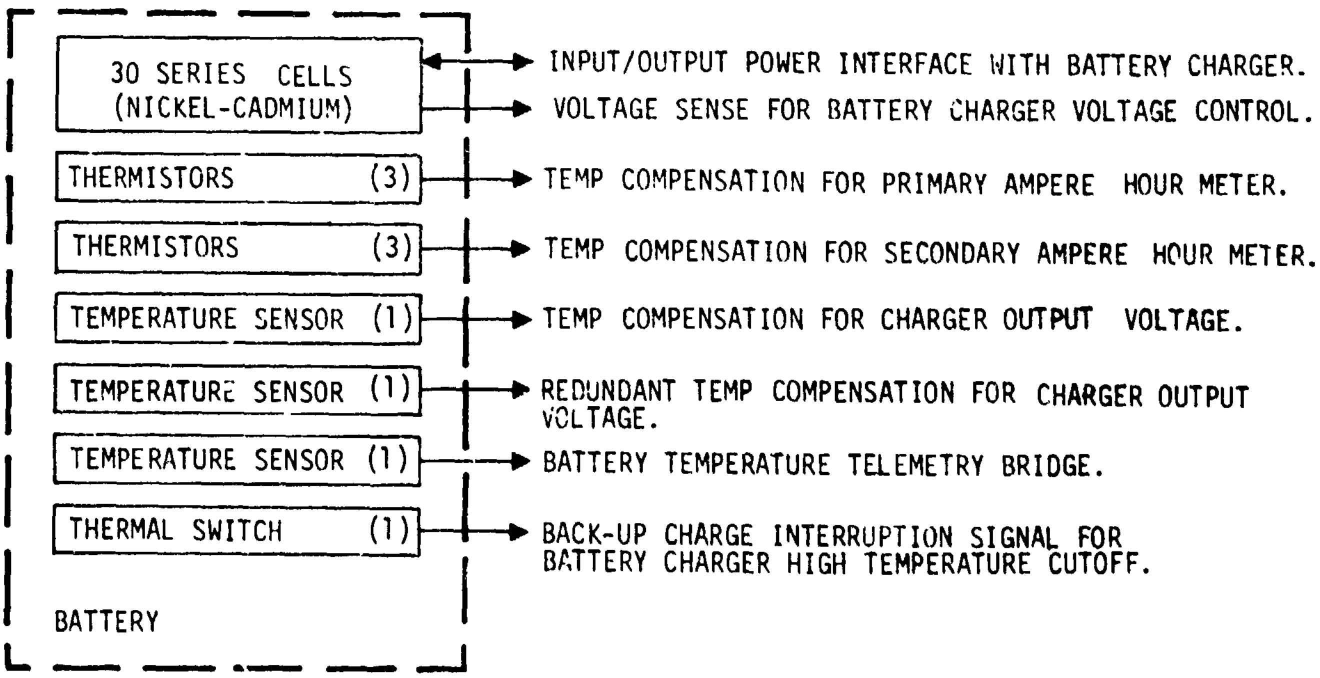


FIGURE 2.7-11 BATTERY OUTPUT FUNCTION DIAGRAM

such that the top of cell case temperature was monitored rather than terminal or cell interconnect temperatures to minimize or preclude terminal and/or cell interconnect I^2R heating effects. The container was compartmentalized to provide heat transfer from five surfaces of each cell to the container coldplate mounting surface. One electrical connector was provided for power transfer and charge control circuits and another for ground access to individual cell voltages. The battery container also contained a pressure relief valve.

2.7.2.3 Power Distribution System

The AM Power Distribution System received power from the AM Power Conditioning Groups at the AM Reg buses. Power was distributed from the AM Reg buses to the OWS buses for OWS loads; to the AM buses for AM, MDA, and certain OWS loads; and to the AM transfer buses for CSM loads. Transfer of power to the CSM buses utilized an umbilical cable across the MDA/CSM interface which was connected by the crew for each manned mission phase. The AM transfer buses also supplied power to, or accepted power from, the ATM for parallel operation of the AM and the ATM power systems in order to share all orbital vehicle loads.

The AM Power Distribution System utilized two separate isolated DC bus systems. These systems were two wire systems with the exceptions that the OWS buses, the ATM buses, and the CSM buses utilized a common return bus system. The negative return bus system was connected to vehicle structure at one point only, either the single point ground (SPG) in the AM or the vehicle ground point (VGP) in the CSM. Figure 2.7-12 is a simplified power bus system diagram which shows the intra-connections between buses in the AM and their interconnections with buses in other modules. All loads throughout the Skylab were powered from one of the Buses shown on Figure 2.7-12. Circuit breakers utilized by the AM Power Distribution System were located on STS Circuit Breaker Panels 201 and 202. The on-board controls and monitors for the AM Power Distribution System were located on STS Control Panels 205 and 206, with three exceptions: the AM transfer bus to the CSM bus interconnections were independently controlled from the CSM; the ON-OFF controls for the AM EREP buses were controlled from the ATM C&D Panel in the MDA; and the AM transfer bus to the ATM bus interconnections could also be opened in case of emergency by the ATM Power Off switch located on the ATM C&D Panel in the MDA.

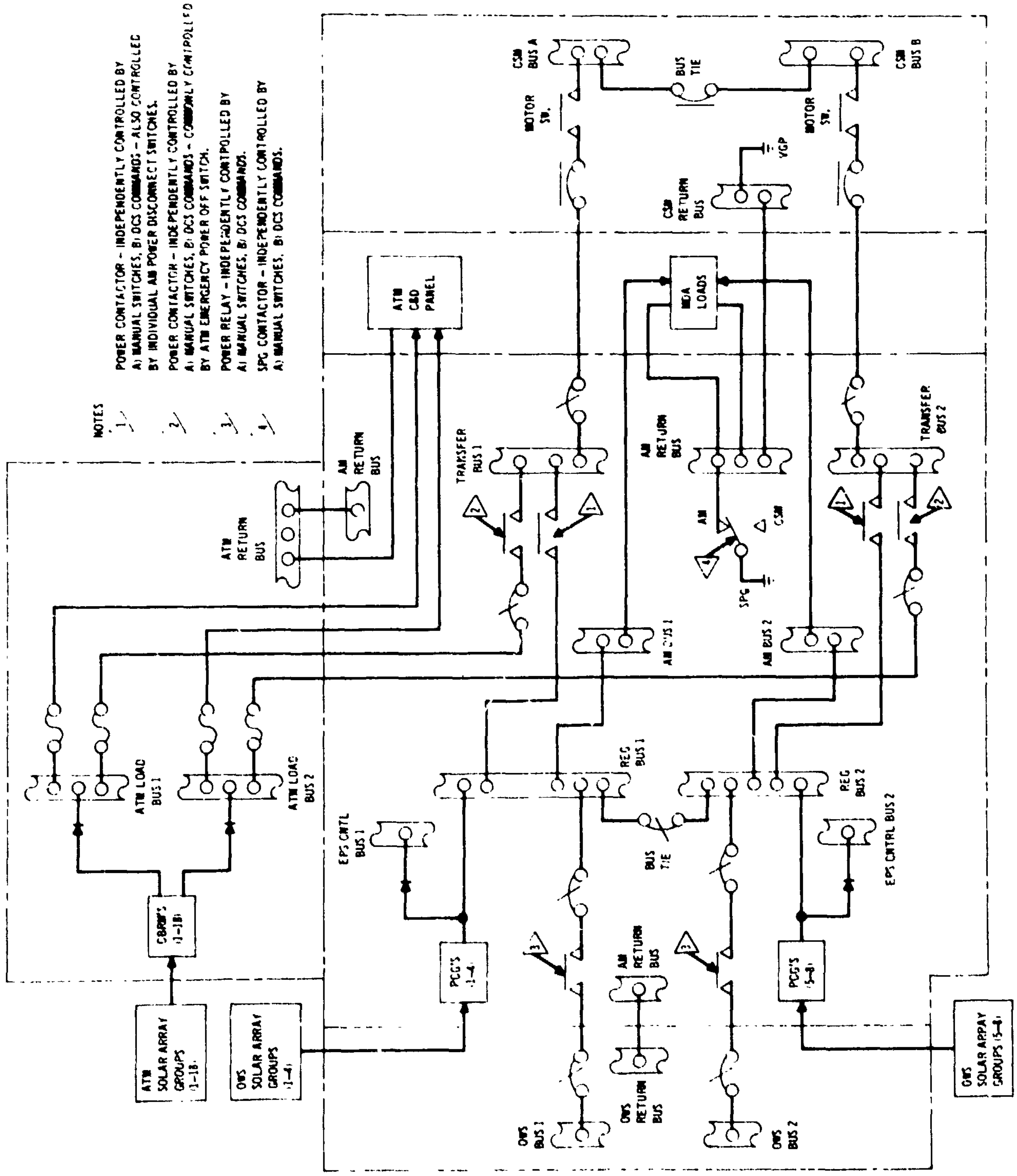


FIGURE 2.7-12 SIMPLIFIED ORBITAL ASSEMBLY POWER DISTRIBUTION DIAGRAM

A. AM Power Buses - The AM Power Distribution System included the following isolated positive buses, as shown on Figure 2.7-13.

- EPS Control Bus 1
- EPS Control Bus 2
- Regulated Bus 1
- Regulated Bus 2
- AM Bus 1
- AM Bus 2
- Transfer Bus 1
- Transfer Bus 2
- Deploy Bus 1
- Deploy Bus 2
- Sequential Bus 1
- Sequential Bus 2
- EREP Bus 1
- EREP Bus 2

The functions, interconnections, and controls associated with each bus in a set of isolated positive buses were identical. Loads were connected to each bus through protective devices, circuit breakers or fuses, to protect the distribution system.

Each EPS Control bus received power directly from four of the eight AM PCG voltage regulator outputs; EPS Control bus 1 from regulators 1 through 4 and EPS Control bus 2 from regulators 5 through 8. The regulator output to EPS Control bus connections were made through diodes in order to maintain bus isolation. The function of the EPS Control buses was to provide the power source for critical loads. The EPS Control buses were, therefore, hardwire connected to the regulators such that power could not be removed from these buses by means of astronaut or ground controls. Loads supplied from the EPS Control buses included: (1) equipment required for primary power system control by ground command or astronaut switching (in addition, the controls for PCG's 1-4 and for PCG's 5-8 were powered from EPS Buses 2 and 1, respectively, as a precautionary design feature), (2) lighting required for astronaut egress from AM/MDA/OWS in an emergency, (3) Caution and Warning System equipments, and (4) Digital Command System, (5) Command Relay Driver Unit, and (6) Electronic Timer.

Each Reg bus could be powered from any of the AM PCG voltage regulators but each regulator could be connected to only one of the Reg buses at a time. The standard operating condition was four regulators supplying each bus; 1 through 4 supplying Reg bus 1, and 5 through 8 supplying Reg bus 2. Power from the Reg buses was distributed to the AM buses and the transfer buses within the AM and to the OWS main buses in the OWS.

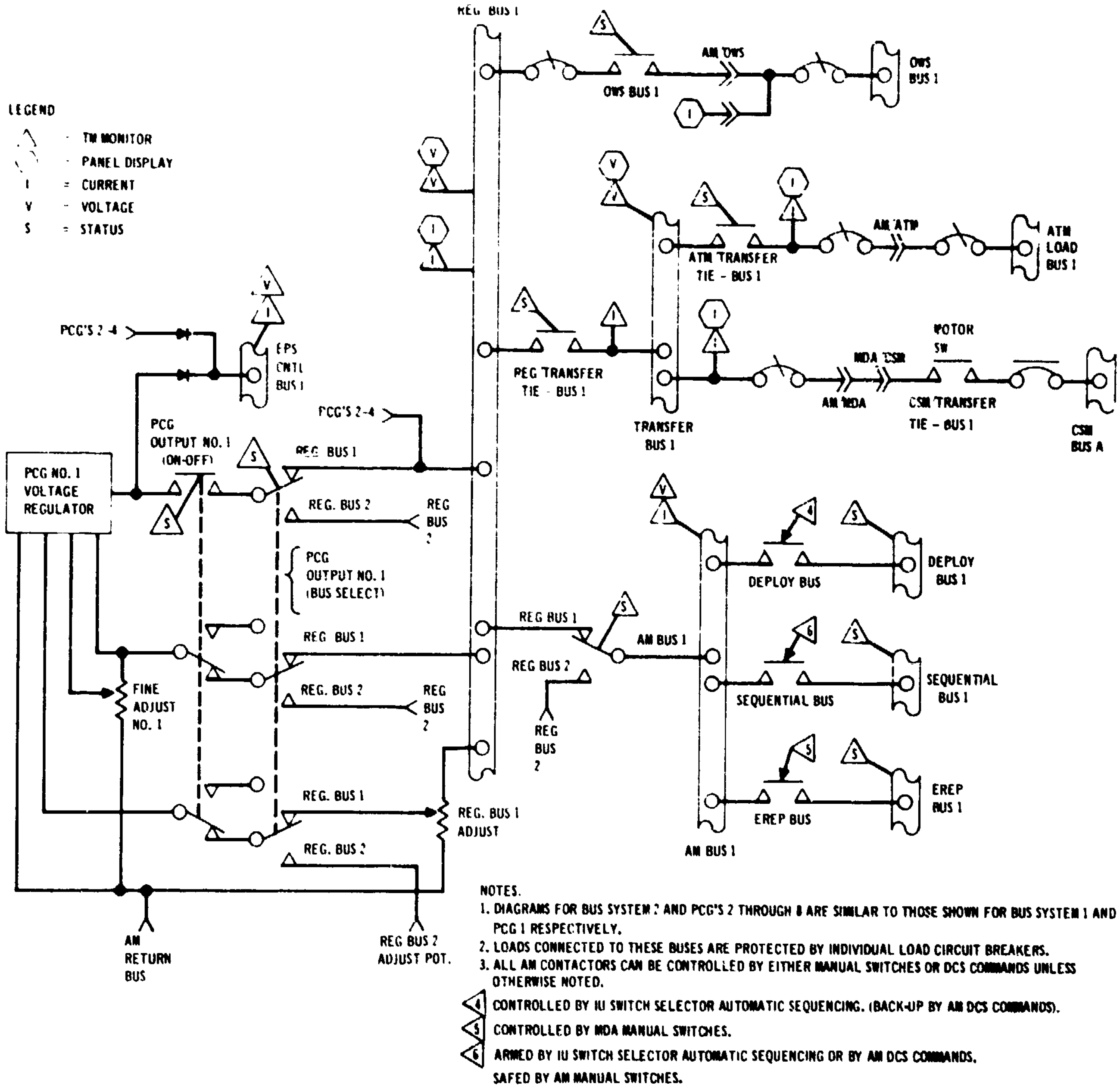


FIGURE 2.7-13 SIMPLIFIED BUS CONTROL AND MONITOR DIAGRAM

The AM buses provided power to all the loads in the AM, except those which were connected to the EPS Control buses. The AM buses also provided power to the loads in the MDA, to certain loads in the OWS; and to the Deploy, Sequential, and EREP buses.

The sequential buses provided the power required for payload shroud jettison, OWS radiator shield jettison, and ATM deployment. The deploy buses provided the power required for antenna, OWS solar array, OWS meteoroid bumper, and ATM solar array deployments. The deploy and sequential buses were disabled after the sequential portions of the SL-1 mission for purposes of safety.

The transfer buses provided the electrical power interface between the AM, ATM and CSM. Bidirectional power transfer between the AM EPS and the ATM EPS could be accomplished by connecting both the AM Reg buses and the ATM load buses to the transfer buses. The CSM, when part of the cluster, also had its power system normally connected to the transfer buses, as shown in Figure 2.7-12. Power for the CSM could, therefore, be supplied by either the AM or ATM EPS or by the parallel combination of the two EPS systems.

The EREP buses, located in the AM, provided power to the Earth Resources Experiments which were primarily located in the MDA.

B. Bus Control Functions - A simplified schematic of the controls associated with the positive power bus system is shown on Figure 2.7-13. For purposes of brevity and clarity, only the circuitry and components necessary to explain the control logic for the output of one AM voltage regulator and for one of the two independent bus systems, System No. 1, are shown on the diagram. The circuitry and components for the other seven regulator outputs and for System No. 2 were similar in function and were related as indicated on the diagram. The power return bus system has also been omitted in the interest of clarity.

The output of each voltage regulator was connected, through an isolation diode, to one of the EPS Control buses. There were purposely no controls associated with this power connection in order to ensure that the connection could not inadvertently be opened. This ensured a continuous power source to the critical loads connected to the EPS Control buses. Each EPS Control bus was supplied from a specific set of four voltage regulators.

The output of each voltage regulator could also be connected to either of the two Reg buses by means of two controls. The PCG Output Bus Select control connected the output to either Reg bus 1 or to Reg bus 2 when the PCG Output On-Off control was in the On position. The Off position of the PCG Output On-Off control isolated the regulator output from either of the Reg buses.

The functions of the following controls were straightforward and as shown on Figure 2.7-13; OWS Bus 1, Reg/Transfer Tie-Bus 1, ATM/Transfer Tie-bus 1, and AM bus 1. One feature to be observed was that a single Reg bus could supply power to both AM buses by means of the AM bus 1 and AM bus 2 switches.

All of the control functions described in this section so far, with the exception of the adjustment pots, were controllable either by astronaut manual switching or by ground control commands. Inflight control of the EPS by the various astronaut manual switches was obtained when the Power System Control switch was placed in Manual position. When the switch was in the CMD (Command) position, control was possible only from the ground by means of AM DCS commands.

There were, however, several controls which were not controlled by the Power System Control Switch. The Power Disconnect switches 1 and 2 were for emergency power down of Reg buses 1 and 2, respectively. They were operational at all times by crew action only. Power Disconnect switch #1, when thrown to its Off position, disconnected the outputs of PCG's 1-4 and disconnected Transfer bus #1 from Reg bus #1. Power Disconnect switch #2, when thrown to its Off position, disconnected the outputs of PCG's 5-8 and disconnected Transfer bus #2 from Reg bus #2. The ELEC GND switch, which controlled the location of the single point ground, was also independent of the Power System Control switch position. The connections between the Transfer buses and the CSM buses were controlled from the CSM and were independent of the AM Power System Control. The connections between the ATM buses and the AM Transfer buses could also be opened in an emergency by an ATM Power Off switch on the ATM C&D panel which was independent of the AM Power System Control switch. The connections between the AM buses and the Sequential and Deploy buses were normally controlled automatically by the Sequential and Deploy systems, respectively. There were Sequential and Deploy switches on the STS panel to provide backup control for these buses.

In addition to the control logic functions discussed above, the Reg Bus Tie circuit breakers between Reg bus 1 and Reg bus 2 could also be considered as part of the control logic. By manual crew control of these two 26.4 amp circuit breakers, the two Reg buses could be operated in parallel. Parallel operation could be used to reduce the effects of unbalanced Systems 1 and 2 load demands or unbalanced Systems 1 and 2 power availability. The normal operating mode was with the Reg Bus tie circuit breakers closed.

C. Power Return and Grounding - The electrical power distribution system, as previously discussed, consisted of a two wire system employing separate buses for both power feeders and negative returns. The return buses were tied to vehicle structure at only one point. This connection to vehicle structure was accomplished in one of two locations. During periods when the CSM/MDA interface connectors were not mated, the grounding was via the SPG in the AM. During periods when the CSM was part of the OA with the CSM/MDA interface connectors mated, grounding was via the VGP in the CSM structure. The connection to the VGP in the CSM was automatic when the CSM/MDA interface connectors were mated. The control switching in the AM was used to connect and disconnect the SPG in the AM. This is shown on Figure 2.7-12. Control of the SPG connection in the AM was by either crew manual operation or by DCS command at all times.

D. Power Feeder Design and Protection - The power feeder lines between the various power and return buses consisted of multiple numbers of wires which were selected both for current carrying capacity and voltage drop requirements. As shown on Figure 2.7-13, circuit breakers were incorporated in the positive feeder lines between buses located in different Skylab modules, with a separate set of breakers located in each of the modules. In addition to these circuit breakers, adequate circuit protection was incorporated into power distribution circuitry to all equipment powered from the AM EPS buses. This circuit protection was comprised of circuit breakers compatible with load requirements which protected the power distribution wiring from damage resulting from system overloads or short circuit conditions.

E. Shunt Regulator - The function of the shunt regulator was to prevent the occurrence of an overvoltage on the AM EPS buses as the result of a PCG voltage regulator module failure. There were two shunt regulators in the AM EPS. One was connected to each of the EPS control buses.

A shunt regulator consisted of a sense circuit, a drive circuit, and a transistor regulator bank of parallel power transistors. Figure 2.7-14 shows a block diagram and a static V-I curve for a shunt regulator. The sense circuit monitored the terminal voltage of the shunt regulator which was the EPS Control bus voltage. When this voltage exceeded a preset level in the range of 30 to 32 volts, the sense circuit signaled the drive circuit to turn on the Parallel Regulator Power Transistors. Since the regulator transistors were connected across the EPS Control bus, their increased collector currents produced an increased load on the bus. The effect of this increased load was to reduce the bus voltage because of the loading effect on the power source and the increased voltage drops from the power source to the bus.

The regulation capability of the shunt regulator is illustrated by its V-I characteristic. Below its sense voltage, the shunt regulator drew negligible current (less than 100 milliamperes). Above the sense voltage, its V-I characteristic exhibited a dynamic impedance in the range of 0.67 to 6.7 milliohms. This very low dynamic impedance was produced by the high gain from the sense circuit input voltage to the transistor regulator bank load current. This high gain, and the corresponding low dynamic impedance provided the shunt regulator with the capability to draw sufficient load current to limit the bus voltage to the desired level. At the same time, the current drawn by the shunt regulator insured the rapid clearing of the fuses in any failed module of the voltage regulator.

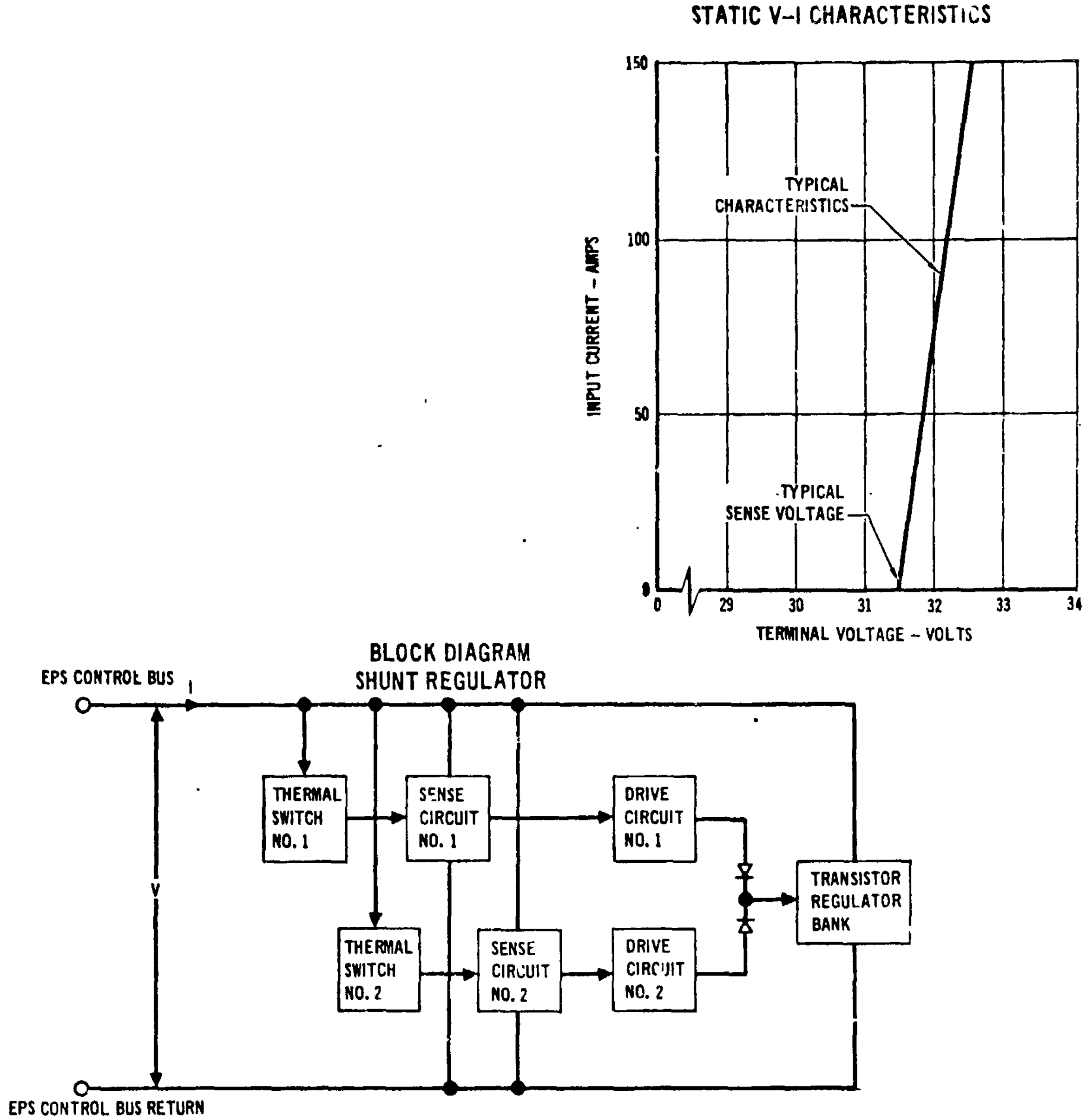


FIGURE 2.7-14 SHUNT REGULATOR

2.7.2.4 Manual and DCS Control Functions

Primary control of the AM EPS was by means of either manual control provisions installed on STS instrument panels in the AM or by means of AM DCS commands from ground control. The functions which were controlled included those associated with PCG control (see Figure 2.7-4), and those associated with power distribution bus control (see Figure 2.7-13). Additional material detailing the EPS control functions that were controllable by manual switching and by DCS command signals can be found in MDC Report E0195.

2.7.2.5 Display and Telemetry Parameters

A number of AM EPS analog parameters were displayed on meters installed on the STS instrument panel in the AM. These parameters were displayed to indicate instantaneous power system status to the astronauts and to assist the astronauts in their manual management of the system. A greater number of parameters were monitored and transmitted by means of telemetry to ground control to aid in ground control analysis and management of the EPS. The status of certain EPS control functions was monitored by means of bi-level signals which were transmitted by telemetry to ground control. These signals also aided ground control in their analysis and management of the AM EPS. More detailed information can be found in Report MDC E0195. Figures 2.7-4 and 2.7-13 show the schematic locations of the display and telemetry points associated with the PCG's and the power distribution system respectively.

2.7.2.6 System Operation and Performance

The AM EPS was a complex and flexible electrical power conditioning and distribution system as a result of its many capabilities and controls. Most of this flexibility, particularly in the PCG area, was not required for normal mission operations. It was associated with maintaining the highest possible level of system performance and reliability in the event of any possible malfunction in system equipment.

This section describes the normal modes of operation and defines performance characteristics. Performance under secondary or contingency modes would only be a modification of that described in this section and could be determined based on this section and the information throughout Section 2.7.2.

A. Power Capabilities - There were two power capabilities which were pertinent to evaluating the AM EPS mission performance. These were the solar inertial (SI) and nonsolar inertial power capabilities. The solar inertial refers to the continuous power capability of the EPS while the Skylab vehicle orbited in the solar inertial attitude. The nonsolar inertial refers to the power capability on a per orbit basis for orbits where the solar inertial attitude was not maintained over the entire orbit. There were two types of nonsolar inertial attitude orbits for the Skylab mission. These were the Z-Local

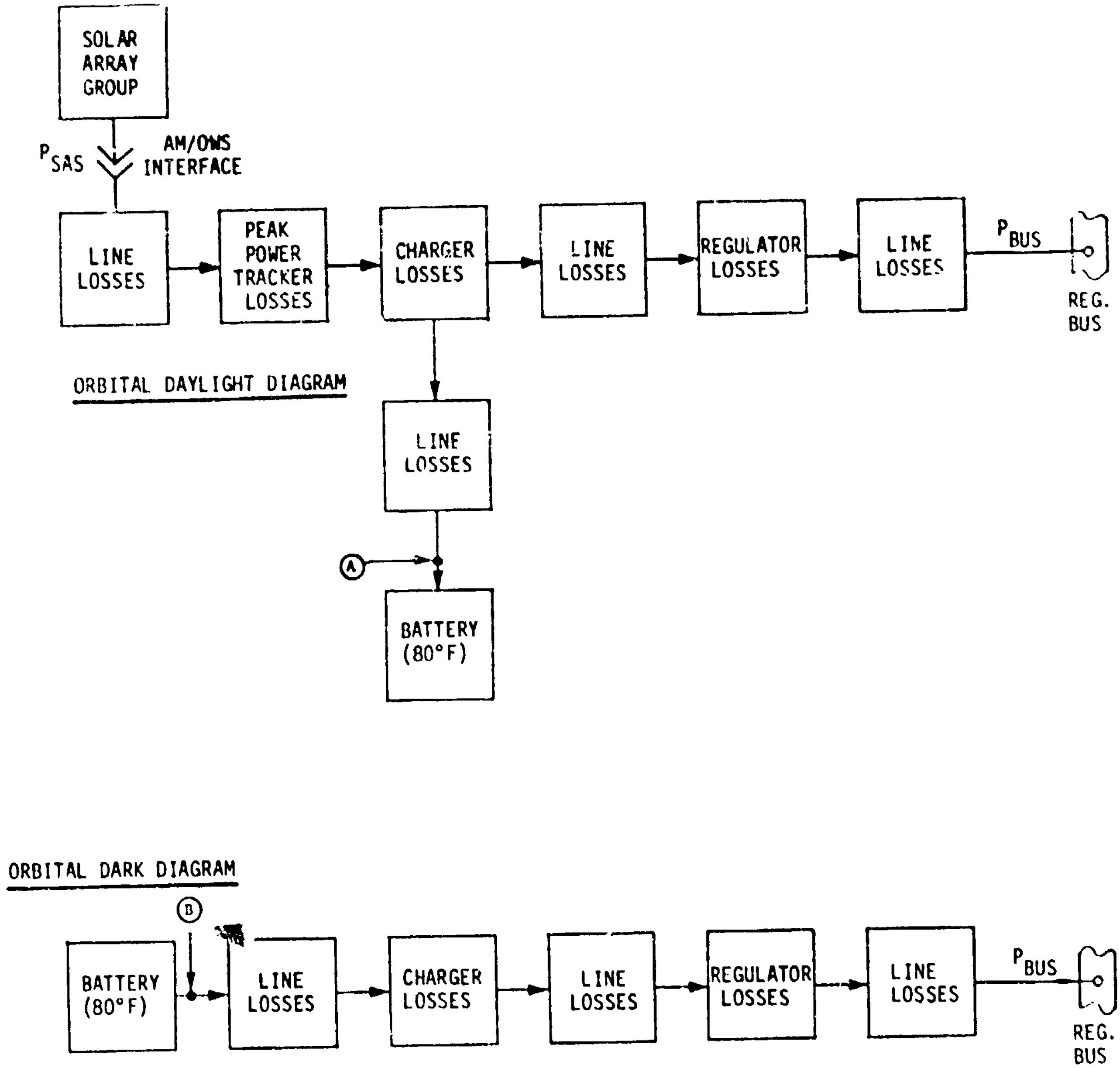
Vertical (Z-LV) orbits performed for Earth Resources Experiment Package (EREP) operations and the rendezvous orbits performed for docking and undocking of the CSM. The following sections define and explain each of these capabilities for an individual PCG and then explain the combining of the individual PCG capabilities into an overall AM Reg Bus power capability.

(1) PCG Solar Inertial Capability - The SI power capability is defined as the constant load supplied to the Reg Bus at which the PCG battery became fully recharged (a 100% SOC amp-hour meter indication) coincident with the end of the daylight charging period.

Orbital parameters affected the solar array power input to the PCG. The operational characteristics of the PCG equipments affected how much of the input power could be delivered to the loads. In the solar inertial attitude mode, the plane of the arrays was maintained perpendicular to the sun's rays, resulting in constant maximum array output power. The vehicle altitude determined the length of an orbit period. The Beta angle, which is the angle between the sun-line and the orbit plane, determines the length of the dark and daylight portions of each orbit period. The specified average solar array power available to an individual PCG during a Beta = 0° SI orbit daylight period was 1312 watts. The Beta = 0° SI orbit will be used throughout this discussion as the basis for a representative numerical value analysis. The length of the daylight period, therefore, determined the total amount of solar array energy available, and also the amount of time available to recharge the battery. The length of the dark period, on the other hand, determined the length of time the battery had to supply power to the bus and this fixed the total energy removed from the battery.

The PCG equipments whose operational characteristics affected the PCG output power capability were the battery, the battery charger, and the voltage regulator. The recharge characteristics of the battery determined how much energy could be returned to the battery during a daylight period. These characteristics were dependent on operating temperature and inherent battery design features. The battery charger had several operating limits affecting output power. It had a 75 ampere total output current limit and an approximately 55 ampere limit on charge current to the battery. It also had a 1500 watt continuous output power rating based on thermal limitations. The voltage regulator had an output current capability of 65 amperes. Approximately 1750 watts maximum could be delivered to the bus. The battery charger and voltage regulator also had efficiency characteristics which contributed power losses. The two block diagrams on Figure 2.7-15 show all the power losses which must be included in the power calculations. The two diagrams illustrate the orbital daylight and orbital dark cases. These two cases must be solved simultaneously for two conditions to satisfy the SI power capability definition. The power to the bus (P_B) must be the same, and the energy returned to the battery (point A) during daylight must return the SOC to 100% to restore the energy removed from the battery (point B) during the dark period. For the 1312 watt average input from the solar array at Beta = 0°, each PCG had a SI power capability of 536 watts at a Reg Bus. This power capability increased as the Beta angle increased because of the increase in daylight time and corresponding decrease in dark time. Beta angles above 69.5° constituted the special case of all sunlight and the capability was a maximum because only a very small amount of trickle charge power was required by the battery and all the rest of the power could be delivered to the bus.

The significance of the SI power capability is illustrated by the curves shown on Figure 2.7-16. The curves are simplified curves of battery SOC versus elapsed orbit time for several bus load conditions. Each curve starts with the battery SOC at 100% at the beginning of a dark period and shows a decreasing SOC during dark periods and an increasing SOC during daylight periods. The curve for load equal to PCG continuous power illustrates the SI power capability condition where the battery SOC just reaches 100% at



NOTE: FOR MAXIMUM CONTINUOUS BUS POWER, THE ENERGY (AMP-HOURS) RESTORED TO THE BATTERY (POINT A) MUST BE EQUAL TO THE ENERGY REMOVED FROM THE BATTERY (POINT B) MULTIPLIED BY THE APPROPRIATE BATTERY TEMPERATURE RETURN FACTOR.

FIGURE 2.7-15 CONTINUOUS PCG POWER DETERMINATION DIAGRAMS

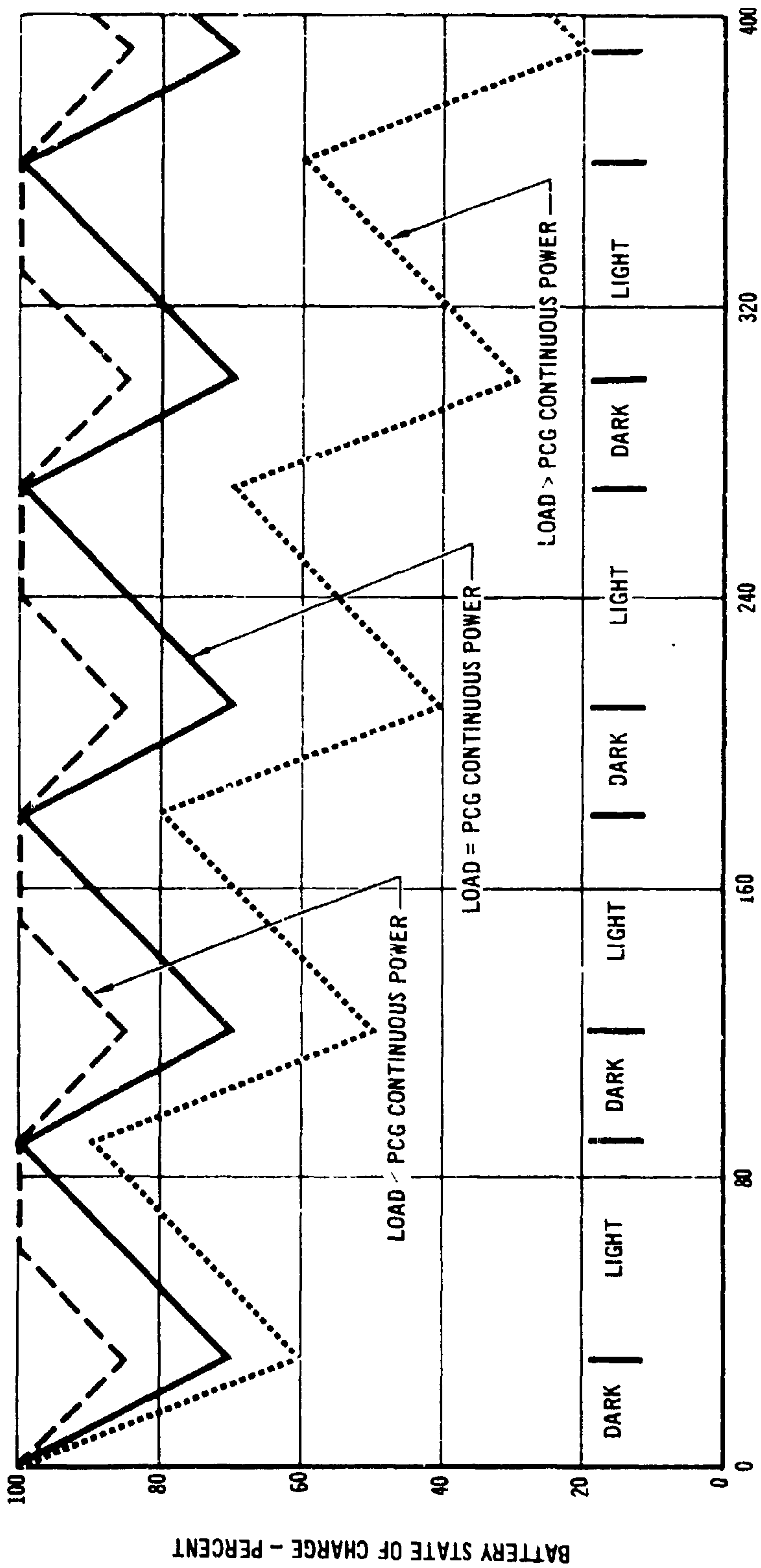


FIGURE 2.7-16 BATTERY STATE-OF-CHARGE VERSUS ORBITAL TIME FOR VARIOUS LOAD CONDITIONS

the end of the daylight or recharge period. The upper curve, where the load is less than the PCG continuous power illustrates that for lighter than SI power capability loads, the battery did not discharge as much and consequently was fully recharged prior to the end of the orbit. The third curve, where the load is greater than the PCG continuous power, shows why the SI power capability was an important operational limit on the AM EPS. It shows that if a load caused the battery to discharge to a depth such that it could not be fully recharged during the next daylight period, this same load would produce the same result in subsequent orbits and the battery would eventually become fully discharged and retain no stored energy.

(2) PCG Z-LV Capability - Nonsolar inertial attitudes were normally grouped together and referred to as the Z-LV attitude. The Z-LV power capability was defined as the average load (over the Z-LV period) supplied to the Reg Bus at which the minimum battery DOD during the Z-LV period was 50%. The Z-LV period started with the battery at 100% at the beginning of the dark period prior to the Z-LV daylight period and ended at the end of the dark period following the Z-LV daylight period. There were two factors which could markedly decrease the available solar array power during non-solar inertial orbits. These were the orientation of the solar array surface with respect to the sunline and the shadowing of array surfaces by the other parts of the vehicle. In solar inertial both the pitch and roll attitudes of the vehicle were controlled so that the array surface was perpendicular to the sunline. This produced maximum solar array output power since the power was proportional to the cosine of both the sunline to vehicle pitch axis angle and sunline to vehicle roll axis angle. In Z-LV, the sunline to pitch axis angle varied constantly from as much as -90° to +90° so that the power could go as low as zero during the daylight period. The sunline to roll axis angle was equivalent

to the Beta angle so the resultant power decrease during Z-LV was also proportional to the Beta angle.

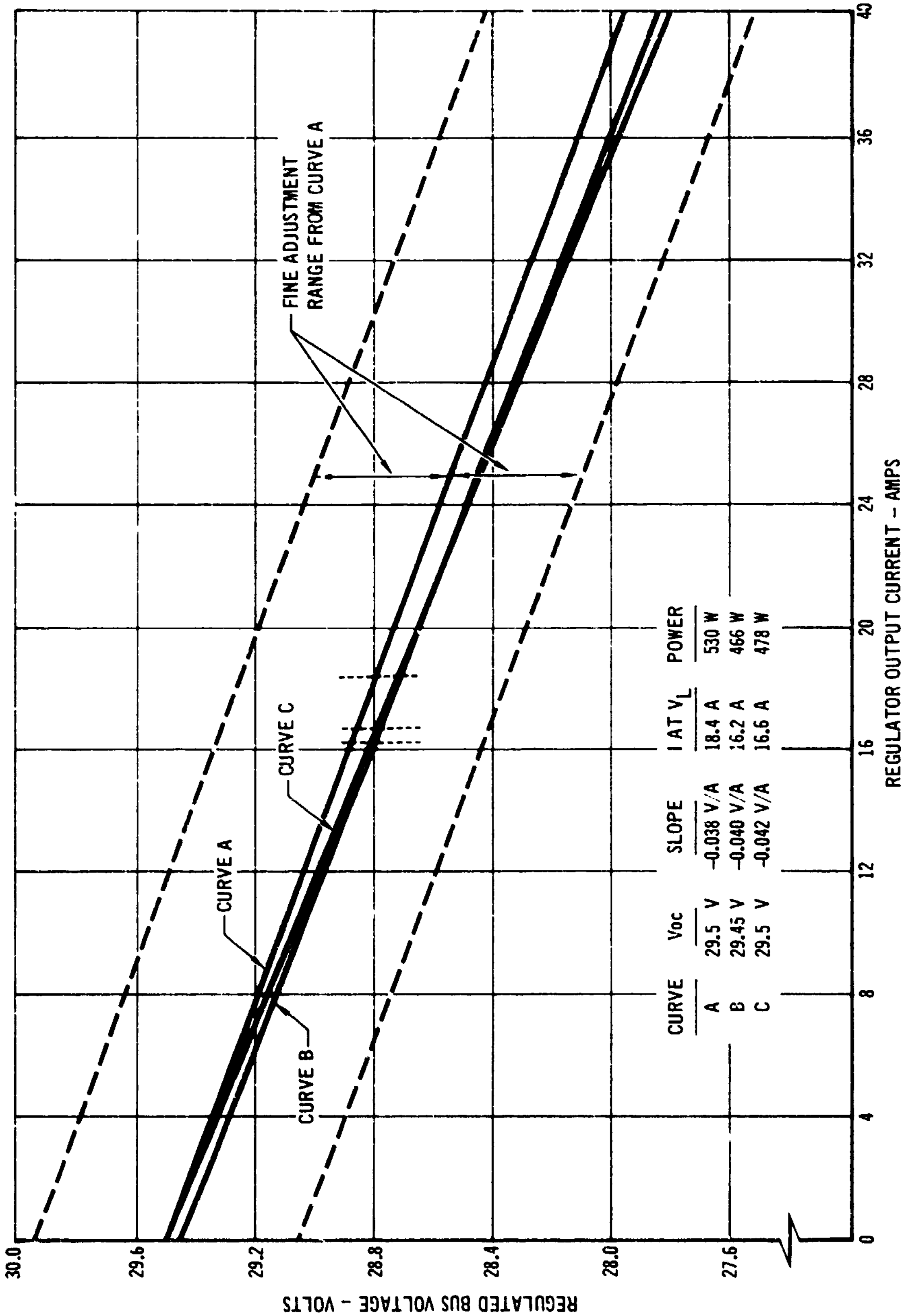
Shadowing of the OWS solar array wings could result from the ATM solar array wings, the OWS array fairing and the body of the OWS vehicle. The ATM shadowing occurred when the sunline to pitch axis angle changed. As the angle changed, different modules were shadowed so that different PCG inputs were affected. The amount of this shadowing and the modules shadowed were also affected by the Beta angle. As the vehicle roll axis angle changed for Z-LV, the body of the OWS shadowed the inboard array modules. Since the wings were at different elevation attachment levels on the OWS body, the effect of body shadowing was different for + and - Beta angles.

A typical Z-LV orbit included a SI to Z-LV maneuver, the Z-LV operational period, and a Z-LV to SI maneuver. Both attitude and shadowing were affected by the period of each maneuver and of the Z-LV operation, and by the orbital position at which they occurred. Because of all these variables, the solar array input power could be insufficient to supply the total bus load during certain portions of the daylight period. At such times the battery would be required to supply the rest of the bus load. Such periods would effectively be dark periods for the battery since the battery was supplying energy instead of receiving recharge energy as it normally did in daylight periods. As soon as the solar array input power exceeded the bus load requirement, the excess power was again used to recharge the battery.

The Z-LV power capability was calculated for each Z-LV case individually because of all the variables and their interdependence on one another. In effect the Z-LV capability involved the trade-off between the type and duration of the maneuvers, and the amount of bus power that could be supplied. Following a Z-LV period, several orbits of SI were required in order to restore the battery to a 100% SOC. The number of orbits depended on the actual minimum battery SOC.

(3) AM EPS Solar Inertial Capability - The AM EPS solar inertial capability was the total continuous power that the 8 PCG's could supply to the Reg buses with none of the PCG's operating above its individual power capability. Each PCG, though identical in physical construction, had some difference in performance. The most significant performance characteristics that differed from PCG to PCG was the amount of solar array power available, the battery recharge characteristics, and the voltage regulator output characteristics. The array power differences could occur because of shadowing and temperature vibrations. Each battery had a somewhat different recharge characteristic affecting the amount of energy required for recharge and the rate at which the energy would be accepted by the battery. The voltage regulator characteristics were important in the way that the individual PCG outputs could be combined at the Reg buses into a total capability.

The AM EPS power capability was based on the two Reg buses operating in parallel (Reg bus tie C/B's closed), and with each bus supplied by 4 PCG's. The arithmetic sum of the individual PCG capabilities of 536 watts was 4288 watts. This value, however, could only be used as an optimum limit value because of practical considerations. The output characteristic of the voltage regulator in each PCG had a slope of -0.04 ± 0.002 volts/ampere. In addition the output level of each regulator could shift by a maximum of 0.05 volts due to temperature variations, ageing, and drift. Figure 2.7-17 illustrates the effects of slope variations and voltage level variations on combining PCG outputs. It does this by showing several regulator output V-I curves having different characteristics (within specification limits) and determining their outputs at a common operating voltage level. As shown in the tabulation on the figure, regulators B and C were delivering less power than regulator A.



CURVE	Voc	SLOPE	I AT V_L	POWER
A	29.5 V	-0.038 V/A	18.4 A	530 W
B	29.45 V	-0.040 V/A	16.2 A	466 W
C	29.5 V	-0.042 V/A	16.6 A	478 W

FIGURE 2.7-17 REGULATOR OUTPUT VOLTAGE AND CURRENT CURVES

The output voltage level of each regulator could be controlled relative to the other PCG's by the associated fine adjustment potentiometer. The range of the adjustment is shown on Figure 2.7-17 for regulator A by the dashed line slopes. This capability was designed so that the previously listed variations between PCG's could be overcome along with possible contingencies such as module failures in a battery charger or voltage regulator. This adjustment capability, however, could also introduce some small unbalance between PCG's. For purposes of analysis it was assumed that the fine adjustment potentiometers overcame all variations except the regulator output characteristics as illustrated by Figure 2.7-17. Based on this assumption the SI total power capability was 3930 watts compared to the optimum possible value of 4288 watts.

The total power capability for operating the two Reg buses independently remained the same at 3930 watts or 1965 watts per bus, if the buses were equally loaded. If the loads on the two buses were not equal, only the capability of the heavier loaded bus could be fully utilized and this reduced the total output power to less than 3930 watts. The paralleled bus system therefore had the advantage of improving the utilization of the total AM EPS continuous power capability.

(4) AM EPS Z-LV Capability - The AM EPS power capability for Z-LV was the total of the individual capabilities with none of the batteries going below the 50% SOC limit. The preceding discussion on combining individual PCG capabilities for SI also applies to the Z-LV case. However, the shadowing encountered during Z-LV, as previously discussed, was the predominant factor affecting the Z-LV total capability. Depending on the Beta angle and the maneuvers there could be a wide range of battery SOC's during a Z-LV period. Therefore, all were limited to power capabilities consistent with the most shadowed one reaching the 50% SOC limit during the period. Once again, the Z-LV capability had to be calculated for each specific set of Z-LV orbital conditions.

B. Parallel Operation With ATM EPS - The AM EPS could be operated in parallel with the ATM EPS to supply total cluster load requirements. The objective of operating the AM and ATM electrical power systems in parallel was to utilize the full capability of both of the EPS systems to satisfy cluster load requirements. Parallel operation eliminated cases where one power system might be overloaded while the other system had power available in excess of its individual load demand. Paralleling therefore, allowed load levels to be based on total cluster EPS capability instead of being based on individual EPS capabilities.

Paralleling was accomplished by connecting the AM Reg. buses and the ATM buses to the Transfer buses in the AM. The load sharing between electrical power systems was controlled by means of the AM Reg. bus adjustment potentiometers. These potentiometers adjusted the overall AM Reg. bus V-I curve with respect to the overall ATM Load bus V-I curve. The operation was similar in nature to that of the fine adjustment potentiometers in controlling load sharing between AM PCG regulators. Therefore, some loss in power capability must also be assumed when attempting to share a specific load between the AM and ATM electrical power systems.

A 3% loss in AM Reg. bus power capability had been assumed for paralleling losses. This reduced the AM Reg. bus continuous power capability from 3930 watts to 3814 watts. The capability specified for the ATM EPS was 3716 watts. A total capability of 7530 watts was thus provided, with the 3814 watts supplied by the AM EPS and the 3716 watts by the ATM EPS.

The amount of power transferred between the AM and ATM could be expected to vary over a wide range. To account for contingency conditions, the requirement for allowable power transfer was set at a maximum of 2500 watts. Therefore, the AM EPS distribution system wiring and control equipments which were associated with AM/ATM power transfer were designed to transfer 2500 watts continuous in either direction, as measured at the AM Transfer buses.

C. CSM Connected Operation - The CSM could be powered from the parallel combination of the AM and ATM electrical power systems or from either power system individually during manned phases of the Skylab Program. Interconnection of the AM, ATM, and CSM power bus systems could be accomplished by use of the Transfer buses in the AM. Connection of the AM Reg buses and the ATM buses to the Transfer buses was accomplished by use of controls and switching provided in the AM EPS. The connection of the CSM buses to the Transfer buses was accomplished by use of controls and switching provided in the CSM. When the AM Reg buses were connected to the Transfer buses, the Reg bus potentiometers could be used to adjust the voltage levels on all buses connected to the Transfer buses. They could once again be used to make load sharing adjustments when the AM and ATM were operating in parallel and supplying power to the CSM. The use of proper procedures in adjusting these potentiometers provided the required voltage levels and load sharing. It was possible, however, to adjust these potentiometers out of the desired ranges for either voltage level or load sharing. This condition could occur because of the necessity for the potentiometers to have a range such that required bus levels could be maintained under various conditions. The use of proper procedures assured operation at desired voltage levels and load sharing conditions.

The wiring and control equipments in the AM EPS distribution system which were associated with the transfer of power to the CSM were designed to deliver 2472 watts continuous (1236 watts/bus) as measured at the AM/MDA interface.

An essential operation, related to CSM connected operation, was the operation of the ELEC. GND. Control on the AM STS instrument panel. When the CSM docked and the CSM/MDA interface connectors were mated, the negative return bus system was automatically connected to vehicle structure by the VGP in the CSM. The connection to the vehicle structure by way of the SPG in the AM was also present at that time. It was therefore necessary to position the ELEC. GND. Control on the STS instrument panel to its CSM position in order to disconnect the AM SPG from vehicle structure. Prior to CSM undocking the ELEC. GND. Control was positioned to its Airlock position to reconnect the AM SPG to vehicle structure for orbital storage periods.

2.7.3 Testing

Testing of the AM EPS was conducted throughout the Airlock program at the component, black box, subsystem, system, and flight vehicle levels. The objectives of all the test programs were to assure as much as possible, by testing that the flight vehicle AM EPS could be expected to meet all the Skylab requirements with a high level of confidence. The overall testing can be divided into three categories: qualification; development and confidence; and flight vehicle testing. The timely progression of testing can be seen on the AM EPS testing History chart on Figure 2.7-18.

Qualification testing was required on all individual components and functional units which were to comprise the AM EPS. The qualification testing on the major functional units is discussed in detail in the following section.

Development testing was concentrated in four areas. These were the battery life test area, battery charger design area, voltage regulator design area and the PCG and subsystem area. Development and confidence tests of the AM nickel-cadmium (Ni-Cad) batteries were performed both at MDAC-E and at the battery vendor. An extensive series of tests were performed because of the importance of battery life characteristics to the success of the Skylab mission. The battery charger and voltage regulator vendor performed a substantial amount of development testing in order to meet specification requirements which represented significant advances in the state-of-the-art for power conditioning equipment. The system development test verified individual PCG operation, parallel PCG operation, and the operation of the power distribution system. This test included ambient and temperature-altitude conditions to verify operations under mission environments.

The Silicon Controlled Rectifier (SCR) Panel development test was conducted as a result of a design problem which was encountered during Spacecraft Systems testing. The generation and incorporation of the SCR circuits to overcome the problem had to necessarily be done on an expedited basis so as to minimize the impact on vehicle testing and delivery schedules. The testing was performed to verify the effectiveness of the solution and to enhance the confidence in its flightworthiness.

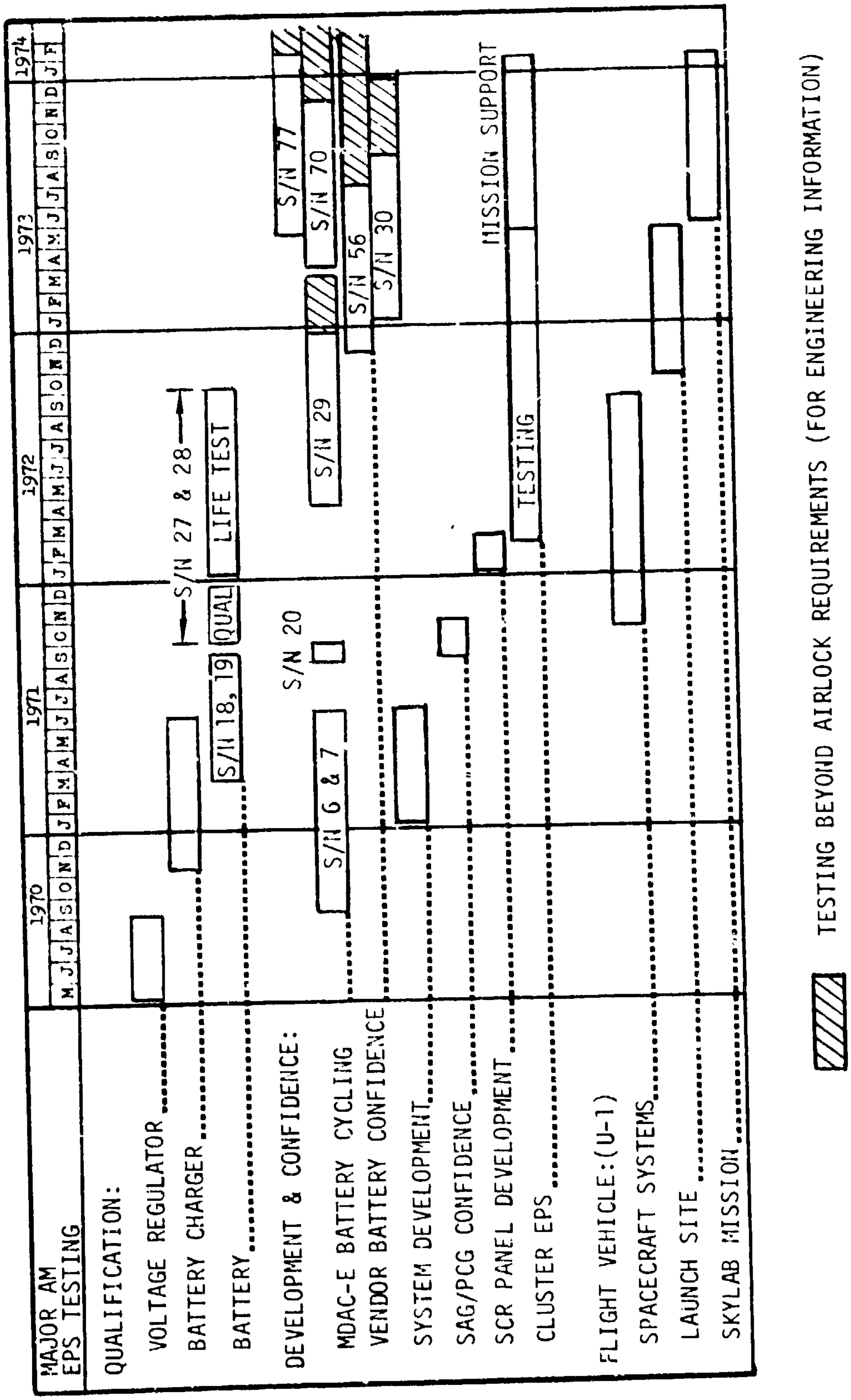


FIGURE 2.7-18 AM EPS TESTING HISTORY

The compatibility of the AM EPS and its interfaces was verified during the development phase of the program by two independent test programs. The Integrated SAG/PCG test verified the interface between the individual Solar Array Group (SAG) of the OWS and the individual Power Conditioning Group (PCG) of the AM EPS. The interfaces between the AM EPS, and the ATM and CSM electrical power systems were verified during the Cluster EPS development test at MSFC. The Cluster EPS consisted of flight type AM and ATM systems and a simulated CSM system. The Cluster EPS testing preceded the actual mating of the flight vehicle power systems and therefore provided a timely early verification of overall EPS compatibility.

The final EPS test before delivery of the Airlock Module by MDAC-E was during the Spacecraft Systems testing on the U-1 vehicle. This primarily verified the compatibility of the AM EPS with all other AM and MDA systems and verified the flightworthiness of the actual flight system of the U-1 vehicle.

Further testing at the launch site verified interfaces with the ATM and CSM EPS and maintained the status of the system up to the launch. The mission performance monitoring after launch continued to check and verify that the AM EPS was meeting all mission requirements.

2 7.3.1 Qualification and Acceptance Testing of Major Hardware Units

A. Battery Charger

1/ Qualification Testing - The battery charger was subjected to and successfully passed an extensive qualification test. The qualification test procedure QTR 714244 defined the testing methods and procedures necessary to comply with the qualification test requirements specified in MDAC-E Procurement Specification 61B769006. Two qualification samples, identical to the actual flight units except for internally mounted thermocouples, were utilized during the testing. The qualification testing included temperature altitude, random vibration (operating), humidity, electromagnetic interference (EMI) and life testing. Fungus resistance requirements were satisfied by vendor certification that all of the materials used in the battery charger are not nutrients to fungus. The only design change to the battery charger that was required as a result of the qualification test program was the addition of a moisture seal to improve the ability of the unit to withstand humid environments. This change was incorporated after the battery charger failed to pass humidity testing. Following a successful retest, the moisture seal was incorporated on all units. Qualification Test Report QTR #2639 is a comprehensive report on the qualification program. All pertinent information relating to the qualification test program is contained in this report.

Two battery chargers successfully passed environmental sections of the qualification test. In addition, one sample was subjected to and passed the 1000 hour life test. The other sample successfully accumulated in excess of 15,000 hours during MDAC-E cycling tests on 61B769004-13 S/N 29 and -19 S/N 70 batteries.

2/ Vendor Acceptance Testing - All battery chargers were subjected to an extensive acceptance test prior to shipment from the vendor's facility. The testing included both performance and environmental tests. The battery charger acceptance testing was performed with the aid of a complex test console. The console included a solar array simulator, a battery simulator and a flight type voltage regulator. Thus, the acceptance testing was performed under conditions which closely represented the actual flight application. A test connector was provided on each battery charger which when used in conjunction with the equipment in the test console, allowed detailed measurements with accuracy not attainable in the actual flight system. The internal redundancy of the battery charger was verified in so far as was practical.

All phases of battery charger performance, encompassing all specification requirements, were checked during the acceptance test. Complete performance verification tests were performed immediately following the non-operation environmental tests of thermal shock and random vibration. The resulting data was recorded in a test record which constitutes a matrix of performance data for each individual unit. The acceptance test procedure ATP 714243 can be consulted for the detailed test procedures and sample test record forms.

3/ Pre-Installation Acceptance Testing - All battery chargers were subjected to pre-installation acceptance (PIA) testing upon receipt of the unit from the vendor. The PIA test was repeated if the unit was held in storage for more than 12 months. The PIA test is essentially a repeat of the physical examination and room temperature performance portions of the vendor acceptance test. The temperature shock, burn-in and random vibration tests of the ATP were not repeated in the PIA test. A test console identical to the vendor test console was used for PIA testing.

B. Battery

1/ Qualification Testing - Qualification tests were conducted at the vendors facility per test procedure QTP 107 in the period between May 1971 and October 1972. The batteries were tested for design adequacy when subjected to the environmental, operational, and life requirements commensurate with integration into the Skylab EPS system.

Simulation of flight environmental and operational conditions for various qualification tests required design and fabrication of some major pieces of test equipment. Battery thermal control was accomplished utilizing spacecraft coolant and a special coolant bench plumbed to flight configuration battery coldplates. This bench provided the capability to adjust flow and inlet temperature conditions to simulate anticipated levels of spacecraft operation. Radiated heat transfer was negligible during altitude-temperature testing and was precluded during ambient life testing by encapsulation of the batteries in a granulated insulation material. Charge and discharge controls which simulated those commensurate with flight conditions were provided for each battery by a special automatic test console. These consoles provided; programmed loads, adjustable day-night cycle periods to compensate for Beta angle variations, adjustable charge voltage limit and recharge fraction to compensate for battery temperature variations, and battery level data monitoring and recording capabilities. Later configuration batteries had provisions for monitoring individual cell voltages and a data acquisition system console was added to the test complex to monitor cell voltages. All qualification tests were performed on complete batteries with the exception of the burst pressure test of cell and battery containers. The tests on complete batteries consisted of Temperature-Shock, Vibration, Acceleration, Pulse Discharge, Installed Storage, Temperature-Altitude and Cycle Life.

61B769004-9 configuration batteries, S/N 18 and 19 were the initial qualification test specimens. These units completed all the qualification tests with the exception of the Cycle Life test. Testing of this configuration was stopped when information became available from Engineering Confidence Life Testing that a plastic plate hold-down block within the cell caused increased plate deformation with cycle accumulation eventually resulting in cell shorts. A new 61B769004-13 configuration was established in which the plate hold-down block was eliminated. Two specimens, S/N 27 and S/N 28 of the Dash 13 configuration were subsequently subjected to Vibration, Acceleration, Installed Storage and Life Testing. These tests were successfully passed by the 61B769004-13 configuration specimens with the exception of the Life Test. Battery S/N 27 satisfied simulated mission loads at required voltage levels throughout the 4000 cycle life requirement, however, S/N 28 failed the voltage requirement on cycle 3029. Life testing was concluded on 3 October 1972 when S/N 27 accumulated its 4000th cycle. Detailed results of the Qualification tests can be found in report QTR 107. Failure analysis results indicated the basic problem experienced with qualification units S/N 27 and S/N 28 was one of workmanship inconsistency in the plate tab shaping process. Confidence in the battery design concepts remained high based on failure analysis findings of the qual units and the excellent performance of a life test unit (S/N 29) of the same configuration at MDAC, which was six months into the eight month test. Because of possible workmanship problems within existing batteries, a decision was made to fabricate new batteries, 61B769004-19 configuration, for the flight by using a special tab shaping tool to assure controlled shaping of the tabs with an adequate strain relief loop for plate tabs interior to the cell pack. Faced with insufficient time to demonstrate life adequacy of the new batteries, two additional specimens of the 61B769004-13 configuration (S/N 30 and S/N 56) were placed on life test at the vendor in late October 1972 for the purpose of maximizing confidence in the design concept. Two batteries of the newly fabricated 61B769004-19 configuration (S/N 70 and S/N 77) were placed on life test at MDAC-E when they became available in March and May of 1973. Since that time, S/N 29 which was in the sixth month of a life test at MDAC in October 1972 and S/N 30 and S/N 56 which were placed on test at the vendor in late October 1972 have each subsequently demonstrated cycle life in excess of 4000 cycles without cell failure. The S/N 70 and S/N 77 batteries placed on test in March and May 1973, respectively, also satisfactorily completed 4000 cycles without a cell failure. MDAC-E tests on batteries S/N's 29, 70, and 77 were accomplished utilizing flight configuration chargers.

2/ Vendor Acceptance Testing - Acceptance testing was performed at the vendors facility per test procedure ATP-180 to verify compliance with the requirements of significant physical and performance criteria of the design specification. Data from these tests also provided engineering information pertinent to individual battery performance characteristics and overall design.

Acceptance tests included subassembly and final assembly testing. At the subassembly level battery cells were tested for performance and physical characteristics; the battery container was tested to verify proof pressure integrity; and the relief valves and thermal switch were verified for operation actuation points. Assembled batteries were tested to verify wiring integrity, temperature sensor indication consistency, and battery tolerance to anticipated launch vibration exposure. Battery operational characteristics were determined during mission type acceptance cycling which simulated flight charging and discharging conditions. During this cycling and subsequent capacity discharging the batteries were maintained at a constant (75° ± 5°F) temperature utilizing flight type coldplates and a test coolant bench. Physical inspection and dimensional verification completed the acceptance tests.

All batteries were subjected to and passed the acceptance test requirements. A significant design benefit derived from the evaluation of early acceptance test data was an early insight into a temperature gradient situation within the battery which was inconsistent with long cycle life. This condition was subsequently resolved by a change in the battery container material on later configurations.

3/ Pre-Installation Testing - The AM flight batteries (-19) completed the fabrication process and were delivered via NASA aircraft directly to KSC very near the time schedule for vehicle installation. The entire fabrication process and the acceptance test of these units at the vendors facility were witnessed by MSFC and MDAC Quality Assurance personnel. The KSC pre-installation testing was reduced to a visual inspection, a wiring integrity check and a full charge of the batteries for installation. A full capacity cycle, normally performed as part of the Pre-Installation Acceptance test was eliminated. The source inspection coverage, the special delivery arrangements, and the in-vehicle tests scheduled after installation warranted the time saving change.

C. Voltage Regulator

1/ Qualification Testing - The Voltage Regulator was subjected to and successfully passed an extensive qualification test. Qualification test procedure QTP 714282 defined the testing methods and procedures necessary to comply with the qualification test requirements specified in MDAC-East Procurement Specification 61B769005. Two qualification test samples, identical to the actual flight units except for internally mounted thermocouples, were utilized during the testing.

The qualification testing included temperature altitude, electromagnetic interference (EMI), humidity, random vibration (operating), and life testing. Fungus resistance requirements were satisfied by certification. All qualification testing was successfully completed without a regulator malfunction or failure. No design changes were incorporated as a result of the qualification test program. Qualification Test Report QTR 2586 is a comprehensive report on the qualification program.

2/ Vendor Acceptance Testing - All voltage regulators were subjected to an extensive acceptance test prior to shipment from the vendor's facility. The testing included both performance and environmental tests. The voltage regulator acceptance tests were performed with the aid of a test console. The console simulated the AM electrical power system in the areas of input voltage range, wire resistances, remote sensing, etc. A voltage regulator, identical to the flight voltage regulator, was wholly contained within the test console for the parallel operation testing. All phases of voltage regulator performance, encompassing all specification requirements, were checked during the acceptance testing. Complete performance verification tests were performed immediately following the environmental tests of thermal shock and random vibration. The resulting data was recorded on a test record form which constitutes a matrix of performance data for each individual unit. The acceptance test procedure ATP 714281 can be consulted for the detailed test procedures and sample test record forms.

3/ Pre-Installation Acceptance Testing - All voltage regulators were subjected to pre-installation acceptance (PIA) testing upon receipt of the unit from the vendor. The PIA test was repeated if the unit was held in storage for more than 12 months. The PIA test was essentially a repeat of the physical examination and room temperature performance portions of the vendor acceptance test. The temperature shock, burn-in and random vibration tests of the ATP were not repeated in the PIA test. All phases of the voltage regulator performance were tested in the PIA test. A test console identical to the vendor ATP console was used for PIA testing.

2.7.3.2 Development and Confidence Testing

A. Vendor Battery Testing - In addition to Acceptance and Qualification tests, battery and cell tests were conducted at the vendors facility to assist in design change decisions, to establish improved procedures for installed battery maintenance, and to increase confidence in the life capability of the design.

1/ Prototype Test - The initial development test conducted at the vendors facility was performed on a battery specimen made from cells fabricated from pre-AM plaques and a proto-type AM battery magnesium container. The objectives of the test were to determine the following: recharge amp-hour efficiencies at selected temperatures and rates for states-of-charge ranging from 25 to 100 percent; current charge characteristics at selected voltages, temperatures and states-of-charge; and waste heat generation during overcharge at selected temperatures and charge rates. The parametric data from this test program was to be used for power system performance calculations and computer simulations. One of the major difficulties experienced in this testing was the inability to establish the battery at an accurately predictable state-of-charge at the beginning of each amp-hour efficiency test sequence. Inefficiencies of the small magnitude anticipated for recharges less than 80% could not be established because of this problem. The test was terminated without attaining all of its objectives. However, the information obtained was useful in formulating a new definition of battery operation for power system calculation purposes.

2/ Battery Parametric Data Test - This test was a follow-on test to the Prototype Test. The test specimen was one of the early AM production batteries. The test was conducted to investigate the magnitude of the temperature gradient condition first detected during acceptance testing. It was also conducted to substantiate the validity of the new definition of battery operation with respect to full recharge efficiencies at various temperatures, and with respect to charge current profiles as a function of temperature, voltage limit and percent of discharge ampere-hours returned during the recharge. Test results verified that a significant temperature gradient problem did exist. Subsequent changes in case and cell potting materials and a coolant system change to lower coolant inlet temperatures were made to alleviate this problem. Test results substantiated the validity of the performance definition being used for power system analysis.

3/ Modified Cell Vibration Susceptability Evaluation - This evaluation was conducted at the vendors facility in June 1971 after the premature cell failures in the life testing of 61B769004 batteries were attributed to the effect of the plate hold-down used in these particular batteries. The tests were performed on three cells having different internal configurations. One had the existing hold-down used in Dash 3's; the second had a modified hold-down; and the third had no hold-down. The tests were run to determine the effects of random vibration environments on the cell integrity and performance. X-rays were taken after the significant steps of activation, condition cycling, and vibration exposures. Electrical performance was monitored for voltage stability and continuity during the vibration exposures. No detrimental effects were experienced by the three cells during or after any of the vibration exposures.

The use of the cell configuration without the hold-down to eliminate the life problem was therefore, also deemed acceptable from the vibration exposure aspect based on the results of these cell tests. Dash 9 batteries (S/N 27 and S/N 28) which incorporated the no hold-down cell configuration subsequently demonstrated their ability to successfully withstand the qualification random vibration exposure level of 7g(rms).

4/ Battery Maintenance Procedure Development - A battery maintenance procedure was necessary to maintain the charged AM batteries in a state of launch readiness during the period from their flight vehicle installation until the Skylab launch. The original procedure, recommended by the battery vendor, was to trickle charge for 5 hours on a weekly basis. When the trickle charge level was changed from 2.0 amperes to

0.75 amperes, it was deemed undesirable to change the procedure from 5 hours to over 14 hours. An alternate procedure was then recommended which was to use a constant potential boost charge on the batteries for one-half hour on a weekly basis. This procedure was then checked on qualification Dash 9 batteries at the vendor facility. The four week test resulted in some loss in ampere-hour status of the batteries. This precipitated a comparative test of several maintenance methods on a cell level basis. These tests were also performed at the vendors facility in September and October of 1971. A total of fourteen cells were used in this testing. The methods evaluated were: 1) a weekly constant potential boost charge for time durations of 1/2, 1, and 2 hours, 2) a weekly trickle charge period of 14 hours at 0.75 amperes, and 3) continuous trickle charging. Evaluation of the tests indicated that the 2-hour constant potential boost charge and both trickle charge procedures, periodic and continuous, were effective in maintaining the ampere-hour status of the batteries over an extended period of time.

The 2-hour weekly boost charge was, therefore, selected as the final procedure to minimize the impact on vehicle launch preparation activity, and to minimize coolant pump usage prior to launch. This procedure was subsequently verified on the battery level in the qualification test of Dash 13 batteries.

5/ Evaluation of New Production Tool - A new production tool was designed to cope with a fabrication workmanship problem evident from the October 1972 failure analysis of cells from the life test of Dash 13 battery. The subject evaluation program was conducted at the vendors facility in November 1972 to verify that plate straightness, adequate tab strain relief, tab shaping uniformity, and bottom of cell plate growth allowance would consistently result from the use of the new tool.

The evaluation was performed on 9 cells and consisted of extensive quality coverage during fabrication and X-ray inspection after cell assembly, activation, vibration, and simulated orbit cycling. The cycling was for a total of 100 cycles with X-ray inspection after 25, 50 and 100 cycles. A tear-down analysis was performed on all cells as a final step in the evaluation.

All radiographic inspections of the cells showed consistent and uniform tab shaping. Plates and tabs did not assume the bent or distorted configurations seen in the cells from the -13 qualification batteries. There were no indications of pressure points or bending of the plates near the tab location after the major portion of the anticipated positive plate growth had taken place. The evaluation demonstrated that the use of this production tool eliminated the deficiencies detected in the analysis of the shorted qualification battery cells. Therefore, the tool was approved for the new battery (-19) fabrication.

6/ Battery Confidence Life Test - A battery confidence life test was begun at the battery vendor facility in October 1972. Two 61B769004-13 batteries, S/N 30 and S/N 56, were subjected to this test. The prime purposes for this test program were: to increase the life test sample size from three to five; to obtain life test data at updated flight conditions; and the thereby increase the confidence in the battery and cell designs.

Both batteries passed the 4,000 cycles of simulated orbital operation with no cell problems; S/N 56 in July 1973, and S/N 30 in August 1973. A capacity check was performed on each battery after its required 4,000 cycles. S/N 56 had a capacity of 34.76 A-H compared to an acceptance test value of 40.2 A-H. S/N 30 had a capacity of 33.37 A-H compared to an acceptance test value of 41.3 A-H. The completion of the required 4,000 cycles with no cell failures increased the confidence that the improved flight batteries should be able to complete the Skylab mission. The ampere-hour values obtained after 4,000 cycles also indicated that the flight batteries should maintain a performance level in excess of the requirements throughout the Skylab mission.

The battery vendor elected to continue the life testing on these batteries beyond the 4,000 cycle requirements for engineering information. S/N 30 battery's life testing was terminated on 20 December 1973 after 5704 cycles had been accumulated. This battery experienced its first cell failure at 5427 cycles and subsequently lost four others, the last two coming on consecutive cycles at the time of termination. Test conditions were unchanged during the cell failure period.

S/N 56 battery accumulated in excess of 7250 cycles. This battery has had one cell failure at 6251 cycles. The proximity of an automatic test console malfunction which caused the battery to be overdischarged to this failure supports a cause and effect relationship. Because of this the charge voltage limits were reduced to a level comparable with a 29 cell battery and the test was continued.

B. MDAC-E Battery Testing - The purpose of the testing was to determine battery operating characteristics over the expected life of Airlock (240 days) as follows: battery performance as a function of accumulative discharge/charge cycling under mission simulated conditions; cell voltage characteristics during discharge and charge; and thermal characteristics. All batteries were tested in the Space Simulation Laboratory by the Spacecraft Electrical Systems Laboratory of the McDonnell Aircraft Company in St. Louis. Two 61B769004-3 batteries S/N 6 and 7 were tested in a 10-month period ending June 1971 and a 61B769004-15 battery, S/N 20 was tested in September 1971. Battery 61B769004-17, S/N 29, was tested for 11 months ending in March 1973. Two 61B769004-19 batteries, S/N 70 and 77 were placed on test in March and May 1973, respectively; these tests were completed in February 1974.

1/ Test Specimens - All batteries tested were production batteries. The batteries were instrumented for temperature monitoring with 20 temperature sensors mounted on the outside and within the battery. Some of the temperature sensors were mounted on top of the cells and others inserted in the internal webbing of the battery case. The -3 batteries had magnesium cases while the others had aluminum cases.

Cells used in the -15 battery contained modified geometry cell plates. The -17 battery had cells without internal hold-down blocks and its cells were x-rayed on a sampling basis to check for proper alignment of cell plates and plate tabs. The -19 batteries were similar to the -17 except all their cells were x-rayed for plate and plate-tab alignment and their plate-tabs were formed on a fixture for stress relief. The -19's are of the same configuration as those flown on the Skylab vehicle.

2/ Description of Tests - Battery testing consisted of discharge/charge cycling of the batteries to simulate the Skylab night/day orbital conditions. The DOD (depth-of-discharge) was varied from one cycle to the next. Additionally, the average DOD was varied to simulate the manned and unmanned phases and on battery S/N 70 and 77 tests the night/day times were varied to simulate the effects of anticipated Beta angle variations.

Figure 2.7-19 outlines the differences in the tests performed on the various batteries in regard to battery temperature, DOD, trickle charge current, charge return factor, maximum charging current and total number of cycles performed on the batteries. These differences were brought about by design changes in the PCG hardware and by the updating of mission requirements and conditions to those predicted at the time of the particular test.

3/ Test Results - S/N 6 and 7 batteries completed the specified test cycles with five and nine failed cells, respectively. S/N 20 battery completed the required 448 test cycles successfully. After 4011 cycles (3840 cycles required to 240-day mission), S/N 29 battery delivered 29.4 ampere-hours when discharged at 18 amperes to 30 volts. When testing of S/N 29 was terminated on 15 March 1973 at 4932 cycles to free the test setup for S/N 70 battery, S/N 29 had no failed cells and was capable of providing all anticipated mission load requirements. S/N 70 and 77 have completed 5091 and 4021 cycles, respectively without cell failure; 4000 cycle capacity discharges were 29.0 AH and 32.18 AH, respectively.

S/N	TOP-OF-CELL BATTERY TEMPERATURE	TRICKLE CHARGE CURRENT AMPS	CHARGE RETURN FACTOR	MAXIMUM CHARGE CURRENT AMPS	AVERAGE DOD		TOTAL CYCLES PERFORMED
					MANNED PHASE	UNMANNED PHASE	
6	90°F	1.5	1.220	50	11AH/2140 CYCLES	4AH/996 CYCLES	3136
7	70°F	1.5	1.150	50			3136
20	50°F	.75	1.076	50	11AH/448 CYCLES	NONE	448
29	50°F/3611 CYCLES	.75	1.080	63	10AH/2475 CYCLES	5AH/1536 CYCLES	4932 TOTAL
	60°F/1321 CYCLES	.75	1.098	63	8.5AH/654 CYCLES	3.5AH/267 CYCLES	
70	47°F	.75	1.076	42	8.5AH/3507 CYCLES	3.5AH/1584 CYCLES	5091
77	47°F	.75	1.076	42	3.5AH/2437 CYCLES	3.5AH/1584 CYCLES	4021

FIGURE 2.7-19 MDAC-E BATTERY TEST PARAMETERS

C. Vendor Battery Charger Testing - Because of the extremely rigid specifications requirements of high efficiency, reliability, functional accuracy and power density, an extensive development program was necessary to develop the battery charger circuitry.

1/ Circuitry Development - The areas which required the most extensive development were the peak power tracker circuitry, the battery temperature compensated charge voltage limit circuitry, the ampere hour meter return factor circuitry, and the power transistor drive circuitry. Development of each of these circuits required extensive breadboard testing. During the initial development phase, the battery trickle charge and charge cutoff circuitry included a mechanical relay. A review of the relay application concluded that this design was inadequate for reliable performance for an eight month mission. Therefore, the vendor was directed to replace the relay with solid state circuitry. All solid state circuitry was developed which was doubly redundant in each of the five power modules. This re-design significantly increased the reliability of the battery charger.

2/ Power Transistor Procurement - As the battery charger neared the production stages, a serious procurement problem arose. The vendor was experiencing difficulty in obtaining power transistors in sufficient quantities to maintain a timely production schedule. The cause of the problem was poor manufacturing yield of the high voltage, high power, high speed transistors which are required for the main power switching circuits in the battery charger. An evaluation program was initiated to develop an alternate transistor source. When the evaluation program indicated that an alternate transistor was satisfactory, the alternate transistors were installed in the battery charger qualification unit No. 2. This qualification unit was then subjected to the battery charger qualification tests of humidity, random vibration temperature altitude, and electromagnetic interference. It successfully passed each test.

D. Vendor Voltage Regulator Testing - The voltage regulator specification requirements of high efficiency, high power density and high reliability required significant advances in the state-of-the-art for power conditioning equipment. For this reason, the development of the voltage regulator required the vendor to perform a substantial amount of testing and design evaluation.

1/ Power Transistor Evaluation - Developmen of a high efficiercy voltage regulator required the implementation of a non-dissipative, pulse-width modulated type design. To achieve the high level of efficiency desired, the main power transisto. were required to be of the high speed switching type. In addition, the specification requirement that the voltage regulator input voltage range from 0 to 125 volts required that the power transistors be screened for high collector to emitter breakdown voltage (BV_{CEO}). A transistor which met the high switching speed, high BV_{CEO} and high power requirements had not previously been used for high reliability applications. T extensive evaluation of various transistors was performed by the vendor in order to determine a suitable transistor specification and a device which best met all of the above criteria. In June 1968, vendor specification EM 712989 was issued and a Westinghouse transistor determined to be an acceptable device.

2/ Filter Capacitor Selection - To achieve high efficiency, capacitors of extremely low series resistance were required. Thus, the vendor selected the tantalum wet slug type capacitor for the voltage regulator input and output filters. This type capacitor offered a substantial improvement in equivalent series resistance and density (microfarads per cubic inch) over other types of capacitors. However, because there had been little previous use of tantalum wet slug capacitors in high reliability applications and because some manufacturing problems had been incurred with this type capacitor, an extensive testing program was conducted to establish the performance, life stability and failure rate of these capacitors.

3/ Thermal Analysis - A thermal analysis was conducted in order to develop a voltage regulator thermal model for analysis of both steady state and transient thermal conditions. The results of this analysis are contained in Gulton Industries Report No. 2440.

4/ High Temperature - Altitude Confidence Tests - The AM voltage regulator in normal use was powered from the battery or from the output of the battery charger. In a charger bypass contingency mode, the regulator could be powered directly from the solar array. Thus, the voltage regulator was required to operate with a maximum input of 125 volts. A formal confidence test was performed by the vendor in May 1971. The test demonstrated the design adequacy of the voltage regulator under the worst case conditions of high input voltage, load, coolant temperature and low pressure. Both cycling and continuous sunlight conditions were simulated for this test. The voltage regulator successfully passed all testing in this contingency mode.

E. System Development Test

1/ Purpose - The purpose of this test was: (a) to demonstrate the designed capability of the Airlock Electrical Power System to accept simulated solar array power and to control, condition, regulate and apply this power to the AM EPS buses and the nickel-cadmium batteries, and provide bi-directional transfer of power between the AM trans buses and the ATM and between the AM transfer buses and the CSM, (b) to verify the feasibility of the design approach and provide confidence in the hardware, (c) to develop operational procedures for use in ground test and flight phases, and (d) to obtain performance and operational characteristics. The test started on 12 January 1971 and was successfully completed on 18 June 1971.

2/ Test Article - The test article was one production battery module and one Laboratory EPS Simulator. The battery module included one-half of the Airlock Power Conditioning System (4 PCG's, Group #5 through #8). The Laboratory EPS Simulator included all of the wiring, controls, indicators, buses and interface connectors to essentially duplicate the Airlock two bus power control and distribution system plus necessary DCS and TM Simulation. A Solar Array Wing Simulator (SAWS) capable of simulating the instantaneous and time varying current-voltage (I-V) output characteristics of the OWS solar array was used as an input power source for the PCG'S. A CSM and an ATM simulator simulating their respective power and distribution systems were also used.

3/ Test Phases - Testing was performed in three phases as follows. Phase one tests were pre-installation acceptance tests at ambient conditions of all flight configuration equipment used in the test article. The laboratory EPS simulator was tested to verify that all wiring, switching and indicating functions performed in accordance with applicable system design requirements. Critical circuit resistances were measured and recorded. Phase two testing, conducted at ambient conditions, consisted of testing each PCG as a subsystem followed by testing in steps of 2, 3, and 4, PCG'S operating in parallel. This phase of testing verified each individual PCG performance and parallel PCG performance. AM EPS performance in parallel with other cluster sources was also verified during this phase of testing by utilizing ATM and CSM simulators. One coolant loop was operated throughout phase two testing. The coolant inlet temperature was maintained at 65°F unless otherwise required. The coolant flow rate was maintained at 115 lbs/hr split equally between two battery coldplates. All switches and controls were exercised and functioned properly.

The maximum load capability of each individual PCG was determined for five orbital cases. The results compared favorably with predicted values as shown in Figure 2.7-20.

The maximum load capability of 4 PCG's in parallel connected to one regulated bus was determined under equal regulator load sharing

MAXIMUM LOAD CAPABILITY OF INDIVIDUAL PCG'S							
COOLANT INLET TEMP	BETA ANGLE	SIMULATED ATTITUDE	REGULATED BUS LOAD IN WATTS				
			PCG #5	PCG #6	PCG #7	PCG #8	PREDICTED
65°F	0°	INERTIAL	544	563	540	540	530
65°F	58.5°	INERTIAL	890	930	900	903	850
65°F	73.5°	INERTIAL	1354	1415	1423	1388	1500
65°F	0°	ZLV	325	375	320	300	300
65°F	73.5°	ZLV	90	95	85	80	40
36°F	0°	INERTIAL	538	538	537	530	533

MAXIMUM LOAD CAPABILITY OF 4 PCG'S IN PARALLEL				
COOLANT INLET TEMP	BETA ANGLE	SIMULATED ATTITUDE	MEASURED POWER (WATTS)	PREDICTED POWER (WATTS)
65°F	0°	INERTIAL	2150	2120
65°F	0°	ZLV	1300	1200

FIGURE 2.7-20 MAXIMUM LOAD CAPABILITIES OF PCG'S

conditions for two orbital cases. The results compared favorably with the predicted values as shown in Figure 2.7-20. It was concluded from this test and was confirmed from individual PCG testing that the maximum load capability of a PCG depends on the battery charging characteristics, i.e., the higher the battery charge current at constant potential near the end of the orbital daylight period, the higher the maximum load capability of the PCG.

The failure of a voltage regulator resulting in a bus overvoltage condition was simulated to verify proper operation of the shunt regulator. This test was performed at no bus load and various regulator input voltage levels. The shunt regulator cleared the simulated voltage regulator failure and maintained the bus voltage at acceptable levels in all cases.

Phase three testing was a thermal vacuum test simulating electrical and temperature altitude flight conditions. The EPS was tested at the simulated orbital conditions of $\beta = 0°$, $\beta = 58.5°$ and $\beta = 73.5°$ in the solar inertial attitude. Coolant loop temperatures at the battery module coolant inlet were held constant at the temperature levels predicted for each test condition. The Reg bus load was a typical mission day load at maximum load capability. That is, the typical day load profile was adjusted u[illegible]d such that the average load over 15 orbits was equal to the [illegible]lculated maximum load capability. The result of testing at the three orbital conditions showed that the AM EPS would meet the mission requirements under normal operating conditions. Contingency operations with simulated critical EPS failures such as with a failed battery charger or a failed voltage regulator were verified utilizing developed contingency operational procedures. The AM EPS management and control system was effectively utilized to isolate the failed component, to minimize the effect of the failure on mission objectives and to identify operational constraints.

4/ Anomalies - After approximately 23 hours into the $\beta = 73.5°$ all sun test, the load on the bus was increased from 1850 watts to 3000 watts. Ten minutes after the load was increased, the output voltage level of PCG #7 regulator decreased from 28.795 volts to 28.560 volts, indicating possible failure in power module #3 in the voltage regulator. (The suspected failure mode was later verified during failure analysis). This condition did not jeopardize subsequent testing. Whenever this condition occurred again during subsequent testing, the voltage decrease in the regulator output was compensated for by adjusting the regulator fine adjust potentiometer during the orbital dark period such that all batteries were discharging at approximately the same current rate. This workaround procedure was a designed contingency mode and would have been used in flight had the need arisen.

5/ Conclusions - The Airlock Electrical Power System demonstrated by these tests its designed capability to accept simulated solar array power and to control, condition, regulate and apply this power to the AM buses and the nickel-cadmium batteries. Paralleling of PCG's was successfully accomplished using appropriate EPS controls. The test results showed that under normal operating conditions, the maximum average output power was obtained when all PCG's shared the load equally. Load sharing among parallel PCG's was successfully obtained by adjusting the regulator fine adjust potentiometers during the orbital dark period such that all batteries were discharging at the same current rate. Bi-directional transfer of power between the AM transfer buses and the ATM and between the AM transfer buses and the CSM was successfully demonstrated. Results of testing at ambient conditions and then under simulated flight environments, including orbital attitude, demonstrated the AM EPS capability to meet mission requirements. Ths AM EPS management and control system was effective in isolation of major failed components such as regulators or chargers and at the same time provided the maximum continuous power possible, thus minimizing the effect of the "failure" on total mission objectives. In the case of a failed charger, testing demonstrated that with the solar array connected directly to the regulator input and with the regulator fine adjust pot adjusted to the maximum clockwise position the PCG in this contingency mode would reduce the load imposed on the remaining PCG's during the orbital daylight period and result in a significantly greater system load capability than would be possible with remaining PCG's operating alone. If the battery had failed but the charger was functional, the battery circuit should be opened and the charger left in the circuit. This would result in greater utilization of array energy than the case where the array feeds the regulator directly. The regulator fine adjust pot should again be adjusted toward the high voltage limit, so as to assume a greater output power level. The amount of the pot adjustment would be determined by real time observations.

The effects of an inadvertent reset of the A-H meter to zero were determined by test. The test showed that the coolant system could not limit battery temperature but did reduce the temperature rise with respect to time, providing adequate time for monitor and shut down before the temperature sensors in the battery reached thermal cutoff. There was no evidence that the cooling system could limit battery temperature to a level below thermal cutoff if the battery was allowed to charge in the TEMP LMTD mode for an extended period of time with the battery at 100 percent SOC. Therefore, if the battery TEMP LMTD mode was initiated, the battery temperature would require continuous monitoring.

In summary, the test results showed that the AM EPS would meet or exceed mission requirements under normal operating conditions. With the flexibility built into the EPS management and control system it could be effectively utilized under contingency conditions to maintain the highest output power level possible using verified contingency operational procedures.

F. Integrated SAG/PCG Testing

1/ Purpose - The purpose of the testing was to verify the compatibility between an individual Solar Array Group (SAG) and an individual Power Conditioning Group (PCG). The testing was conducted under the technical direction of the Martin Marietta Corporation at their Sunlight Test Facility in Denver, Colorado. The testing took place between 11 September and 7 October 1971.

2/ Test Set-up - The test set-up closely represented one flight configured SAG/PCG. The SAG consisted of 30 solar cell modules which were properly connected through a power unit to the input of the PCG. The PCG consisted of an Airlock battery charger, battery, and voltage regulator mounted on a coldplate for thermal control. Connected at the regulator output was a set of loads so that expected flight load profiles could be simulated. An automated I-V curve instrument was used to determine the SAG peak point during the peak power tracker test.

3/ Test Results - The testing consisted of the Solar Inertial Orbit Test, Z-LV Orbit Test, Z-LV Beta Angle Test, Peak Power Tracker Test and Transients Test. Throughout all testing the SAG proved to be very compatible with the PCG. The battery charger peak power tracker properly extracted peak power from the SAG upon demand and the battery charger properly charged the battery or shared the load with the battery when the load demanded it. The voltage regulator properly regulated bus voltage. Transients induced by the various subsystem switching functions resulted in no apparent effects on SAG/PCG operations. In all, no problems were discovered during the Integrated SAG/PCG Testing.

G. SCR Circuit Development Testing

1/ Purpose - Electrical testing was performed to support the anomaly investigations and also to provide confidence in the operation of the final design solution. See paragraph 2.7.3.4D for a discussion of the anomaly. Specific test objectives were:

- To verify the analytical conclusions on the causes of the problem.
- To verify the basic concept of the proposed solution.
- To determine stress levels for the SCR.
- To accumulate "worst case" operational cycles on a flight equivalent relay circuit to develop a high degree of confidence in the selected SCR and the total circuit.

2/ Testing - The analytical conclusions were verified by using capacitors to simulate PCG equipment filters and by simulating the Charger and Battery relay operation in connecting the unequally charged capacitors together. This set-up was also used to verify the basic concept of protecting the relay contacts from current surges at the instant of contact "make" by shunting the current at this time through a parallel connected SCR. The observed performance was satisfactory in all respects. It was also discovered during this testing that the peak current surge through the SCR was significantly less than the current surge which would be expected through the unprotected relay contacts. The cause was the additional wiring resistance included in the SCR circuit. This was important since it reduced the stress level requirement on the SCR's.

The testing of the final SCR design in an actual PCG was implemented by using the PCG equipment available from the previously run Integrated SAG/PCG test. This equipment consisted of one flight configuration PCG including wiring and controls. The SCR circuits were assembled as a breadboard and integrated with the PCG. A JAN-2N2030 SCR (commercial equivalent of the flight 61C769018-1 units) was selected to meet the circuit requirements for the SCR application. The wiring resistances in the SCR circuits were designed to be equivalent to the anticipated flight equipment installation. The operation of each SCR circuit was observed under all combinations of circuit conditions which would exercise their associated relay contacts. Also all relay closing sequences were simulated by controlling relay closure times with operation again being observed.

3/ Conclusions - The testing demonstrated the ability of the SCR circuits to successfully protect the Charger and Battery relays. Current surges caused by the charging and discharging of the voltage regulator and battery charger capacitors were conducted through the appropriate SCR during the closure of the associated power relay. The JAN-2N2030 SCR operated satisfactorily throughout the testing. There were no SCR failures and no evidence of any SCR degradation was observed during the testing. No SCR failed to fire when properly triggered and there were no inadvertent, i.e., without a trigger pulse, firing of any SCR.

A confidence test of more than 500 cyclic operations was performed at the circuit conditions causing worst case transient current. All aspects of operation were successful throughout this test which indicated that the SCR and SCR circuits could meet any possible flight requirements.

H. Cluster EPS Testing

1/ Purpose - The overall purpose of this testing was to verify proper operation of the Cluster EPS systems in their parallel modes of operation prior to the mating of the actual flight systems. To accomplish this purpose a Skylab Cluster Power Simulator (SCPS) was developed at the Marshall Space Flight Center (MSFC). The specific primary objectives of the testing on the SCPS were as follows:

- To demonstrate the capability of the AM and ATM power systems to operate in parallel and to verify stable operation of the two systems when subjected to the flight power profile.
- To demonstrate that flight circuit wiring is adequate for proper load sharing by the power systems.
- To analyze the effects of the single-point-ground system concept with the cluster power systems in its various configurations.
- To demonstrate and analyze power system failures and contingency modes of operation.
- To determine short-term and long-term effects of simulated orbital operation on the systems and particularly on their batteries.

2/ SCPS Description - The SCPS hardware consisted of both flight systems hardware and electrical support equipment (ESE) hardware. The flight (or flight equivalent) hardware included: two AM battery modules (8 PCG'S); 18 ATM Charger Battery Regulator Modules (CBRM'S); an ATM Power Transfer Distributor; an AM power distribution system; and three AM control, display, and circuit breaker panels. The ESE equipment included: ATM solar array simulators; OWS solar array simulators; cluster load banks; a CSM source and load simulator; network control and switching equipment; a digital data acquisition system; a low temperature test unit; an airconditioning system; and various ESE control and display panels. All power distribution interconnecting cabling was made equivalent to flight wiring.

3/ Testing - Testing on the SCPS commenced in February 1972. Tests were performed in compliance with the "Skylab Cluster Power System Breadboard Test Requirements" Document, 40M35693. The SCPS was also used as a training aid during training classes on the cluster EPS for flight control and astronaut personnel.

4/ Test Results- Testing on the SCPS was very successful. Testing was initiated early enough so that any problems could have been solved without affecting launch schedules. However, no problems were encountered with the parallel operations of the cluster EPS systems. The testing verified the compatibility of the AM and ATM power systems and their capability to interface with the simulated CSM power system. Flight procedures associated with the EPS systems were verified on the SCPS. Contingency procedures to overcome simulated system malfunctions were also verified by SCPS testing.

5/ Mission Support Status - The SCPS was maintained in an up-to-date flight system equivalent status for the Skylab mission phases to allow for system support testing. Several such tests have been performed during the mission. They include the SL-1/SL-2 Battery Storage Test; the SL-3 AM EPS Shutdown Procedures Test; and the SL-3 SAS #4 Current Anomaly Test. These tests are discussed in Section 2.7.4.8 of this report.

2.7.3.3 Spacecraft System Testing

The Spacecraft System Test (SST) was performed on the flight Airlock Module at MDAC-E from October 1971 through September 1972. The progression of the Electrical Power System through the SST is shown by the flow diagram on Figure 2.7-21. This diagram shows the major tests, changes, and retests which verified that the EPS design met all requirements and that the EPS hardware was satisfactory for delivery to the customer for flight usage.

The procedures for the SST and the test results are given in the various Service Engineering Data Reports (SEDR's) as referenced on the flow diagram. Dynamic resistance measurements were performed on the end-to-end PCG and feeder/distribution wiring in SEDR D3-N14-1 to verify the wiring design prior to powered operations. Pre-power bus isolation checks [illegible] the initial bus power application were performed in SEDR D3-[illegible] and included:

- Bus isolation checks.

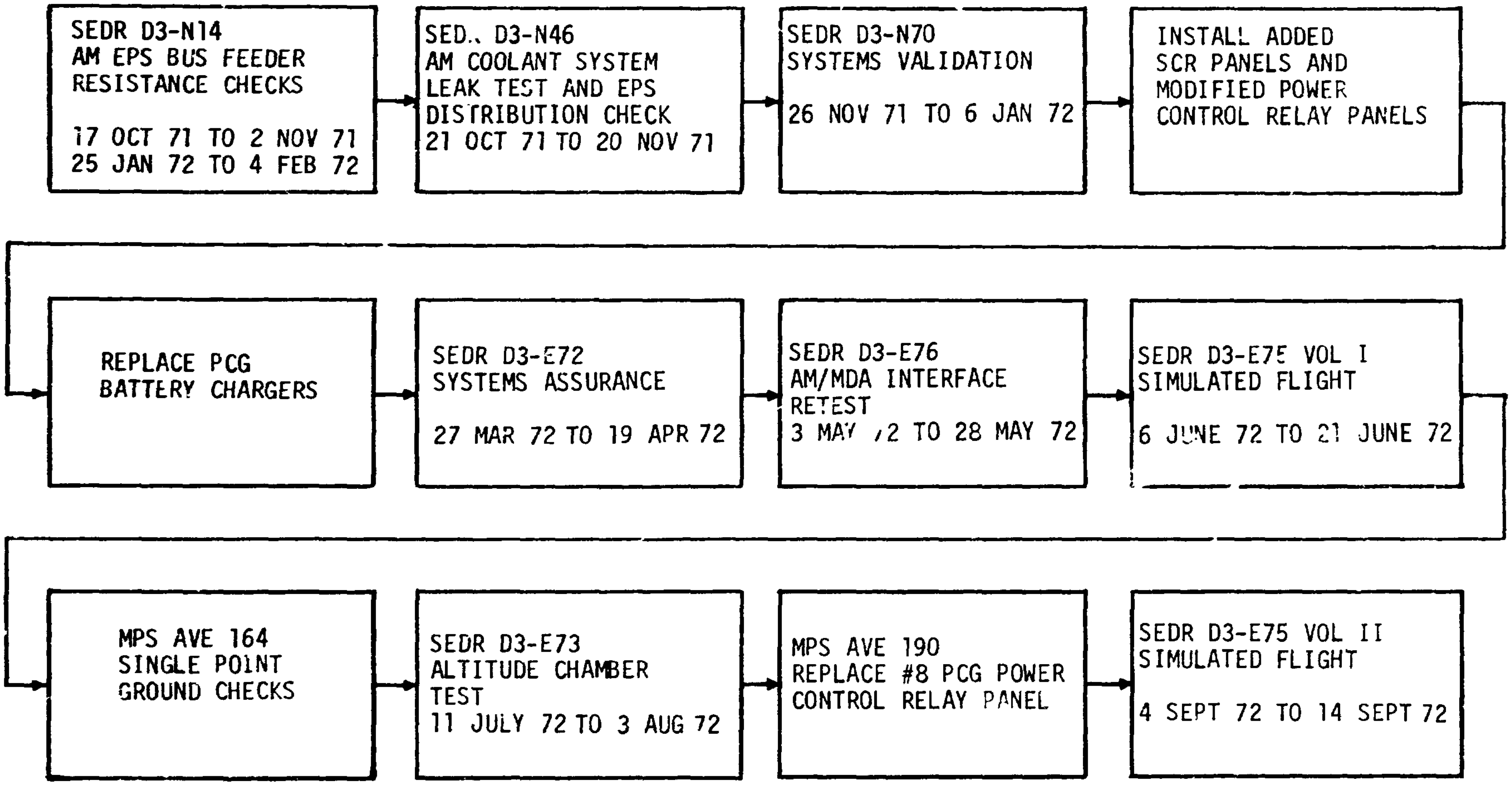


FIGURE 2.7-21 ELECTRICAL POWER SYSTEM – SST FLOW DIAGRAM

- Bus power application and distribution.
- System status light verification (excluding PCG) and light test circuit checks.
- Power distribution and polarity checks to ECS and coolant components prior to component hookup.
- Fan and pump inverter output power phase-to-phase checks prior to component hookup.

The systems validation test, SEDR D3-N70-1, accomplished an overall check of the Airlock power system and monitored and applied signals at the AM/OWS, AM/MDA, AM/ATM and DA/PS interfaces. Included were the following checks:

- Individual PCG end-to-end voltage distribution and polarity varification.
- Individual PCG operation in all modes (manual & command).
- Individual PCG and combined bus voltage droop measurements.
- Determination of voltage trigger points for EPS C&W parameters, emergency lighting and bus shunt regulators.
- Emergency power disconnection.
- Maximum interface power transfer per the associated ICD levels.
- EPS telemetry and on-board parameter correlation.

During the systems validation test, several PCG power control relays were damaged by large transient currents. After SEDR D3-N70-1 silicon controlled rectifier (SCR) panels were incorporated to protect the relay contacts from these current transients. Also, at this time all PCG battery chargers were replaced with units which had been upgraded to include a reduced battery current trickle charge rate and a reduced return factor. During the systems assurance test, SEDR D3-E72-1, the systems validation checks on the PCG's were repeated due to the extent of the PCG modifications. The following checks were also performed:

- Verification of the new battery current trickle charge rate.
- Verification of operation of the SCR protective circuits.
- Verification of maximum power transfer across AM/MDA and MDA/CSM interfaces in accordance with the respective interface specifications.

- Verification of proper load current sharing on power feeders and distribution networks.
- Emergency power down circuit verification for on-board and GSE initiation.
- Voltage polarity and distribution to MDA components prior to component connection.
- Correlation of telemetry, C&W and on-board display parameters.

During AM/MDA interface retest, SEDR D3-E76-1, the correct mechanical and electrical operation of the modified Reg bus adjust potentiometer installation was verified. The shunt regulators were replaced by upgraded units and measurement of the primary trigger voltage level was performed.

During simulated flight test, SEDR D3-E75-1 Volume I, a simulated flight mission profile of the AM/MDA was performed to support EMC critical circuit measurements and to demonstrate an all system EMC compatibility. The AM EPS batteries supplied all AM electrical bus power requirements from simulated lift off through the completion of the simulated OWS solar array wing deployment sequence. PCG's were operated in their cyclic mode from simulated solar array wing deployment to the termination of the test which occurred after operation in the orbital storage configuration. Sequential and deploy bus activation, ATM/AM EPS paralleling, CSM ground switchovers, SWS/CSM EPS paralleling and unparalleling were all performed during the cyclic operation. The isolation between the single point ground and the Airlock power returns on the AM/MDA were continuously monitored, per the Mission Preparation Sheet (MPS) for Aerospace Vehicle Equipment (AVE) 164, during GSE cabling and equipment installation, and whenever the AM/MDA was not being powered. This MPS was initiated prior to the altitude chamber test, SEDR D3-E73-1, and continued until shipment to KSC.

The altitude chamber test was performed in three (3) phases; the unmanned altitude phase, the manned ambient phase, and the manned altitude phase. Prior to initially operating the AM/MDA in a low pressure environment the ability to remove all electrical power from the AM busses under simulated emergency conditions was verified. During the unmanned run the power system was operated to a simulated flight profile with simulated electrical loads as required. From liftoff to completion of the solar array deploy sequence, AM flight type batteries supplied all AM bus loads. The PCG's were operated in a cyclic mode for the remainder of the unmanned run. Single point ground transfer to the CSM simulator was accomplished. ATM and CSM simulator power systems were paralleled with the AM EPS. All CSM simulator loads were transferred to AM power. The EPS remained operative for an additional 84 hours in support of the outgassing test. The manned ambient run was an evaluation of the procedures to be used in the manned altitude run. Prior to ascent to simulated altitude during the manned run, the capability to remove all electrical power from the Airlock was reverified. The manned runs began with the EPS operating in the normal orbital mode with the AM and ATM paralleled and CSM simulator receiving power from the AM busses. The crew evaluated the responses of the EPS to manual system controls. The CSM simulator EPS was reactivated and isolated from the AM EPS. Single point ground was restored in the AM by the crew. The objectives of EPS verification at simulated altitude conditions were satisfied by this testing.

A ground fault detected during SEDR D3-F90-1 Volume IV, was isolated to PCG #8 control relay panel. After replacement of the panel all functions in the new panel were verified per MPS AVE 190 with satisfactory results.

Simulated flight tests, SEDR D3-E75-1 Volume II, demonstrated both the operational and the electromagnetic compatibility between the AM/MDA supporting systems and the earth resources experiment package (EREP). In conjunction with the above demonstrations, AM/MDA critical circuit measurements were performed with satisfactory results. After the removal of the ground support equipment used to measure critical AM/MDA circuits and the reconnection of the AM/MDA flight circuits, the EREP simulated flight was repeated to assure flight integrity of these circuits. This was done with the EPS configured in the orbital mode and with nominal EREP orbital loads applied.

During SST a joint NASA/MDAC-E Review Team reviewed all testing; evaluated the significance of problems as they occurred; and evaluated the suitability of solutions. This joint cooperation allowed for quick classification of problems and the timely implementation of acceptable solutions.

SST testing on the Airlock Module EPS was very successful. In addition to passing the specific EPS tests, the power system successfully performed its normal function of supplying AM electrical power during many of the tests on the other AM systems. The problems that were uncovered during the testing were thoroughly investigated and their resultant solutions were expedited to minimize the retest effects on the overall test program. The Spacecraft System test program, therefore, met its objectives of verifying that the EPS design met its requirements and that the U-1 hardware was satisfactory for flight.

2.7.3.4 Design Modifications

A. Battery - The design selected for the Airlock batteries was one which had a successful flight history. At the outset of the Airlock program no battery hardware development was anticipated. However, it became necessary to modify this design in order to resolve problems which were encountered in the course of the Airlock program. In all, three design configuration changes were incorporated.

1/ Case Material Change and Cell Plate Shape Modification - The first design configuration change was twofold. First, three shorted cells were detected during early cell conditioning activity. Vendor analysis of these cells revealed inadequate clearance existed between the top of one plate and the tab attachment to the adjacent plate. This condition would allow little pack misalignment during assembly and represented a potential life limiting problem in the event of plate shifting after cell assembly. Second, newly manufactured batteries indicated during acceptance testing that operating temperature gradients within the battery case were greater than anticipated. This condition also represented a detriment to attaining the necessary cycle life. To correct these conditions, a new design configuration was established in March 1971. The new configuration incorporated a change of the battery case material from magnesium to aluminum in order to significantly reduce the temperature gradients and a modification of the cell plate geometric shape in order to provide greater clearance between the top of one plate and the tab attachment to the adjacent plate.

2/ Elimination of Cell Holddown - The second configuration change followed the first by four months. In the intervening time, two of the original batteries were undergoing an engineering test when numerous cells shorted. MDAC-E analysis of the failed cells disclosed not only the tab clearance problem but a positive plate dimension growth condition. This plate growth was restricted by the presence of nylon pack holddowns within the cells. Plate deformation and interplate pressure points developed causing premature cell failure. Tests of cells without holddowns were conducted to determine their susceptability to vibration exposures at and above Skylab levels. Results indicated the pack constraint was not necessary for the Airlock application and a new design configuration, withouth the holddown, was established. Provisions for monitoring individual cell voltages were also incorporated as part of this new configuration.

3/ Pack Assembly Process Control - The last design configuration change was initiated in September 1972. One of the two qualification batteries had successfully completed a demonstration of the 4000 cycle life requirement while the other failed after 3028 cycles. Analysis of cells from both units revealed no inherent design problems. However, evidence did indicate that workmanship quality with respect to pack assembly and tab shaping was not consistent. Because of this deficiency the quality of batteries previously assembled was suspect and could not be verified. A limited make of new batteries for flight was undertaken whereby heavy emphasis was placed on quality control of the assembly operation. Utilization of a special tab combing tool which afforded exacting control of the critical pack assembly process was the major improvement for this new configuration.

B. Voltage Regulator

1/ Elimination of Power Transistor Oscillation - During the production phase of the voltage regulator program the vendor encountered a problem in the manufacture of voltage regulators. During testing at regulator input voltages greater than 90 volts, a high frequency oscillation of the power transistors was occasionally observed. If this oscillation was allowed to continue, extensive heating and eventual destruction of the power transistors would occur. The oscillation problem was eliminated with a design modification which added a small inductance in the base lead of each power transistor.

2/ Bias Converter Modification - During acceptance testing a total of three complete regulator failures occurred on two different units. The failures were complete in that all power modules were found to be inoperative following the failures. The failure analyses isolated the problem to the two redundant ± 12 volt bias converters contained within the voltage regulator. Further analyses revealed a marginal design where component tolerances combined with high regulator input voltages caused the complete failure of the voltage regulator. The corrective action required that the design be modified to change the values of three pertinent components in each bias converter circuit. In addition, all units not containing test thermocouples were modified to include test points for verification of individual bias converter operation. The flight configuration designation was changed from dash number -3 to dash number -9.

C. Battery Charger

1/ Redundant Ampere-Hour Meter Addition - The original EPS design contained one ampere-hour meter per battery charger. Airlock reliability studies concluded that the EPS reliability would be significantly increased with the addition of a redundant ampere-hour meter to each battery charger. To allow for the packaging of a second ampere-hour meter a mechanical redesign of the battery charger was required. The complete redesign was incorporated prior to the battery charger qualification program or the assembly of any production units.

2/ <u>Ampere-hour Meter Usage Modifications</u> - The redundant, secondary ampere-hour meter was originally specified as an unpowered, stand-by circuit which would only be activated with a failure of the primary ampere-hour meter. The ampere-hour meter state-of-charge (SOC) readout was intended only for ground analysis. The NASA specified that three modifications to this original design be incorporated. One was that both the primary and secondary ampere-hour meters be powered simultaneously. A second was that on-board meter display capability be provided for both ampere-hour meters. The third was that the charge mode switching function be provided so that in the event that the controlling ampere-hour meter progressed to 100 percent SOC in advance of the "true" battery SOC, the capability for remaining in the voltage limited charge mode at 100 percent SOC would exist. The above modifications were incorporated prior to the battery charger qualification program or the assembly of any production units.

3/ <u>Array Voltage Increase</u> - The original solar array design exhibited a maximum voltage of 110 volts. The battery charger was originally designed to meet this input voltage interface requirement. However, the solar array specification was subsequently changed to allow a maximum array voltage of 125 volts. This higher input voltage interface required that the voltage rating of the power transistors and input capacitors in the battery charger be increased. These design modifications and the selection of higher voltage components had a significant impact on the development of the battery charger design.

4/ <u>Peak Power Tracker Circuit Modification</u> - During vendor development testing, prior to the battery charger qualification program, a problem was noted with the peak power tracker operation. Under some combinations of load and array conditions, a load transient would occasionally cause a temporary collapse of the solar array voltage. The problem was analyzed to be contained within the closed loop response of the peak power tracker circuit. A design modification which eliminated the problem was incorporated in the peak power tracker circuit.

5/ Bias Converter Modification - Just prior to the start of the battery charger qualification program a failure of the bias converter circuit occurred during vendor acceptance testing. Failure analysis revealed that the main power transistor in the bias converter circuit was overstressed during initial turn-on of the battery charger at high input voltage. A change of power transistors and a design modification of the associated power transistor drive circuitry was incorporated in order to correct the problem.

6/ Mositure Seal Addition - The battery charger initially failed to pass the humidity qualification test. After a mositure seal was added to all mating surfaces of the removable battery charger cover plates, the charger passed the test. The seal was incorporated in all units.

7/ Ampere-Hour Meter Reset Circuit Modification - During the battery charger temperature-altitude qualification test the ampere-hour meter reset circuit was observed to be susceptible to noise generated by the testing equipment. The problem was corrected with the addition of a filter capacitor in the reset circuit and all prior testing was successfully repeated. This modification was incorporated in all units.

8/ Trickle Charge Current and Return Factor Reduction - In mid-1971 NASA directed that the trickle charge current be reduced from 1.5 $\pm$ 0.5 amperes to 0.75 $\pm$ 0.5 amperes and that the return factor be reduced approximately 4 percent for the expected battery operating temperature range. The trickle charge current reduction required the replacement of six resistors per power module or a total of thirty resistors per battery charger. The return factor reduction was accomplished with the replacement of six resistors per ampere-hour meter or a total of twelve resistors per battery charger. Both design modifications were incorporated in all units. The battery charger qualification status was not affected. The flight configuration designation was changed from dash number -5 to dash number -11.

9/ Operational Amplifier Protection - A redundant pair of operational amplifiers in the battery charger voltage control circuitry failed in four units. In each case, the failure occurred while the battery charger was in an unpowered condition; that is, during troubleshooting, dielectric testing, etc. Failure analysis disclosed that the input junctions to the operational amplifiers were damaged by excessive input current. The analysis did not pinpoint the exact cause of failure. However, the addition of current limiting resistors in series with the operational amplifier input leads was projected to be a necessary precaution in order to preclude the reoccurrence of this type of failure. All units were modified to incorporate the current limiting resistors. The battery charger qualification status was not affected. The flight configuration designation was changed from dash number -11 to dash number -17. No further problem of this type occurred.

D. SCR Circuit - During spacecraft systems testing at MDAC-E, several failures of a charger relay (Normal/Bypass positions) were encountered during switching operations. The physical problem was determined to be the welding of the relay contacts in their Normal position. The test conditions on the charger relays and associated circuits during the time interval surrounding the failures were analyzed to determine the cause of the failures. The analysis revealed that the cause was a severe current transient through the relay contact when the relay was switched from the Bypass to the Normal position. This current transient was being produced by a large voltage differential between the voltage regulator input filter capacitors and the battery charger output filter capacitors. The charger relay contact closure to the Normal position completed a circuit from the voltage regulator input filter capacitors to the battery charger output filter capacitors. The contact closure resulted in a transient current of approximately 1400 amperes flowing through the relay contacts for several hundred microseconds.

A review of other PCG circuits indicated that a similar problem would exist with the battery relay. This would occur if the battery relay (Off/On positions) was positioned to the On position at a time when the battery charger filter capacitors were not charged. A subsequent test revealed that this operation did cause current transients of approximately 400 amperes to flow for up to one millisecond duration. Repeated operations on one battery relay also caused these relay contacts to weld together in the ON position after several operations. Several designs to solve these problems were investigated. The best solution proved to be the addition of SCR (silicon controlled rectifier) circuits across each battery and charger relay contact. The SCR circuits shunted the high current transient around the relay contact thereby eliminating any damage to the contact. The SCR was triggered ON prior to relay contact closure and was turned OFF by being shorted out by the relay contact closure. The SCR's were also fused for fault isolation purposes.

2.7.3.5 Deviations

The Electrical Power System was granted one deviation. MDAC-E-1 vs ICD 40M35659-3, Definition of ATM/AM Electrical Interface.

2.7.3.5.1 Description

The 12-gage power feeder wires between the AM OV bus and the ATM bus was designed in accordance with standard electrical practice for all other power feeders, using nontwisted wires. Later submittal of the interface document ICD 40M35659-3 to MDAC-E included a description of these feeders as being twisted pairs. A request submittal to change the ICD to conform to the existing design was rejected, therefore a deviation was requested and approved.

2.7.3.5.2 Justification

The cost and schedule impact of modifying the existing design was not considered to be warranted by the theoretical benefits of twisted wires.

2.7.4 Mission Results

2.7.4.1 Electrical Power System Performance

The Airlock Module electrical power system performed all of its required functions and operations during the Skylab mission. Because of the mechanical problems encountered with the deployment of the OWS solar array wings, evaluation of the AM EPS performance was separated into two time periods. The first period consisted of the time frame from SL-1 launch through DOY 159. During this period the AM EPS operated to utilize essentially all of the available solar power, even though significant mission support was not possible until wing deployment. The second period started with the successful deployment of OWS solar array wing #1 on DOY 159 and continued through the end of the Skylab mission. The AM EPS was fully operational during this second time period and provided an average of 46% of the total cluster power required despite the absence of one-half of the expected OWS solar array power.

A. SL-1 Launch Through OWS SAS Deployment - OWS Solar Array Wing #2 was completely lost during the SL-1 launch. Solar array Wing #1, which was only partially deployed, could not supply sufficient power to the EPS during this period to allow the EPS to supply any significant amount of power to the cluster load buses. The available solar array power was used, however, to charge some of the batteries at low charge rates until they reached 100% SOC. At all times, the stored energy of the AM batteries was available to supplement the ATM source when and if required. Other EPS equipments, such as battery chargers and voltage regulators, also operated under abnormal conditions during this period. All EPS equipments operated acceptably under the abnormal conditions they encountered and subsequently exhibited normal performance characteristics when higher input power levels were achieved. The flexibilities of the AM EPS control and distribution system were used extensively during this period to manage the AM EPS in the most optimum manner.

B. OWS SAS Deployment Through End of SL-2 - AM EPS activation took place on DOY 159 with the full deployment of all three wing sections of solar array wing #1 at 0020 GMT. All AM batteries were fully charged after only a few orbits and the system was returned to stabilized cyclic operation by DOY 160 The AM EPS performed up to expectations throughout the remainder of the SL-2 manned mission phase without problems. Therefore, there were no control switching operations required. The Reg Bus pots were adjusted several times, however, to optimize load sharing between the AM EPS and the ATM EPS based on their respective capabilities and on load requirements. These adjustments were readily accomplished by the crew with minimum impact on crew time and with results that closely matched flight control predictions. Load sharing among the 8 PCG's was very nearly equal and remained very stable throughout the mission. This reflected the accuracy of the pot setting procedures utilized during prelaunch checkout and the stability of the voltage regulators and the potentiometer sensing circuits during the mission. The fine adjustment potentiometers, therefore, were not repositioned any time during the SL-1/SL-2 mission.

Only one abnormal condition associated with the AM EPS was encountered during this mission phase. The SAS #4 current monitors, onboard and telemetry, indicated a SAS #4 current consistently lower than the other SAS currents. See Paragraph 2.7.4.7 for a detailed discussion of this problem.

The AM EPS continuous bus power capability, based on solar array wing #1 only, varied as shown on Figure 2.7-22. The solar inertial capability started at approximately 3000 watts on DOY 159 at β = -12° and increased to approximately 4900 watts on DOY 172 when the all-sun condition was reached at β = -69.5°. The daily average power usage during solar inertial attitude was as shown by the AM EPS load curve on the figure. The orbit average load varied from 1600 watts to 3500 watts, and the battery DOD's varied from 0 to 16%. Operation at the full AM EPS power capability was not required during this mission phase. Figure 2.7-22 also shows the AM EPS power capability for various EREP orbits based on a 50% battery DOD constraint. The maximum DOD for any actual EREP orbit was 41% for EREP #11 on DOY 165.

C. SL-2 to SL-3 Storage Period - The AM EPS performed satisfactorily with no problems throughout this mission phase. No commands were sent to the system during this period, therefore, the system remained in a baseline configuration.

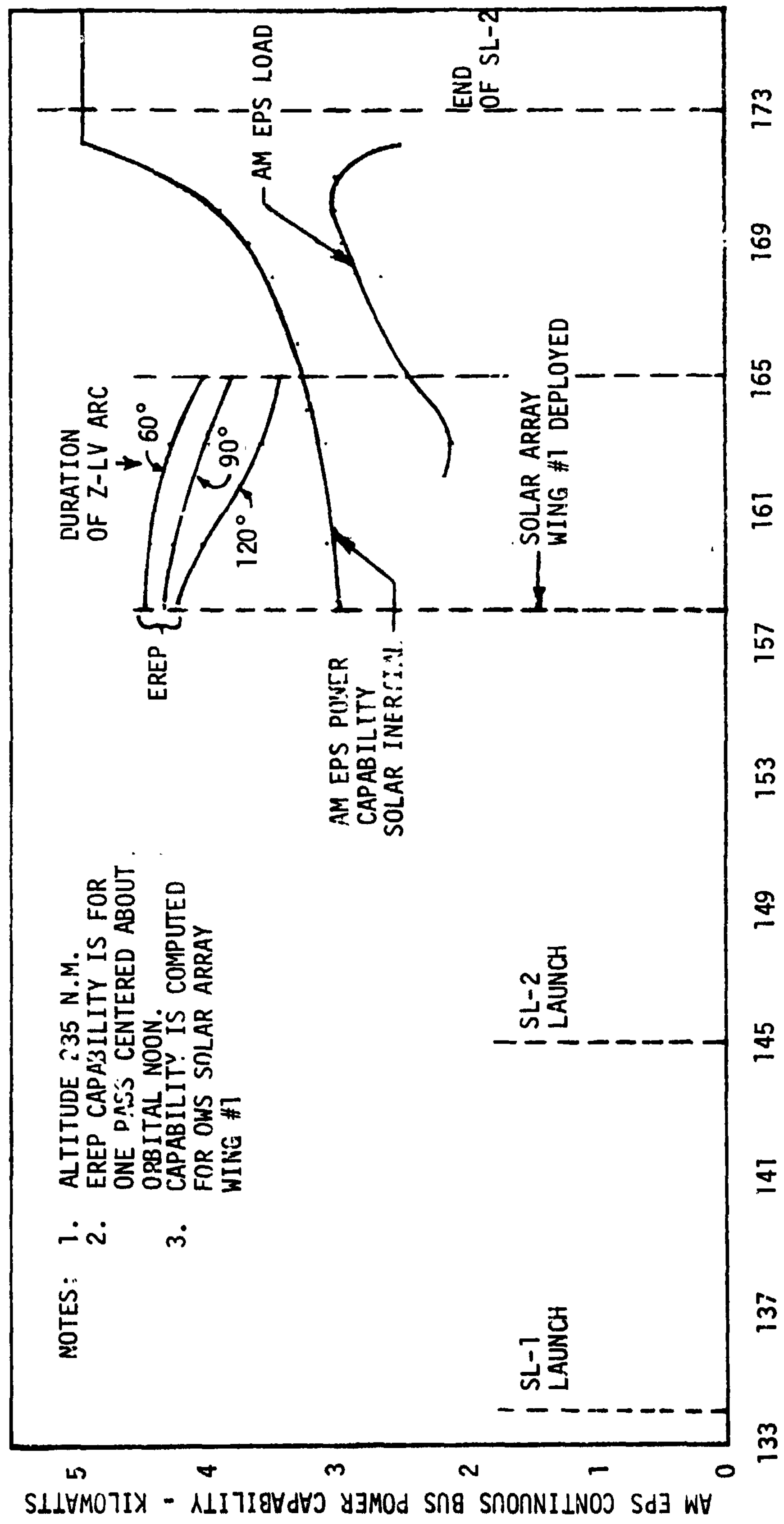


FIGURE 2.7-22 CALCULATED AM EPS BUS POWER CAPABILITY VERSUS DAY-OF-YEAR

Figure 2.7-23 shows the calculated AM EPS power capability over this period. The high plateau at the beginning corresponds to the 3 day period where the magnitude of the β angle continued to exceed 69.5°. The capability then decreases as the magnitude of the β angle decreases. From DOY 185 and on, the β angle remained in the -30° to +30° range and the capability remained at approximately 3,000 watts. The AM EPS load curve on Figure 2.7-23 shows that the actual load was well below the capability at all times. This is also reflected in the battery DOD's which averaged 9 to 10% during this mission phase.

D. SL-3 Manned Phase - The AM EPS continued to perform well throughout this mission phase and no failures of EPS equipments were noted. All required operations involved with activation, deactivation, paralleling, and EREP periods were successfully completed. Reg bus potentiometer adjustments were made a number of times to control AM/ATM EPS load sharing to the desirable levels. Several voltage regulator Fine Adjustment potentiometers were adjusted for the first time during SL-3. This was done to optimize the AM EPS power capability because of the several periods of minimal power capability during SL-3. All EPS parameter monitors, telemetry and onboard, continued to function properly. The anomaly affecting the Reg bus 1 current and SAS #4 current readings continued during this period but the work around procedure proved satisfactory for mission evaluation purposes.

The AM EPS continuous (SI) bus power capability was calculated as shown on Figure 2.7-24 over the duration of the SL-3 mission. The daily average AM EPS load curve is also shown on this Figure. It can be seen from these curves that a minimum power margin between the actual load level and the power capability occurred around DOY 220 and again around DOY 255. A downward drift trend was noted on several of the primary (controlling) A-H meter indications during these same time periods. It was also noted that in some of these instances the secondary A-H meter indications showed a more pronounced downward drift than their associated primary A-H meter indications. Real-time analysis of other battery parameters; voltage, current, and temperature; indicated that the batteries in question were being fully charged and that the drift was associated with A-H meter operation only. Further analysis of flight

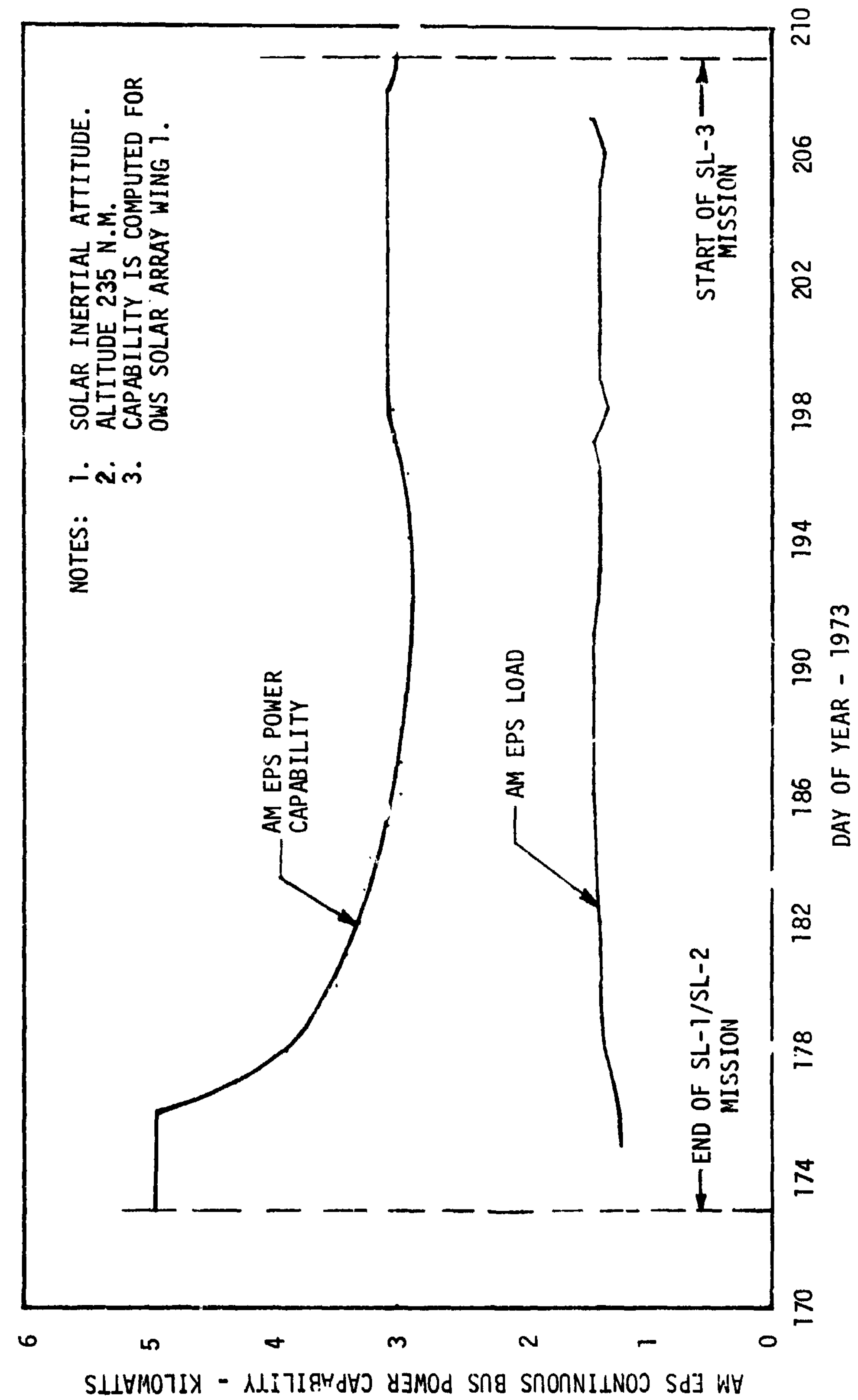


FIGURE 2.7-23 AM EPS BUS POWER FOR SL-2 TO SL-3 STORAGE PERIOD

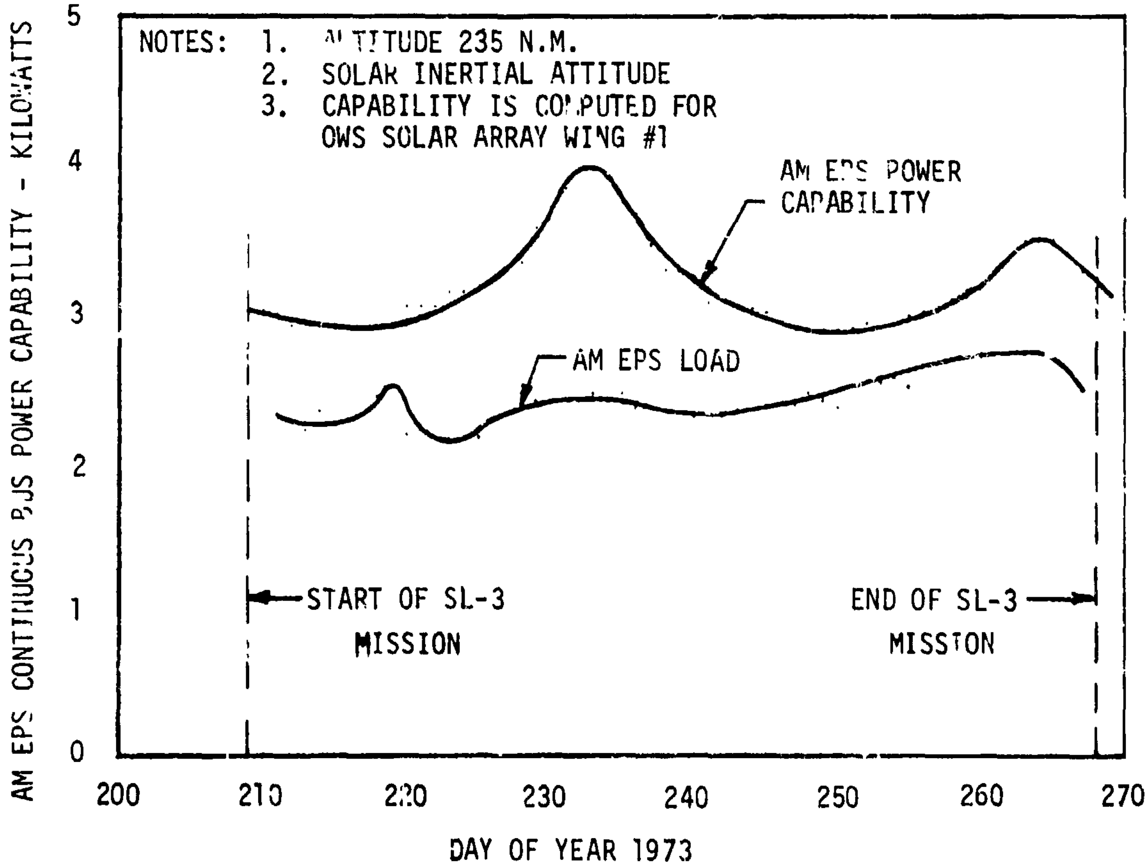


FIGURE 2.7-24 AM EPS BUS POWER CAPABILITY VERSUS DAY-OF-YEAR – SL-3 MISSION

data indicated that for the minimum power margins encountered during these periods, the return factor of the ampere hour meter was not being satisfied even though the battery was actually being fully charged. The erroneous A-H meter indications had no effect on system operation and following the minimum capability periods the indicated SOC's recovered to normal levels corresponding to actual battery status. Battery DOD's were as great as 18% for SI operation and 44% for EREP's during SL-3.

An inflight battery capacity determination test was performed on two AM batteries during SL-3. Battery #6 was tested on DOY 238 and #8 was tested on DOY 239. The inflight capacity values indicated an actual battery degradation rate less than the premission predicted degradation rate for AM batteries. These test results, coupled with observations of the other battery parameters, indicated that the batteries were continuing to provide good performance throughout this mission phase.

The cluster EPS configuration for the SL-3 to SL-4 storage period was changed from the normal storage configuration. This was done as a precautionary measure to ensure sufficient cluster power capability for the SL-4 manned mission. The possible problem of concern arose during SL-3 when the primary AM coolant loop was shutdown because of a loss of coolant fluid from the loop. This condition indicated that there would probably be only one AM coolant loop in operation during the SL-3 to SL-4 storage period and there would be no back-up system available. This would leave the AM battery modules without coolant flow if any contingency occurred during the storage period causing a shutdown of the operating coolant loop. This in turn would have caused a power system problem because analysis showed that the AM EPS could not be reconfigured by DCS commands during the storage period to protect all it's equipment from thermal damage if coolant loop operation was terminated.

Several actions were taken to verify the contingency problem and generate an acceptable plan to overcome it. A thermal test on an AM voltage regulator was performed at MDAC-E which verified that an equipment problem would exist under the contingency conditions. Several procedures were developed to permit a work around by DCS commands if necessary, while not adversely affecting power system operation or capability should the contingency not occur. These procedures were evaluated on the Skylab Cluster Power Simulator at MSFC for completeness, complexity, effectiveness, and system compatibility.

The procedure selected modified the SL-3 deactivation as follows. The Reg buses 1 and 2 potentiometers were set for OCV's of 27.15 volts after the CSM was disconnected and the two Reg/Transfer tie relays were then opened. This resulted in a storage configuration where the AM and ATM would operate isolated and each would supply only its own electrical loads. This would be changed only if the backup mode became necessary because of the contingency loss of the AM coolant loop. For the backup mode, the Reg/Transfer tie relays would be closed and the AM EPS would be reconfigured to a shutdown configuration. This would now be possible because the low OCV's of the Reg buses would allow the ATM EPS to supply the total cluster load. All AM EPS equipment would then be protected from thermal damage because the AM EPS would not be supplying any of the load even though all AM loads would be powered. The backup mode would continue until the AM coolant loops could be serviced and returned to operation at the beginning of the SL-4 manned mission. At that time the AM EPS, which had been protected from any damage, could be reactivated and the total cluster power system would be intact for the SL-4 mission.

E. SL-3 to SL-4 Storage Period - The AM EPS performed satisfactorily with no problems throughout this mission phase. Throughout this phase the system remained in a baseline configuration with the exception that the two reg/transfer tie relays remained in the open positions which were accomplished per the modified SL-3 AM EPS shutdown procedure. The Reg Bus 1 and 2 OCV's were set at 27.17 and 27.10 volts, respectively.

Figure 2.7-25 shows the calculated AM EPS power capability over this period. From the beginning of the period to DOY 315, the power capability varied in the range of 2600 to 3100 watts as the β angle varied from 45° to -45°. However, near the end of the period the power capability steadily increased as the β angle increased from 45° to 64° on DOY 320. The AM EPS load curve on Figure 2.7-25 shows that the actual load was

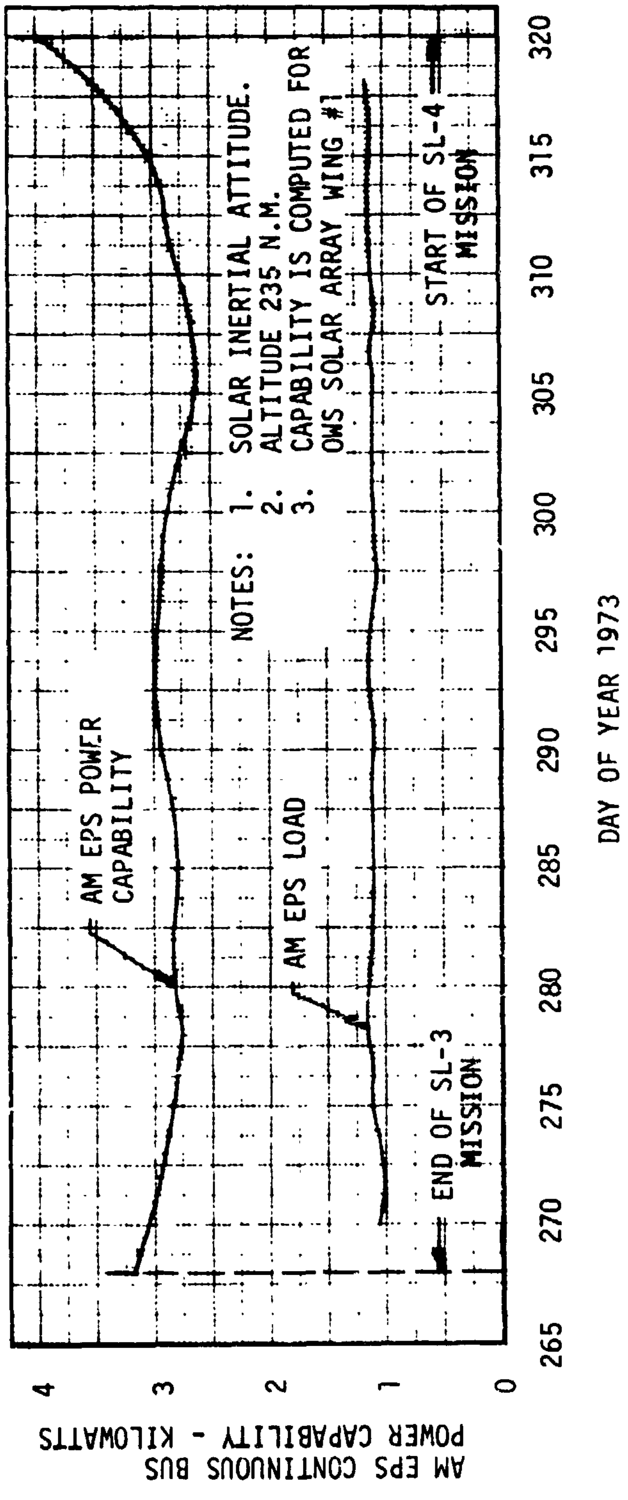


FIGURE 2.7-25 AM EPS BUS POWER FOR SL-3 TO SL-4 STORAGE PERIOD

well below the power capability at all times. The actual load was also an average of approximately 300 watts lower than that for the SL-2 to SL-3 storage period. This was due to zero power transfer from the reg buses to the transfer buses during this period and is also reflected in the lower battery DOD's which averaged approximately 7% during this mission phase.

A minor discrepancy was observed during this period. Beginning on DOY 298 through the end of the period, telemetry data indicated PCG #5 battery voltage slightly higher than normal during both the charge and discharge periods. Based on a thorough investigation as to the cause of the discrepancy, it was concluded that the high readings were due to an upward shift in the M137 telemetry parameter.

F. SL-4 Manned Phase - The AM EPS continued to perform well throughout this mission phase. All required operations involved with activation, deactivation, paralleling and EREP and Kohoutek observation periods were successfully completed.

The AM EPS continuous (SI) power capability was calculated to be as shown in Figure 2.7-26 over the duration of the SL-4 mission. The daily average AM EPS load curve is also shown on this figure. During the low beta period, (DOY 332, 1973, through DOY 9, 1974, and DOY 23 through the end of SL-4), the AM EPS power capability was closely approached and at times exceeded by the AM EPS load. In particular, during the many EREP and Kohoutek passes this condition occurred. This condition contributed to a downward drift of the ampere-hour meters during this period and some amp-hour meters never had the opportunity to recover to 100 percent SOC indication by the end of the mission. However, real-time analysis of other battery parameters indicated that all AM batteries were being fully charged throughout SL-4 except during some EREP and Kohoutek passes. Battery DOD's were as great as 19% for SI operation; 57% for EREP passes; and 32% for Kohoutek passes and 46% for JOP passes during SL-4.

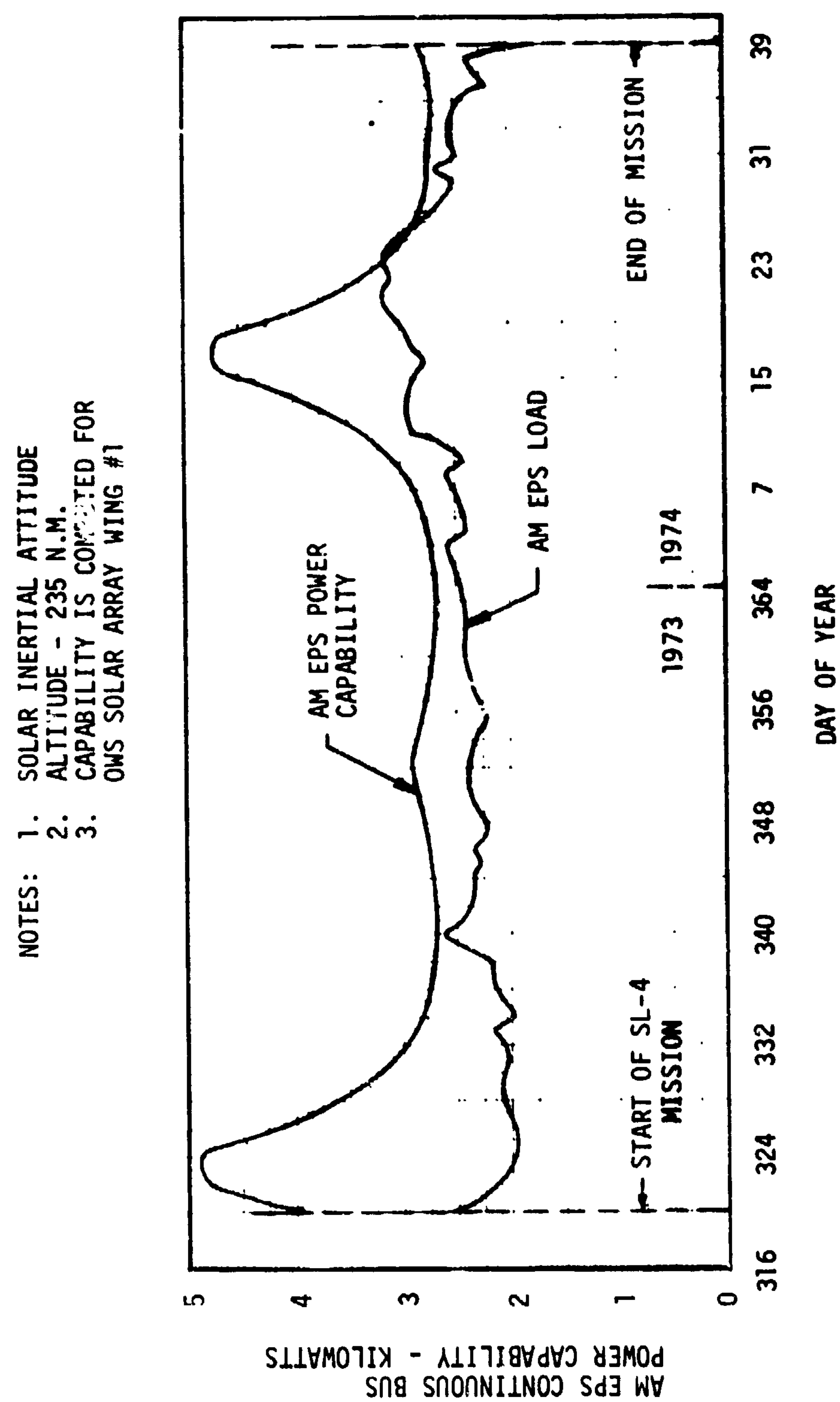


FIGURE 2.7-26 AM EPS BUS POWER FOR SL-4 MANNED MISSION

End-of-mission testing included capacity discharges of several batteries. Refer to Paragraph 2.7.4.3F for details of these tests.

The M137 telemetry parameter discrepancy observed during the SL-3 to SL-4 storage period returned to normal shortly into the SL-4 mission. During SL-4 several other telemetry parameters began to show erratic indications. However, real-time analysis of the AM EPS system was not significantly affected because alternate parameters were available.

2.7.4.2 Battery Chargers

A. SL-1 Launch Through OWS SAS Deployment - The battery chargers exhibited satisfactory performance both during and after the abnormal conditions encountered during partial deployment of SAS Wing #1. The battery chargers in PCG's 5, 6, and 7 operated with dual low power solar array inputs to satisfactorily charge their respective batteries to 100% SOC. SAS currents during the charging of these batteries were approximately 0.6, 1.2, and 0.5 amperes for SAS's 5 through 7, respectively. These current levels were well below the expected operating range for the battery chargers in any mode of operation.

The other batteries could not be charged because the array power available, even from dual solar array group combinations, was insufficient to operate the battery chargers. These battery chargers encountered another abnormal condition because of this extremely low solar array power. This condition can best be described as an oscillating input to the battery charger caused by the repetitive collapse and recovery of the solar array output characteristic. The array voltage would rise to the point where the battery charger bias circuits would turn on. The current drawn by the bias circuits, however, would pull down the solar array voltage to such a level, because of the low solar array power characteristic, that the circuits would turn off again. At this point the array voltage would recover to its original level and the cycle would repeat. Analysis of the battery charger circuits indicated that this condition should not cause any problems. As a safety factor, however, it was recommended that the condition be avoided by operating with the charger switch in the bypass position thereby removing the solar array output from the battery charger input. This recommendation was followed throughout most of this period. The data showed that the battery chargers in PCG's 1, 3, 4, and 8 definitely encountered this abnormal condition. All the battery chargers probably encountered it at some time during the period, however, no adverse effects on system or component performance resulted from this abnormal interface condition. Another abnormal condition occurred when the amp-hour (A-H) meters for PCG #8 were reset to 0% on DOY 147. This resulted when A-H integrator CB #8 on STS Panel 201 was inadvertently opened by crew action. After solar array power became available, the A-H meters for #8 soon returned to synchronization with the actual battery SOC and they operated normally thereafter.

B. OWS SAS Deployment Through End of SL-2 - The eight AM battery chargers performed normally through the end of SL-2. Each battery charger properly conditioned its associated solar array group input such that peak power was extracted upon demand during initial battery charging; limited its battery voltage as determined by battery temperature during the voltage limit charge mode; and regulated its battery current when the battery charger controlling ampere-hour meter indicated a 100% battery state-of-charge. Figure 2.7-27 illustrates the battery charge characteristics typical from OWS SAS deployment through the start of the all-sun attitude condition on DOY 172. The following sections describe battery charger functions in detail.

(1) Peak Power Tracking - Available solar array power was maximum at sunrise and it gradually decreased following sunrise as the solar array group temperature increased. The peak power tracking portion of the charger input power curve in Figure 2.7-27 illustrates the battery charger input power characteristic typical for all eight PCG's in the peak power tracking mode. As expected, the charger peak power tracker extracted maximum power from the solar array group immediately

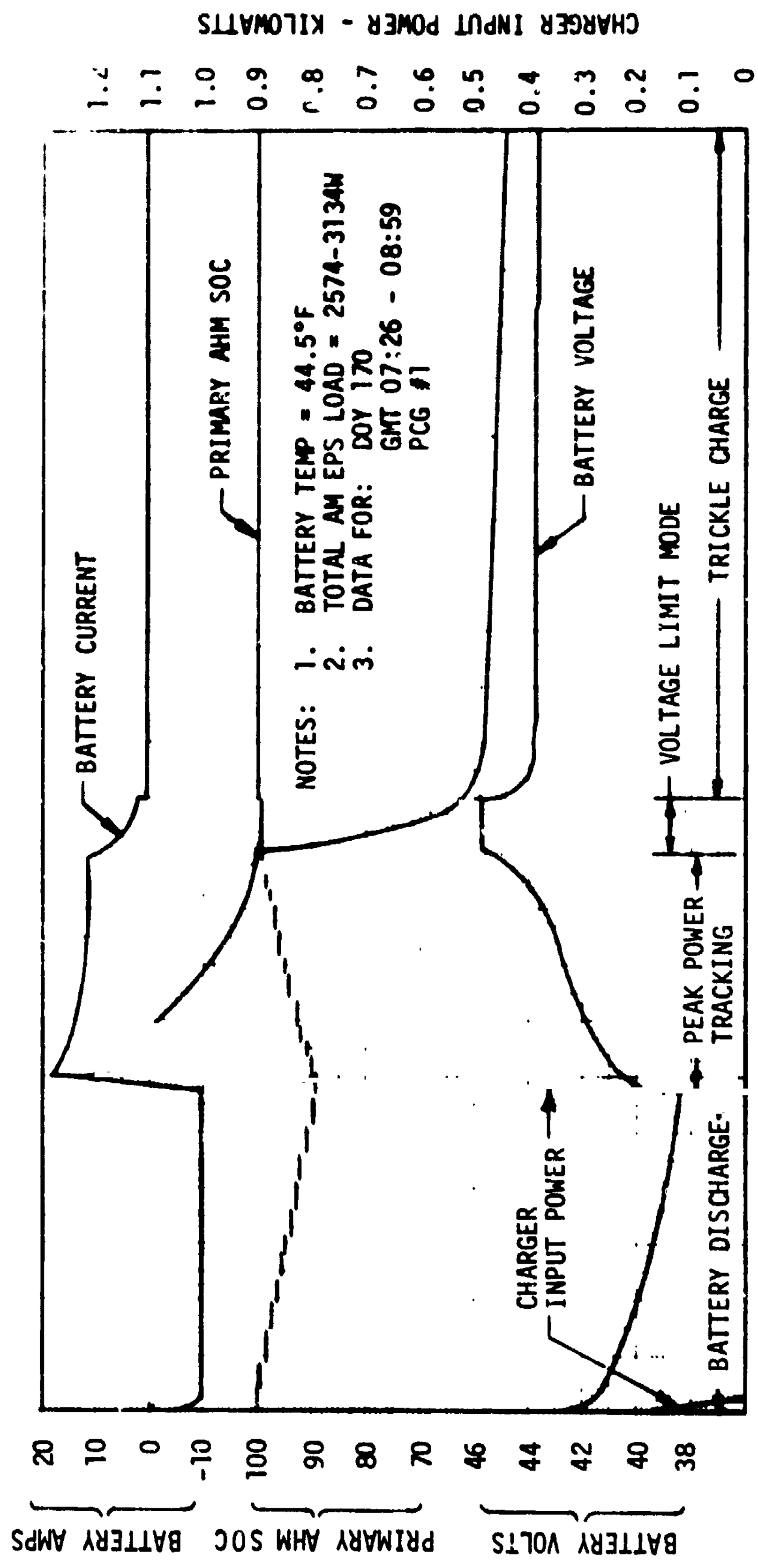


FIGURE 2.7-27 TYPICAL PCG ORBITAL PARAMETER VARIATIONS

upon sunrise and then decreased its demand as the available solar array group power decreased.

(2) Battery Voltage Regulation - The battery charging scheme was designed such that the battery voltage would not exceed a limiting value as determined by battery temperature. Figure 2.7-28 illustrates battery voltage limit values versus battery temperature over the entire operating battery temperature range of 40 to 50°F as experienced through the end of SL-2. Values are plotted for all eight PCG's in various orbits through SL-2. An anticipated performance curve is shown for comparison and a tolerance band for TM accuracy is also shown. The spread on the values for the various temperatures compares very favorably with anticipated performance values based on the battery charger acceptance test data.

(3) Battery Current Regulation - The battery current curve on Figure 2.7-27 shows the drop in current to the "trickle charge" level when the primary (controlling) ampere-hour meter reached 100% state of charge. The battery current then remained stable at 0.9 ampere throughout the trickle charge region.

(4) Ampere-hour Meter Control - Figure 2.7-29 compares the AHM SOC telemetry indications over one orbit to a calculated SOC over the same orbit. The calculated SOC value is based on battery current and temperature telemetry data and includes the temperature compensation factor during charge. Considering the telemetry accuracy limitation involved in the parameters used for the calculated curve and those on the direct SOC readings, the AHM SOC integration accuracy is seen to be very favorable.

(5) Efficiency - The battery charger efficiency was expected to be in the range from 90% to 94% for the operating conditions through the end of SL-2. There is no evidence that the battery chargers did not operate at this high efficiency.

C. SL-2 to SL-3 Storage Period - The AM battery chargers performed normally through the SL-2 to SL-3 storage period. An all-sun condition was experienced from DOY 172 until DOY 177. During this period solar power was continuously available. All eight batteries remained at 100% SOC. The battery chargers supplied the required bus loads through the voltage

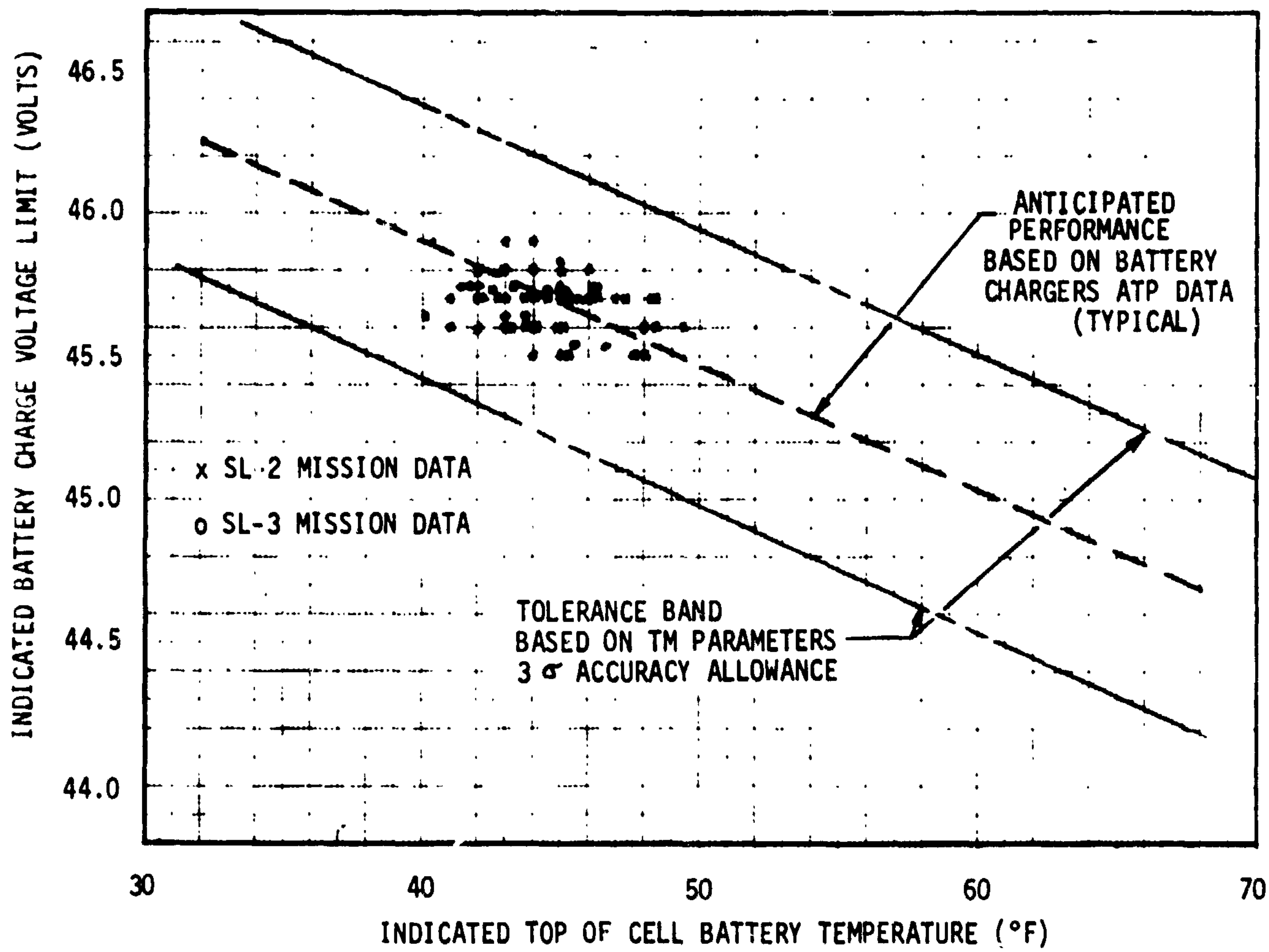


FIGURE 2.7-28 LIMITATION OF AM BATTERY CHARGE VOLTAGE - MISSION COMPOSITE

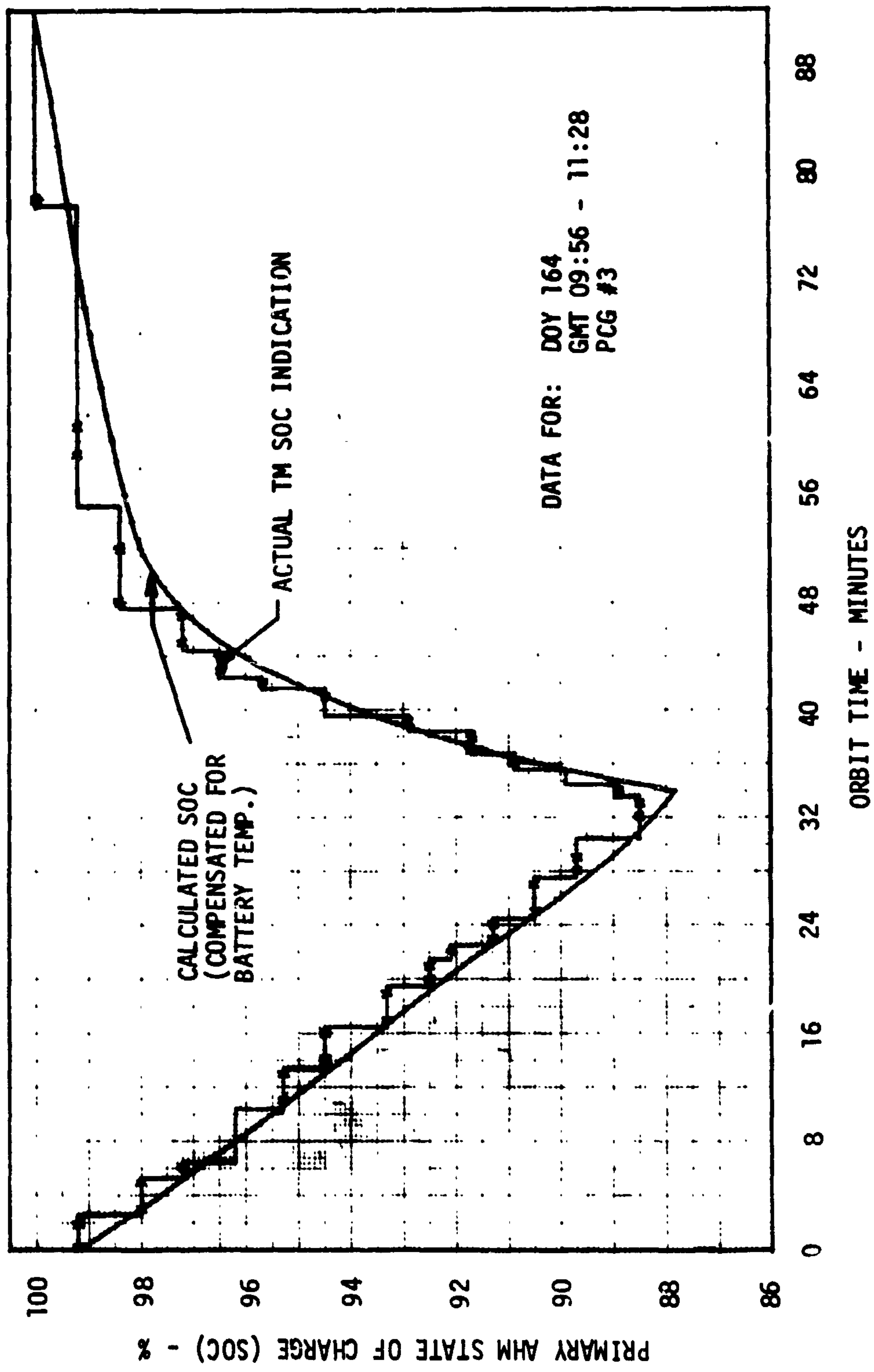


FIGURE 2.7-29 AMPERE-HOUR METER STATE-OF-CHARGE INTEGRATION ACCURACY

regulators and supplied trickle charge current to the batteries. Following the all-sun condition, day/night cycling performance continued to be normal. Performance in the area of charger peak power tracking, voltage regulation, charger efficiency, and ampere-hour meter control and indication continued to be normal.

D. SL-3 Manned Phase - The eight battery chargers performed normally throughout the SL-3 mission. Each battery charger continued to perform its required functions of peak power tracking, battery voltage limiting, ampere-hour meter control and battery current regulation under trickle charge conditions. The study of all available PCG parameters indicated satisfactory battery charger efficiency throughout the SL-3 manned phase. The curves of Figure 2.7-27 for SL-2 are also representative of the observed SL-3 battery charger performance.

(1) Battery Current Regulation - All battery chargers properly regulated the battery current at trickle charge levels when the controlling ampere-hour meter indicated a 100% state of charge. The design of the battery charger is such that the temperature compensated voltage limit cannot be exceeded under any circumstances. In some instances during SL-3, the characteristics of the batteries were such that the battery voltage required to maintain the normal trickle charge current was higher than the voltage limit. In these instances, the battery voltage was limited to the voltage limit value and as a result the battery current was reduced below the normal trickle charge level.

(2) Ampere-Hour Meter Operation - During the SL-3 mission there were several periods when the ampere-hour meter SOC indication did not return to 100% at the end of each daylight period although the battery voltage and current telemetry parameters indicated a fully charged battery. Divergence of controlling and backup AHM SOC indications were also observed during these periods. The reason for the downward drift of these AHM SOC indications was that for load levels approaching the power capability experienced at these times, the return factor of the ampere-hour meter was not being satisfied. The divergence between the controlling and noncontrolling AHM's was caused by the cumulative effects of small differences in the AHM accuracies when the AHM's are not returning to 100%. A

detailed explanation of the reasons for these effects is given in the subsequent paragraphs. The validity of these reasons is borne out by the following performance observations. There were no indications of any hardware malfunctions or failures during these periods and there were no system performance effects as a result of the drift of the AHM indications. The AHM indications returned to correlation with other battery parameters after short periods of operation at reduced load levels.

Although the return factor was only compensated for battery temperature variations, the actual return factor required varied a small amount with a number of other factors including battery depth-of-discharge (DOD), battery aging, etc. To allow for these other factors, the return factor was slightly conservative to assure that the battery was always fully charged when trickle charge was initiated. In addition to the design return factor being conservative, most of the flight AHM's exhibited a tolerance error in the device circuitry which was in the direction to increase the return factor.

At the beginning of a typical charge period, all available array power was delivered to the battery after the load was satisfied. The battery charger was peak power tracking at this time. As the battery accepted charge, battery voltage slowly increased to the temperature compensated voltage limit value. The battery charger then maintained this voltage until the battery SOC, as indicated by the controlling AHM, reached 100%. As the battery approached a fully charged state, the battery current decayed to a low level. If the AHM reached 100% SOC prior to the end of the daylight period, the battery charger switched to trickle charge. If the AHM integrated SOC had not reached 100% prior to the end of the daylight period, the return factor had not been satisfied and the battery charger maintained the voltage limit voltage at the battery terminals. This condition was observed for several ampere-hour meters during the SL-3 mission and was also observed in several ground system test programs. Although sufficient solar array power could have been available, the characteristics of the battery may have been, such, that in the charging time available, the battery current at voltage limit was so low that the battery would not accept sufficient charge to satisfy the AHM return factor. The battery was, in fact, achieving a fully charged state. This was demonstrated in ground test programs by capacity discharge testing of the battery after a number of cycles under these conditions.

Figure 2.7-30 shows a typical discharge/charge cycle during which the AHM SOC indication at sunset was less than the previous sunset. During this cycle, the battery voltage reached voltage limit 31 minutes after the beginning of the daylight cycle. The battery current then decayed to a level of approximately one ampere and maintained this level for the remainder of the charge cycle. The calculated ratio of ampere hours returned to the battery to the ampere hours removed (actual return factor achieved) was 1.061. Since the AHM was designed for a return factor of 1.075 at the battery temperature observed, the AHM SOC indication could not recover to the previous sunset level. If the condition of not satisfying the AHM return factor was maintained over a number of cycles, the AHM indication would decay downward. The AHM was an analog measurement device and as such contained some error, both in battery current measurement and in the utilization of the battery temperature sensor to establish the return factor. Also, there was a small error which could occur in the transition from charge to discharge or from discharge to charge. If the AHM was not returning to 100% SOC, the effect of these errors was not erased each cycle and could accumulate. These cumulative errors could cause divergence between the primary and secondary AHM's and in some instances could aggravate a downward trend of the AHM SOC indication. Thus, if the AHM had not returned to 100% SOC over a large number of cycles, the AHM would not have had a close correlation to the actual battery SOC and a divergence between the controlling and non-controlling AHM SOC indication would have occurred. Either or both of these conditions represented no compromise in system performance. Under these conditions, the battery charging current characteristic was the principal indication of the battery state-of-charge. Near trickle charge levels for the final minutes of charge at the battery voltage limit indicated that the battery was fully charged. Observation of battery voltage during discharge also provided an indication of proper battery state-of-charge and this was observed to be normal in each instance

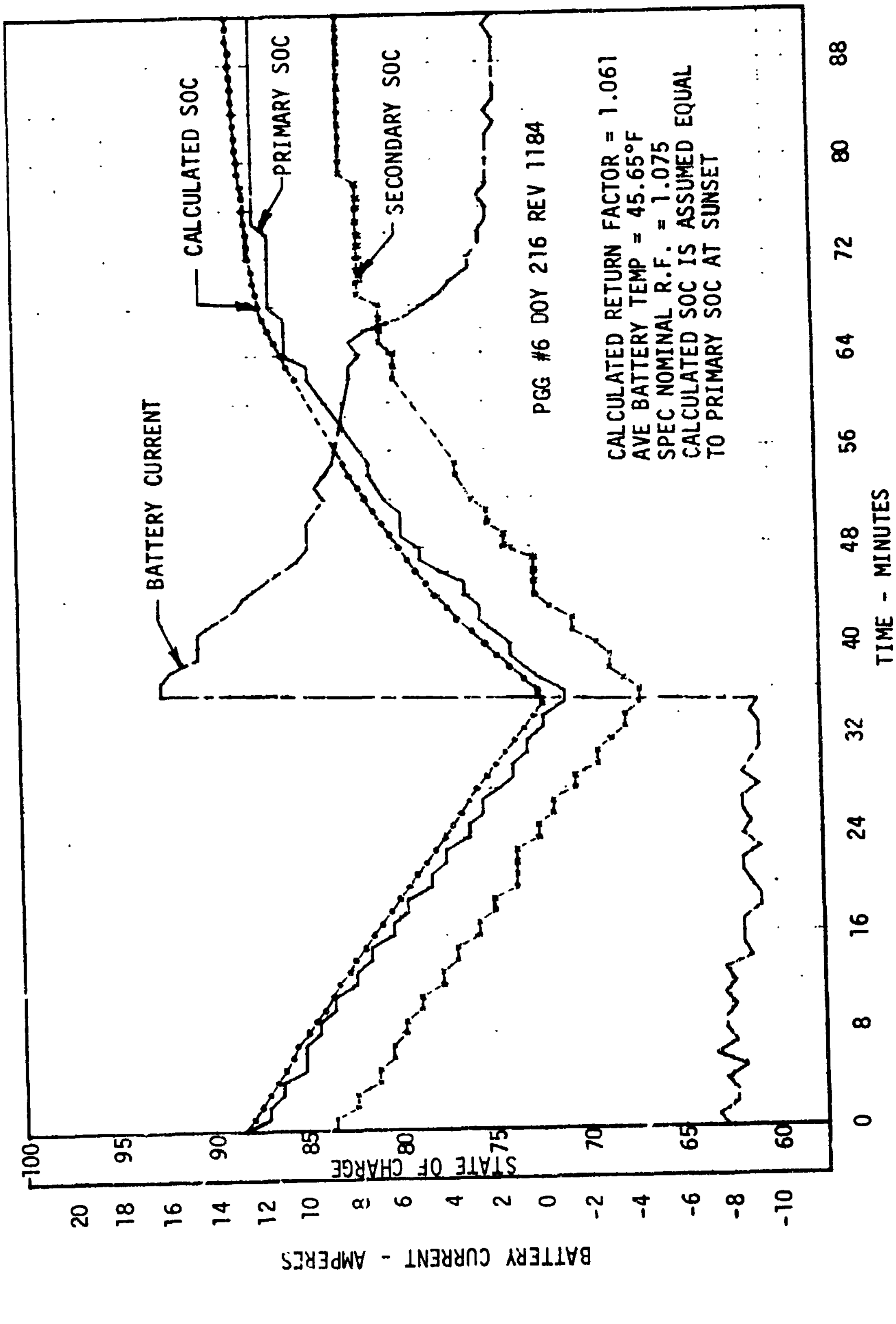


FIGURE 2.7-30 BATTERY STATE-OF-CHARGE INTEGRATION

observed on SL-3. The probability of not satisfying the AHM return factor increased sharply as the AM EPS load demand approached the AM EPS power capability. An analysis of data during SL-3 indicated that the downward trend of the AHM SOC indications occurred during periods in which the actual AM EPS load approached or exceeded the calculated AM EPS continuous power capability. The potentiometer adjustments, both reg bus and fine adjust affected the amount of power provided by each PCG and therefore affected which PCG's exhibited the downward drift in SOC indication at any particular time.

When the conditions of load demand and available array power were such that the AHM return factor was again satisfied, the SOC indication would recover toward a 100% sunset indication. The rate at which the AHM SOC indication recovered was directly dependent on the amount by which the battery recharge ampere hours exceeded the amount required to satisfy the AHM return factor. Figure 2.7-31 represents a typical cycle during which the AHM recovered. The primary AHM began the discharge period with a 75.6% SOC. A depth of discharge (DOD) of approximately 12.9% indicated a moderately light load. During the charging period, sufficient current was delivered to the battery to attain an indicated primary AHM SOC of 78.0 at sunset, a recovery of 2.4%. At the end of SL-3, all AHM's except PCG #5 secondary were returning to 100% at sunset. With the light orbital storage loads, the #5 secondary SOC recovered from a 38% sunset indication to a 100% sunset SOC indication in five days.

E. SL-3 to SL-4 Storage Period - The AM battery chargers operated normally through the SL-3 to SL-4 storage period. Performance in the areas of charger peak power tracking, voltage regulation, charger efficiency and ampere-hour meter control and indication was normal. All primary and secondary ampere-hour meter state-of-charge indications with the exception of PCG #5 secondary AHM, were 100 percent at the start of the storage period. PCG #5 secondary ampere-hour meter state-of-charge indication recovered to 100 percent several days after the start of this storage period.

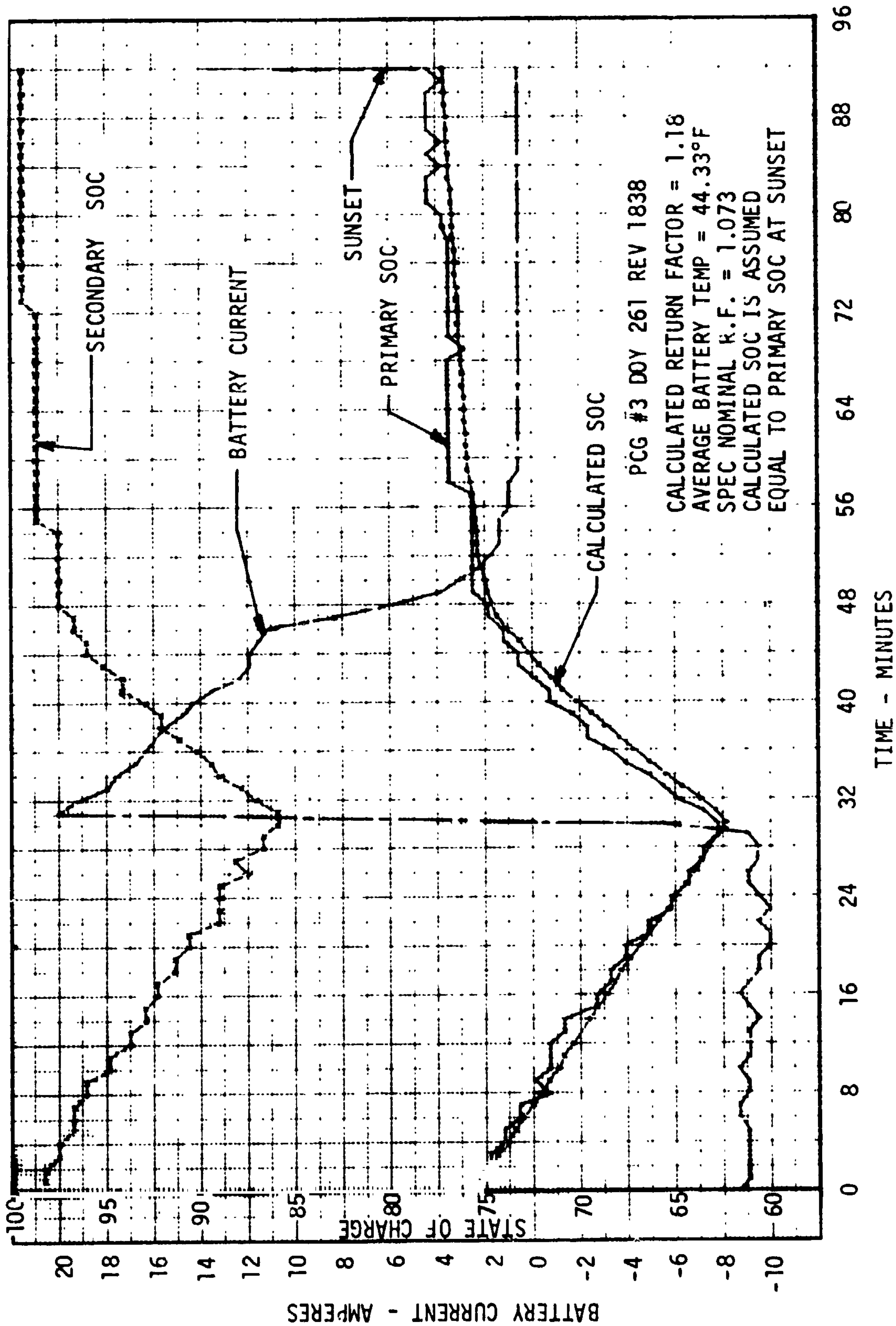


FIGURE 2.7-31 PCG NO. 3 BATTERY STATE-OF-CHARGE ACCURACY

F. SL-4 Manned Phase - The eight battery chargers performed normally throughout the SL-4 mission. Each battery charger continued to perform its required functions of peak power tracking, battery voltage limiting, ampere-hour meter control and battery current regulation under trickle charge conditions. The study of all available PCG parameters indicates satisfactory battery charger efficiency existed throughout the SL-4 manned phase. The curves of Figure 2.7-27 for SL-2 are also representative of the observed SL-4 battery charger performance.

(1) Ampere-hour Meter Operation - During the SL-4 mission, as during the SL-3 mission, there were several periods when the ampere-hour meter SOC indication did not return to 100% at the end of each daylight period although the battery voltage and current telemetry parameters indicated a fully charged battery. Divergence of controlling and backup AHM SOC indications were also observed during these periods. Paragraph 2.7.4.2-D(2) describes the reasons for this ampere-hour meter operation.

2.7.4.3 Batteries

A. SL-1 Launch Through OWS SAS Deployment - The eight AM EPS batteries provided power to cluster loads during the SL-1 launch phase per the mission plan. The batteries were then turned OFF when it was determined that OWS solar array power would not be available in the near future. This occurred at approximately 1930 GMT on DOY 134. Battery SOC's at this time ranged from 65% to 68%. The batteries were turned OFF to retain their stored capability as back-up power sources for peak power periods such as EREP passes, and to retain maximum flexibility in managing the batteries as the mission progressed.

All batteries were turned ON on DOY 144 in preparation for the first EVA attempt to deploy SAS Wing #1; however, they provided power only to the EPS control buses because the PCG output switches were OFF. All batteries were turned OFF again on DOY 145 after approximately 8 hours of operation. Battery SOC's for PCG's 1-4 at this time ranged from 48% to 53%. Batteries 1 through 4 remained OFF until DOY 158 when they were turned ON as part of the preparation for the successful attempt to deploy OWS Solar Array Wing #1. No other interim activities were initiated with these batteries because there was negligible solar array power available to their PCG's. Batteries 5 through 8 were cycled ON and OFF at various times for troubleshooting purposes and attempted charging. Eventually, on approximately DOY 155, batteries 5, 6, and 7 were recharged to 100% SOC. Battery 8 could not be recharged because the available solar array power from SAS Groups 7 and 8 combined was still insufficient to operate the battery charger in PCG #8.

All eight batteries spent the greater part of the initial 24-day mission period turned OFF while in a partially discharged state. This constituted an abnormal storage condition for the batteries. Prelaunch ground storage periods had conformed to one of two recommended storage conditions: (1) storage in a discharged state (18A discharge rate to 30V) for long storage periods, and (2) storage in a fully charged state with a periodic (weekly) boost charge. The abnormal battery storage condition was evaluated in real time and a decision was reached that no special operations would be required to condition the batteries when solar array power became available. One of the key factors in this decision was that the temperatures of the batteries were stabilized in the range of 45 $\pm$ 5°F over this period of time, and the internal chemical reactions at this temperature range would have negligible effect on the battery characteristics. The batteries responded as expected when solar array power became available and regained their full capability in a very few orbits.

B. OWS SAS Deployment Through End of SL-2 - Battery indicated states of charge at times prior to and after solar wing deployment were as shown in the table below.

DOY	GMT	PRIMARY AH METER SOC INDICATIONS (%)								
		BTY	1	2	3	4	5	6	7	8
158	17:00		45.8	45.8	50.7	48.3	96.2	99.0	95.5	0
158	20:07		55.4	54.1	62.7	56.2	99.8	100	100	21.4

NOTE: Wing fairing deployed on day 158 at 18:00 GMT.

All batteries demonstrated an ability to accept charge while exhibiting anticipated voltages. This indicated that no adverse electrolyte distribution pattern had resulted from the partially charged open circuit exposure. Recovery of the state-of-charge indication for battery 8, as shown in Figure 2.7-32, provided qualitative information on charge retention during the abnormal partially charged open circuit exposure. The indicated state-of-charge of battery #8 on day 147 was approximately 45% at which time it was inadvertently reset to zero. Figure 2.7-32 shows that once the measured 55% depletion was returned to the battery, a different rate of recovery of the Ampere-hour meter developed. This suggests that no appreciable capacity loss was experienced due to the open circuit exposure.

Battery cyclic performance from the time of solar array deployment until the undock of the CSM was good. Two hundred nineteen battery cycles were accumulated in the course of the SL-1/SL-2 mission. The battery parameters for a representative orbital cycle are shown on Figure 2.7-27. The discharge and recharge modes of battery operation are distinguishable by examination of the figure.

The depth of discharge range most commonly experienced during the SL-2 mission was 12 to 14%. Depths up to 30% were experienced during high activity periods or periods of other than solar inertial vehicle attitude. Composite battery discharge experience is presented in Figure 2.7-33. The data shown on these graphs covers the operating period from DOY 162 through DOY 1/2 and includes data points for all eight batteries. The telemetry data was scanned to obtain discharge current rates in the three different ranges; 6.0 to 7.9 amperes, 8.0 to 9.0 amperes and 9.1 to 11.0 amperes. Data points were then selected for each range to cover as great an SOC range as possible. A composite curve is then shown on each graph. The curves and the data spread indicate repeatable discharge voltage characteristics over the depths most consistently experienced during the mission. Where special mission activity resulted in depths of discharge greater than normally experienced, the development of a plateau can be seen. Similar results have been experienced in previous AM battery test programs. The data dispersion seen on these graphs results from instrumentation accuracy tolerance allowances; ± .25 volts on individual voltage readings and ± 2.5% on SOC readings.

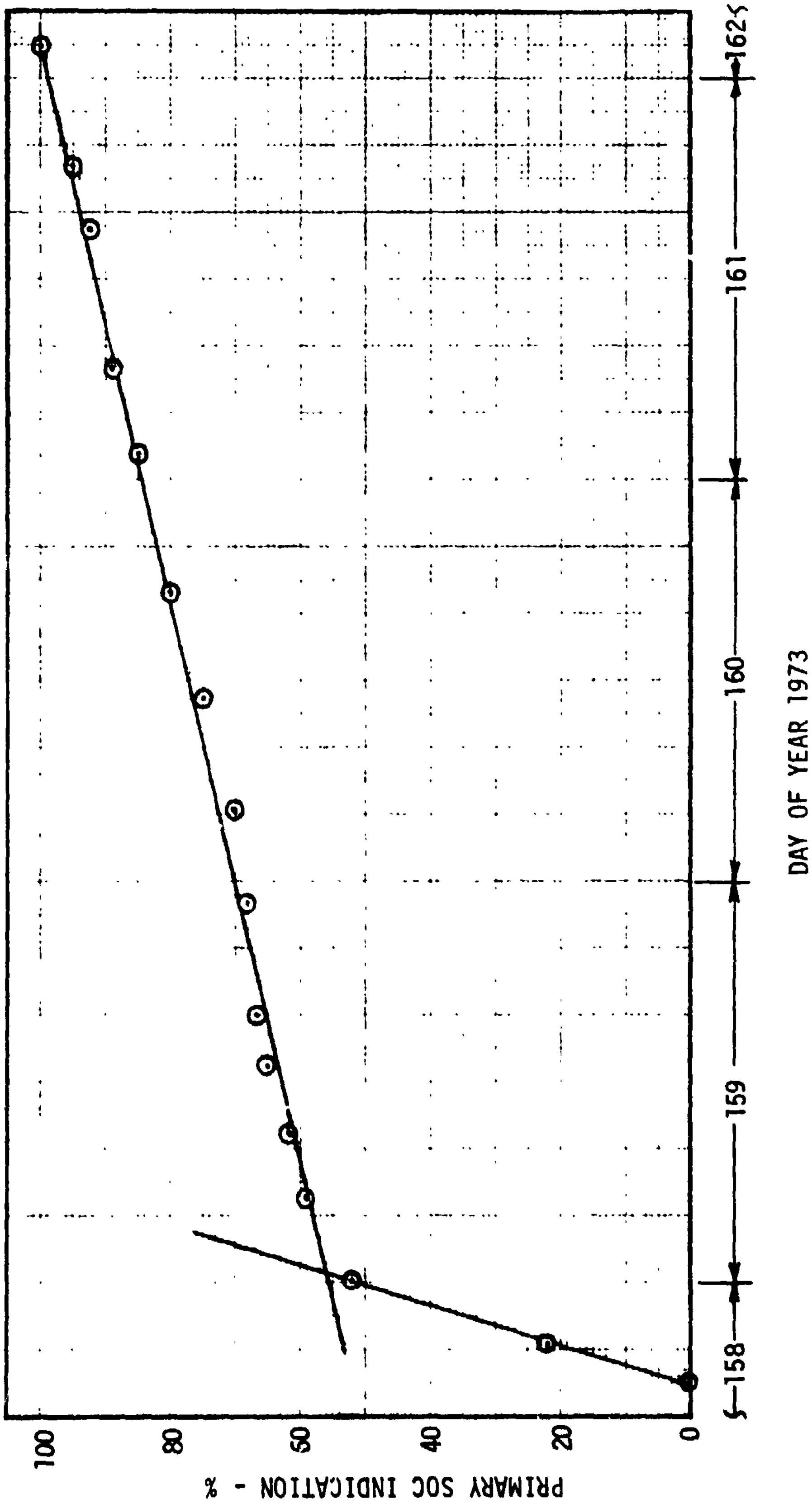


FIGURE 2.7-32 PCG NO. 8 BATTERY STATE-OF-CHARGE RECOVERY

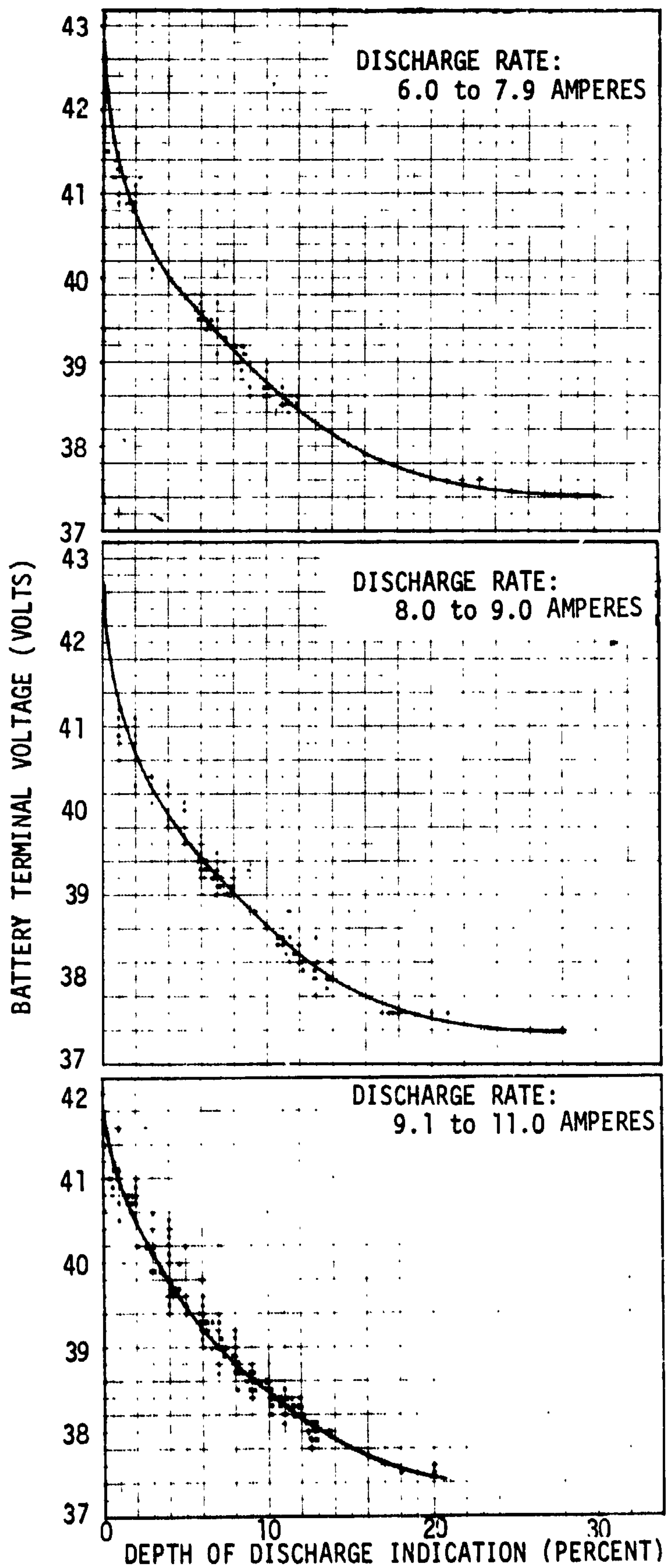


FIGURE 2.7-33 SL-2 MISSION COMPOSITE AM BATTERY DISCHARGE CHARACTERISTICS

It was anticipated that the coolant inlet temperatures to the batteries would exceed the vernatherm control valve setting of 39 ± 3°F and reach as high as 55°F during periods of high crew activity (EVA) or non-solar inertial attitude. These conditions did not materialize and the vernatherms maintained continuous control. Indicated top of cell battery temperatures consistently fell in the 40 to 50°F range. This was a favorable temperature range for battery cyclic life.

Batteries supplying the same AM Regulated bus exhibited a uniformity of performance which made astronaut adjustment of the regulator fine trim pots unnecessary. Typical data which shows this uniformity is tabulated as follows:

Reg Bus	Batt	Start of Discharge		End of Discharge		Min Pri SOC
		Volt	Current	Volt	Current	
1	1	41.78	8.74	38.32	9.44	88.5
1	2	41.79	8.66	38.23	9.37	88.1
1	3	41.99	8.59	38.23	9.38	88.5
1	4	42.08	8.68	38.23	9.39	89
2	5	41.57	8.97	38.02	10.15	87.5
2	6	41.68	9.39	38.03	10.18	86.3
2	7	41.88	9.30	37.83	10.97	87.4
2	8	41.78	9.53	37.83	11.27	86.3

The discharge current of each battery increased slightly as its voltage decreased. The constant regulator power demand on the battery caused this trend.

C. SL-2 to SL-3 Storage Period - Battery performance remained uniformly acceptable for all batteries during the storage period. Continuous solar energy was available to power the vehicle for the initial four days of this storage period because of high Beta angle conditions. The batteries, therefore, were subjected to continuous charging at the trickle charge rate for this four-day period. The battery temperatures, however, remained stable over the period. Also, the charging potentials required to sustain the trickle charge rate for each battery converged toward a more uniform level as expected. The batteries resumed normal cyclic operation on June 26th (DOY 177) and cycled at an average DOD of approximately 9% during the remainder of the storage period. Cycle accumulation at the time of SL-3 launch (DOY 209) totalled 772 cycles.

D. SL-3 Manned Phase - The AM batteries performed well during this period. The batteries had accumulated 1683 flight cycles at the time the SL-3 crew departed on DOY 268. The depth of discharge range most commonly experienced during the SL-3 mission was 13 to 16%. Forty-one earth resource experiment package (EREP) passes were performed during this mission. Battery depths of discharge were generally greater during these passes than during normal solar inertial operation. The maximum DOD experienced, occurred during the final EREP (DOY 264) where it ranged from 36 to 42.7%.

AM batteries were actively cooled. Parallel coolant flow at controlled temperatures (40 + 2 - 4°F) was provided to coldplates for PCG batteries 3, 4, 7 and 8. The coolant from these coldplates flowed to coldplates for PCG batteries 1, 2, 5 and 6, respectively, such that the coolant inlet temperatures were increased by the heat pickup from the battery first in line. A coolant loop system operational change was made on DOY 237. This change decreased the coolant mass flow by approximately 50% and the effects were detectable by an increase of approximately 2°F in the operating temperatures of PCG batteries 1, 2, 5 and 6. Temperature changes of PCG batteries 3, 4, 7 and 8 were too small to be detected in the telemetry scatter. Other than this detected increase, the indicated top of cell (TOC) battery temperatures were comparable to those experienced during the SL-2 mission.

Two of the eight AM batteries were purposely deep discharged during the SL-3 mission to determine their available capacities. Capacity of AM batteries has been determined in ground tests by measuring the ampere-hours extracted at an 18 ampere discharge rate to an end voltage of 30.0 volts. The in-flight discharge procedure deviated from the ground practice in that the astronauts terminated the discharge when they detected a terminal voltage of 33 volts. The charger ampere-hour meter state-of-charge indication was used to measure the obtained capacities during these discharges, i.e., (100 minus final state-of-charge percent indication x 33 ampere-hours = ampere-hours obtained). The results of these flight discharges are shown in Figures 2.7-34 and 2.7-35. The change in the general shape of the discharge characteristics since the acceptance testing of the units can be seen by examination of the figures. The characteristic exhibited at acceptance testing has, in both cases, changed to one where an initial voltage plateau develops at a lower level than the single plateau of the acceptance characteristic. The roll-off from this initial plateau occurs much sooner and is more gradual than the acceptance test curve. The final few data points before the termination of the in-flight discharges indicated the development of a second lower plateau. The formation of a second plateau was compatible with ground test experience. The increased prominence and duration of this second plateau and the recession of the initial plateau was believed to be partially a function of cycle accumulation.

Composite battery discharge experience for the SL-3 mission is presented in Figure 2.7-36. The SL-3 data shown on these graphs cover the operating period from DOY 209 through DOY 268. These data were selected and are presented in the same manner as the SL-2 composite data. A comparison of the SL-3 and SL-2 composite data indicated a detectable recession of the initial discharge characteristic plateau from what it was during SL-2, as previously stated.

A condition where some ampere-hour meters drifted from what was believed to be the actual state-of-charge of the batteries during the SL-3 mission is discussed in the SL-3 battery charger discussion. Whereas battery terminal voltage was not an accurate means of determining individual battery state-of-charge, a comparison of several battery discharge terminal voltages at like delta ampere-hour extraction points provided an indication of SOC status. This was done for several discharges occurring in the ampere-hour meter drift periods of SL-3 and showed comparable voltage levels for all the batteries. This voltage level consistency, coupled with lack of a voltage degradation trend indicated in a qualitative way, that all the batteries were being fully charged irrespective of the ampere-hour meter indications.

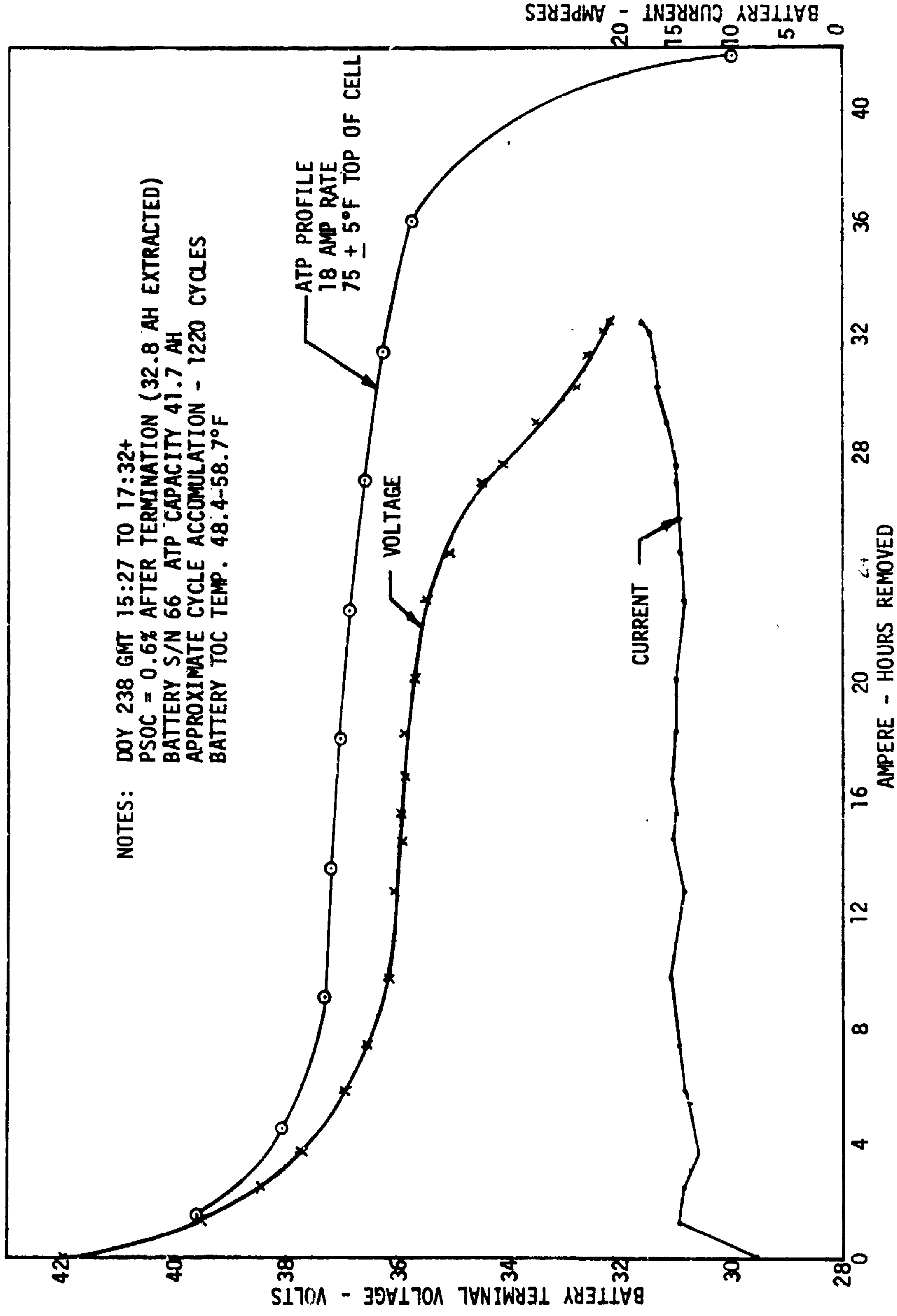


FIGURE 2.7-34 PCG NO. 6 INFLIGHT CAPACITY DISCHARGE

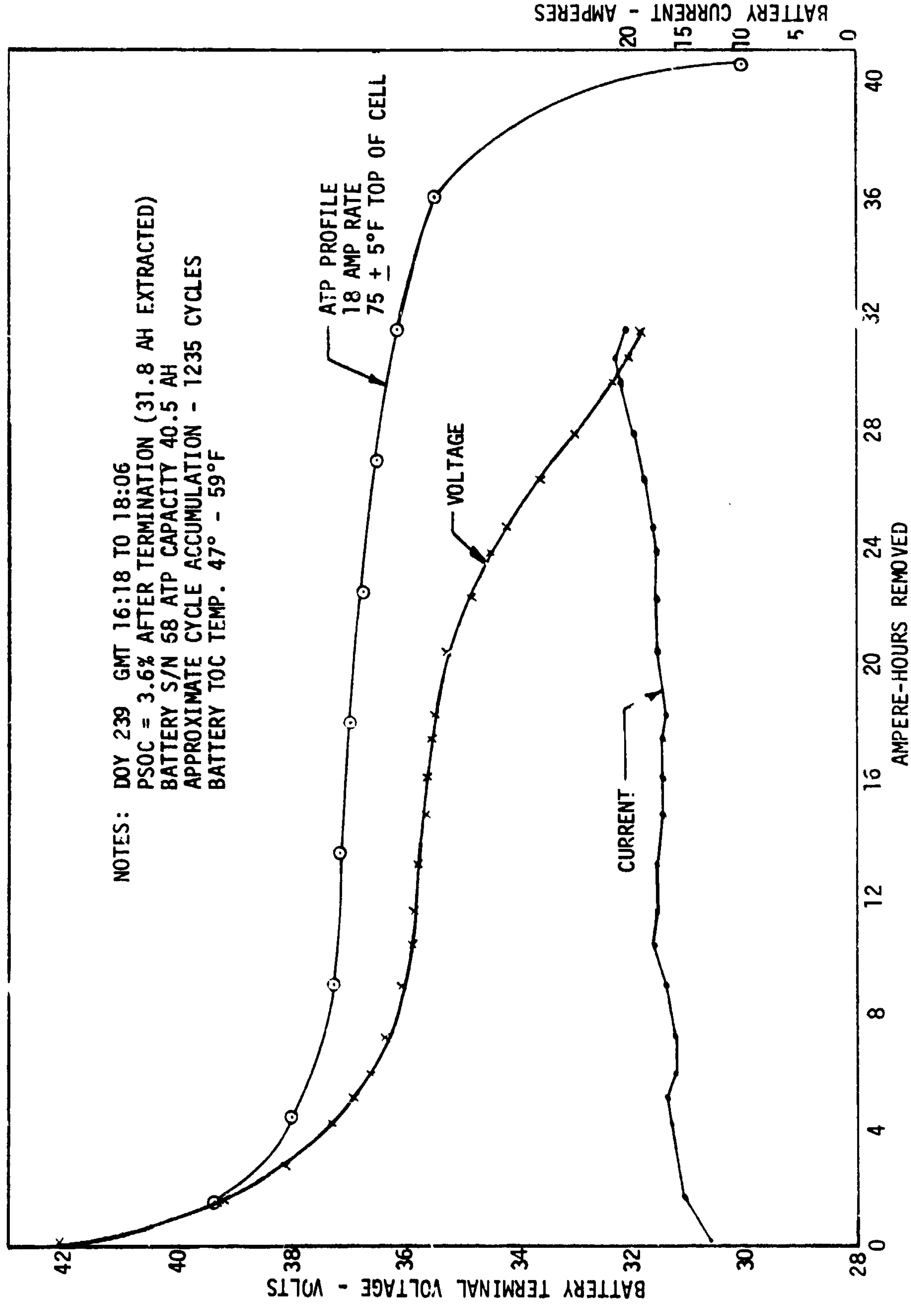


FIGURE 2.7-35 PCG NO. 8 INFLIGHT CAPACITY DISCHARGE

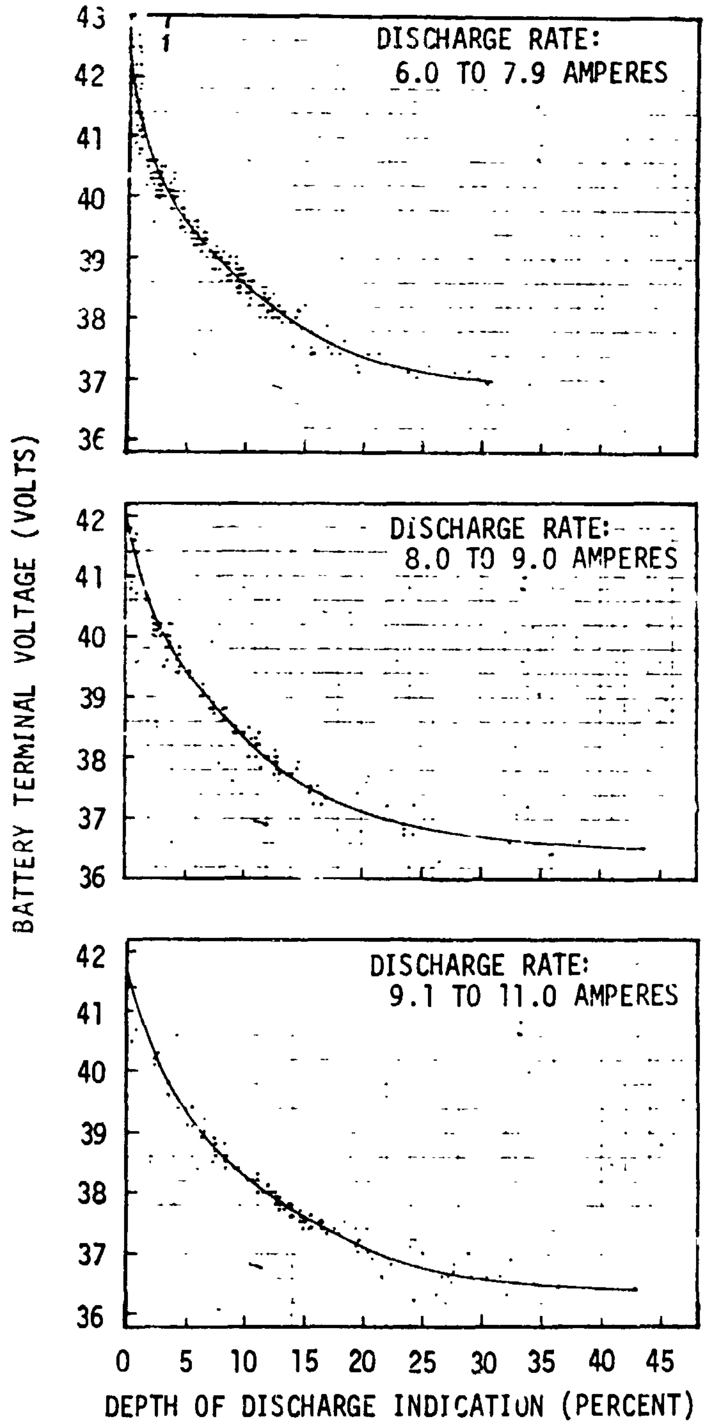


FIGURE 2.7-36 SL-3 COMPOSITE AM BATTERY DISCHARGE CHARACTERISTICS

E. SL-3 to SL-4 Storage Period - All AM batteries continued to perform satisfactorily throughout this period. Contingency planning called for discontinuing PCG operation during this phase in the event of coolant loop depletion. However, the batteries cycled throughout the entire storage period, and execution of the contingency plan was unnecessary. By the time of the launch of SL-4, the batteries had accumulated 2486 cycles. The cycle depths, which averaged approximately 7% during this period, were less than those of the first storage period because of the EPS configuration established per the modified SL-3 AM EPS shutdown procedure.

F. SL-4 Manned Phase and End-of-Mission Testing - The AM battery discharge/charge cycle accumulation, at the time the SL-4 crew departed on 8 February 1974, was 3790 cycles. The range of discharge depths experienced during the solar oriented periods was 12 to 19%. Discharge depths near 50% were common for the off-sun experiment orientations with the maximum depth reaching 57%. Composite battery discharge experience for the SL-4 mission is presented in Figure 2.7-37.

One hundred and ten nonsolar oriented attitudes were established in the course of the mission for Earth Resource and Comet Kohoutek observations. Failure of a control moment gyro, on 23 November 1973, resulted in more off-sun attitude time than normally would have been required to accomplish the desired observations. AM battery performance was uniform and reliable during the mission. Their ability to sustain the heavy depths of discharge dependably contributed to the high success level of the mission.

Capacity discharges were performed on PCG 6 battery at the beginning, in the middle, and at the end of the SL-4 manned phase. The first two discharges were performed according to the procedure used in the SL-3 mission while the 3rd SL-4 discharge was continued until the battery terminal voltage reached 30.0 volts. As mentioned previously, the 30.0 volt termination was consistent with ground test practice. The results of these tests are shown in Figure 2.7-38. A consistent pattern of battery output voltage regulation degradation with increasing cycle accumulation can be seen when SL-3 capacity discharge information for PCG 6 is added to the information contained in Figure 2.7-38. This

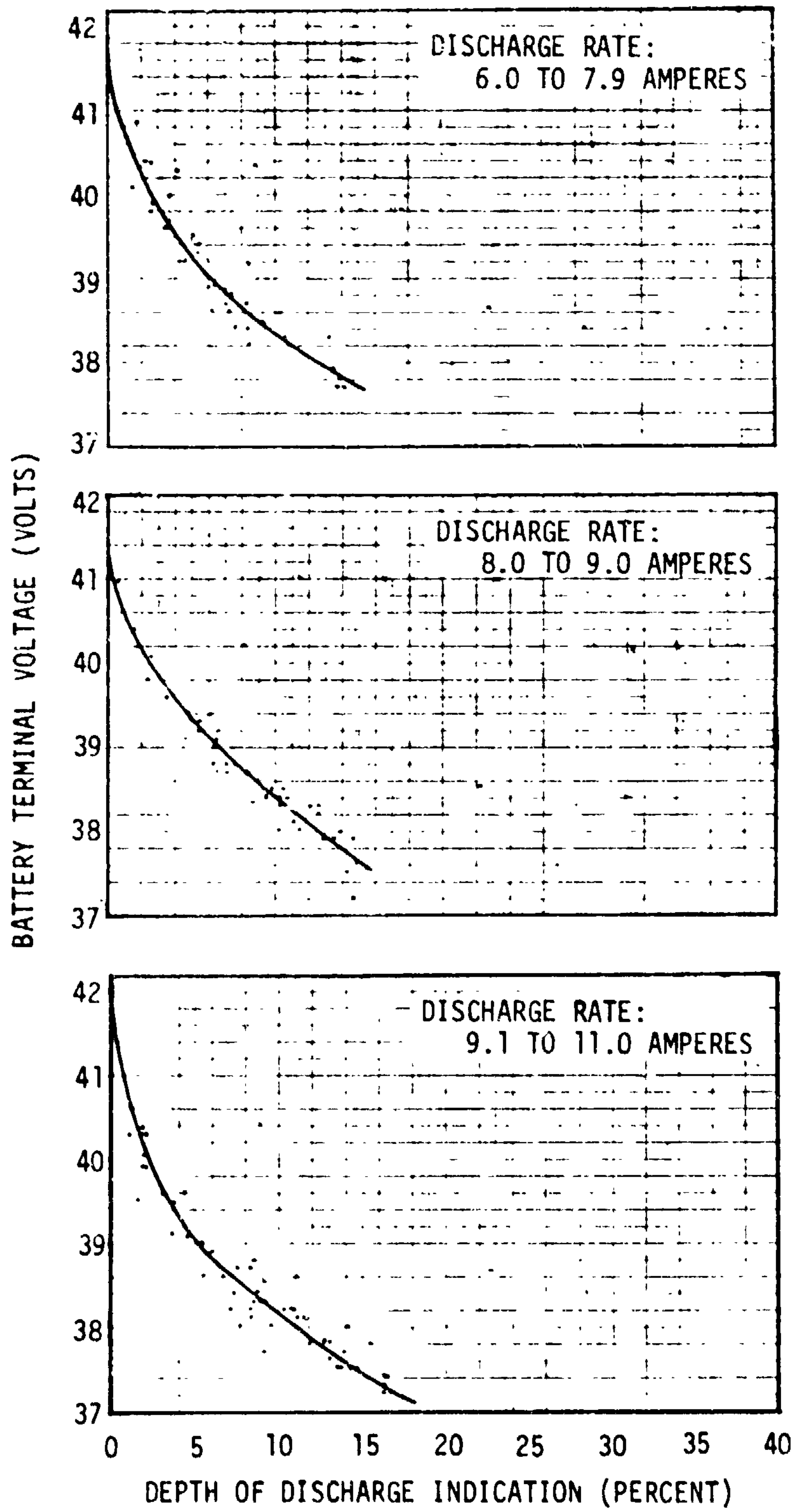


FIGURE 2.7-37 SL-4 COMPOSITE AM BATTERY DISCHARGE CHARACTERISTICS

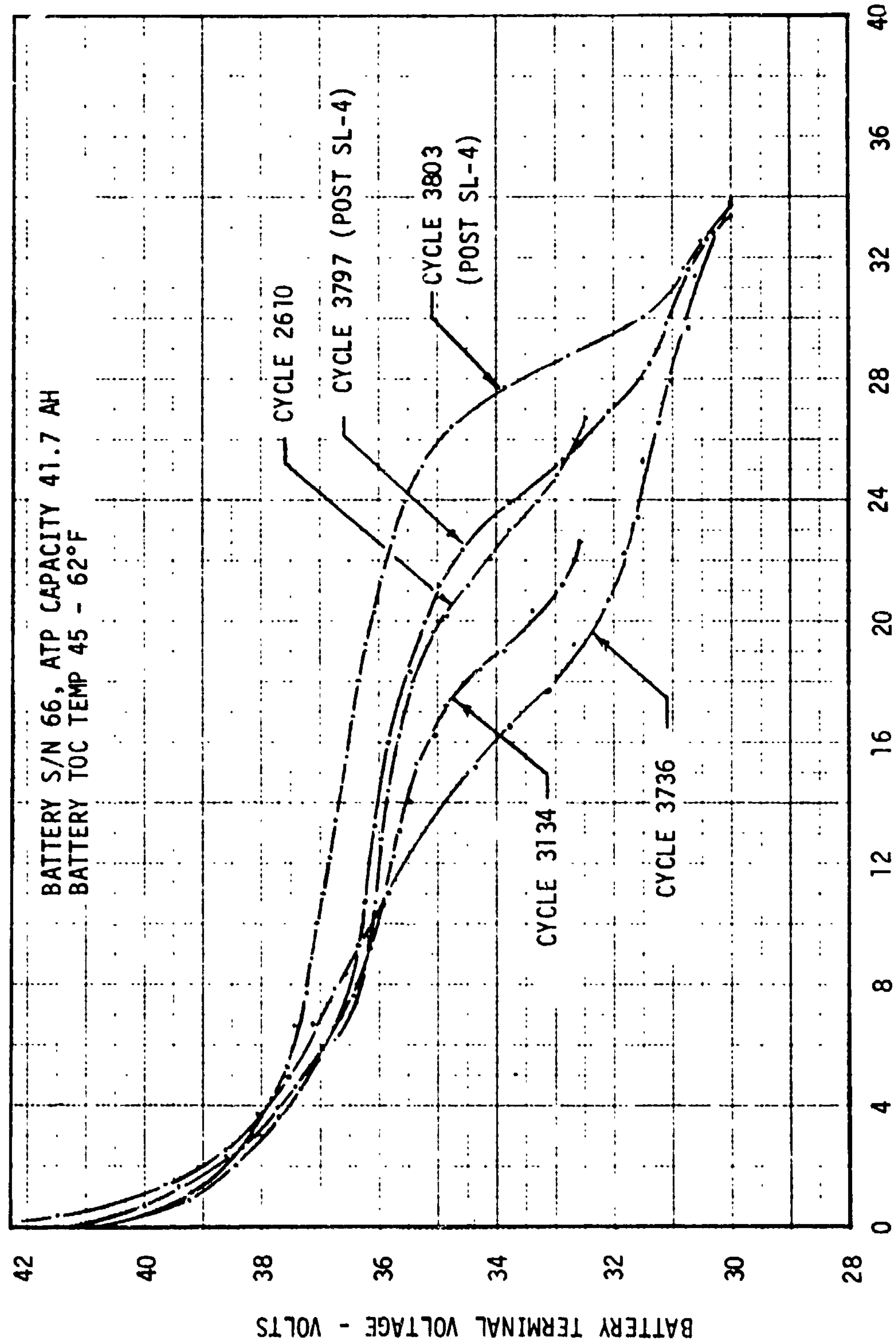


FIGURE 2.7-38 PCG NO. 6 INFLIGHT CAPACITY DISCHARGE

progressive pattern of nickel-cadmium "memory" development is apparently minimally effected by incomplete capacity discharges. More will be said concerning this subject in the end-of-mission test discussion.

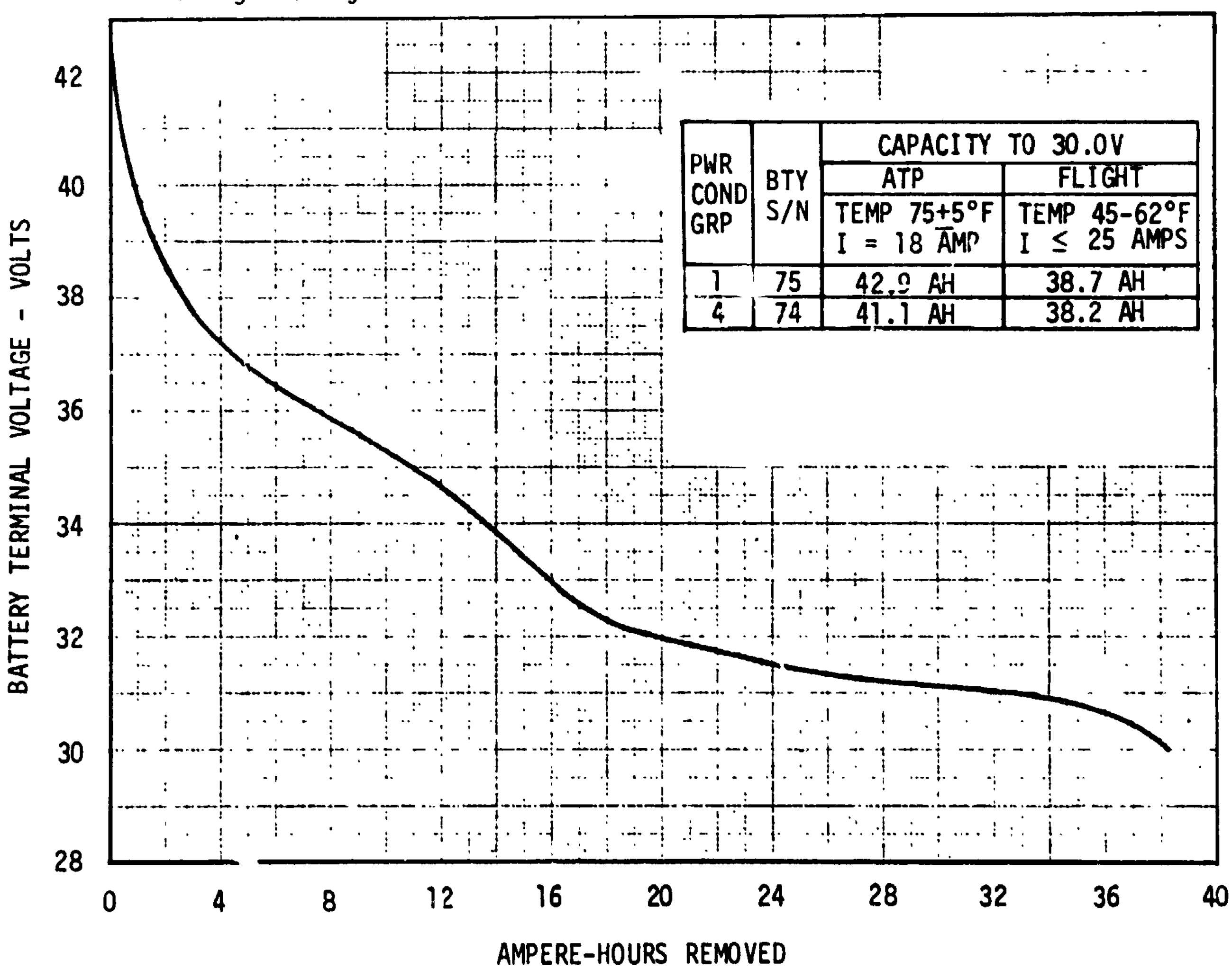
A special EPS configuration was established as part of the SL-4 crew closeout of the Skylab. This was done in anticipation of capacity testing of all AM batteries after crew departure. The devised system configuration allowed ground selection of any one of the eight AM batteries for discharge, established discharge rates near the ground test level of C/2, permitted continuous discharge of the selected battery to a 30.0 volt completion, and provided a self limitation of battery discharge as the battery terminal voltage approached 29.0 volts. The last feature was desirable as ground station coverage could not be assured at every critical discharge time. The flexibility of the AM EPS control capability proved invaluable in accomplishing the test objectives.

All eight AM batteries were discharged to 30.0 volts during the post SL-4 test period. In addition, PCG 6 and 8 batteries received a second full capacity discharge during this period.

Three distinct discharges profiles were found to exist. Figure 2.7-39 depicts the discharge characteristic of PCG 1 and 4 batteries, while Figure 2.7-40 shows the characteristic of the remainder, with the exception of PCG 6.

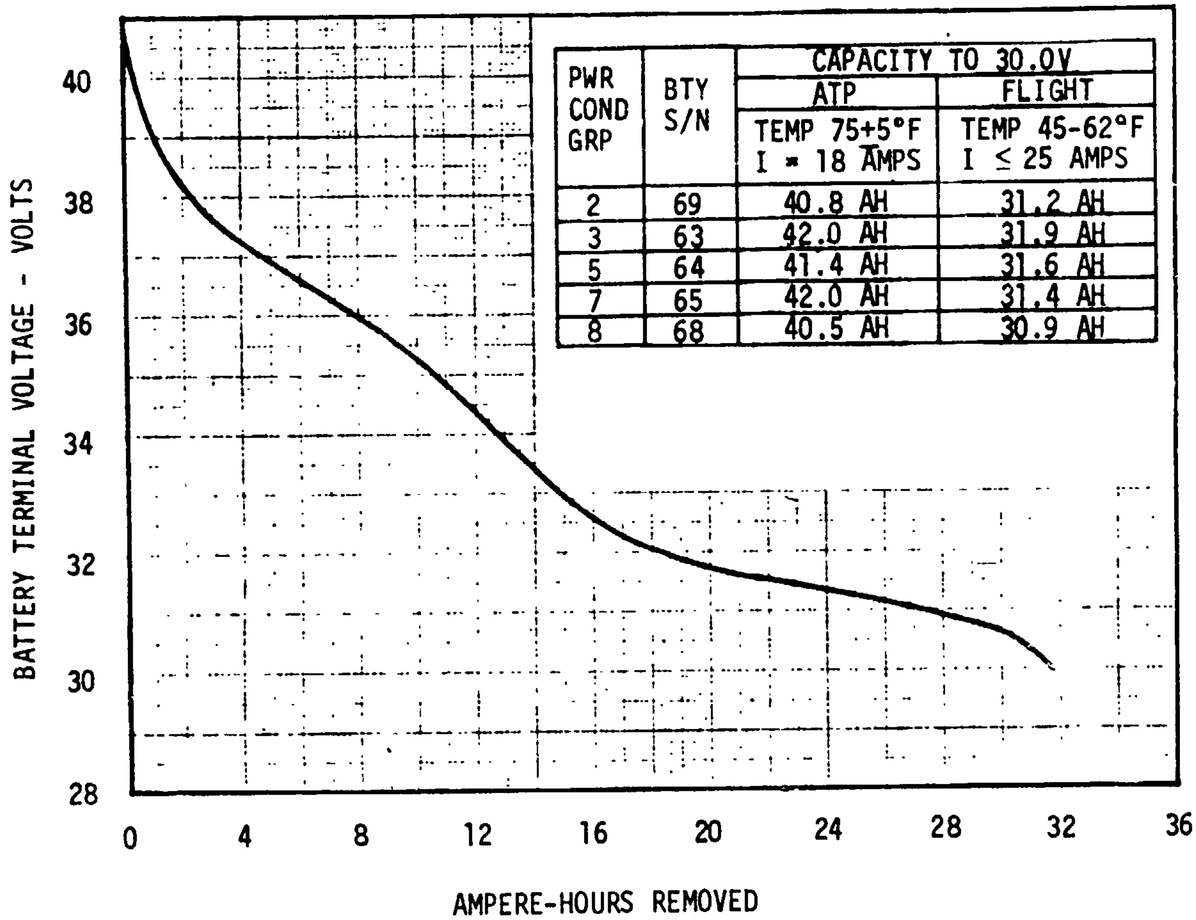
PCG 6 battery, which was discharged to 30.0 volts shortly before the crew departed, exhibited discharge characteristics as shown in Figure 2.7-38. When comparing Figure 2.7-39, Figure 2.7-40, and previous ground test experience on units with similar history, a marked consistency was noted except for the duration of the second voltage plateau which begins at about 16 amp-hours. The second voltage plateau for PCG 1 and 4 batteries, was longer, and resulted in greater amp-hour capacity. One possible contributing factor to this difference is the length of time the various batteries were in the vehicle before launch. PCG 1 and 4 batteries were in the vehicle 22 days prior to launch while the rest were installed sixty-eight days before launch. The apparently lower degradation rate for #1 and #4 batteries may indicate that extended storage in a fully charged condition, or the method of keeping NiCad batteries in such condition, ultimately affects capacity retention. Inadequate information precludes a definite conclusion.

Comparing profiles of Figure 2.7-40 and the 3736 cycle profile of Figure 2.7-38 indicated that PCG 6 battery had a slightly better performance characteristic than other batteries of similar history. As was mentioned earlier, incomplete capacity discharges do not effect the onset of "memory" appreciably. The differences noted here are small and are felt to be the result of periodic partial discharges of PCG 6 battery during the Skylab mission.



PWR COND GRP	BTY S/N	CAPACITY TO 30.0V	
		ATP	FLIGHT
		TEMP 75±5°F I = 18 AMP	TEMP 45-62°F I ≤ 25 AMPS
1	75	42.9 AH	38.7 AH
4	74	41.1 AH	38.2 AH

FIGURE 2.7-39 TYPICAL 3800 CYCLE DISCHARGE PROFILE FOR INDICATED BATTERIES



PWR COND GRP	BTY S/N	CAPACITY TO 30.0V ATP TEMP 75+5°F I = 18 AMPS	CAPACITY TO 30.0V FLIGHT TEMP 45-62°F I ≤ 25 AMPS
2	69	40.8 AH	31.2 AH
3	63	42.0 AH	31.9 AH
5	64	41.4 AH	31.6 AH
7	65	42.0 AH	31.4 AH
8	68	40.5 AH	30.9 AH

FIGURE 2.7-40 TYPICAL 3800 CYCLE DISCHARGE PROFILE FOR INDICATED BATTERIES

The more pronounced effect of full capacity discharges on the subsequent discharge profiles can be seen in Figure 2.7-38 by comparing the 3736 cycle to the 3797 cycle and finally to the 3803 cycle. This same phenomenon is present in PCG 8 battery's end-of-mission capacity data and in AM ground test experience with life cycle batteries.

2.7.4.4 Voltage Regulators

A. SL-1 Launch Through OWS SAS Deployment - The eight AM voltage regulators operated satisfactorily during all periods of operation from launch through full deployment of solar array wing #1. No abnormal conditions were encountered by the voltage regulators during this time span other than the absence of normal operational usage.

B. OWS SAS Deployment Through End of SL-2 - The AM voltage regulators operated satisfactorily through the end of SL-2. Regulated bus voltages were maintained for all input voltage levels, all bus loads, and all Reg adjust potentiometer settings.

(1) Bus Voltage Regulation - During a typical orbit the voltage regulator conditioned power from both the battery and the battery charger. As a result, its input voltage varied in the range from 38 to 46 volts. As shown in Figure 2.7-41, the regulated Reg bus

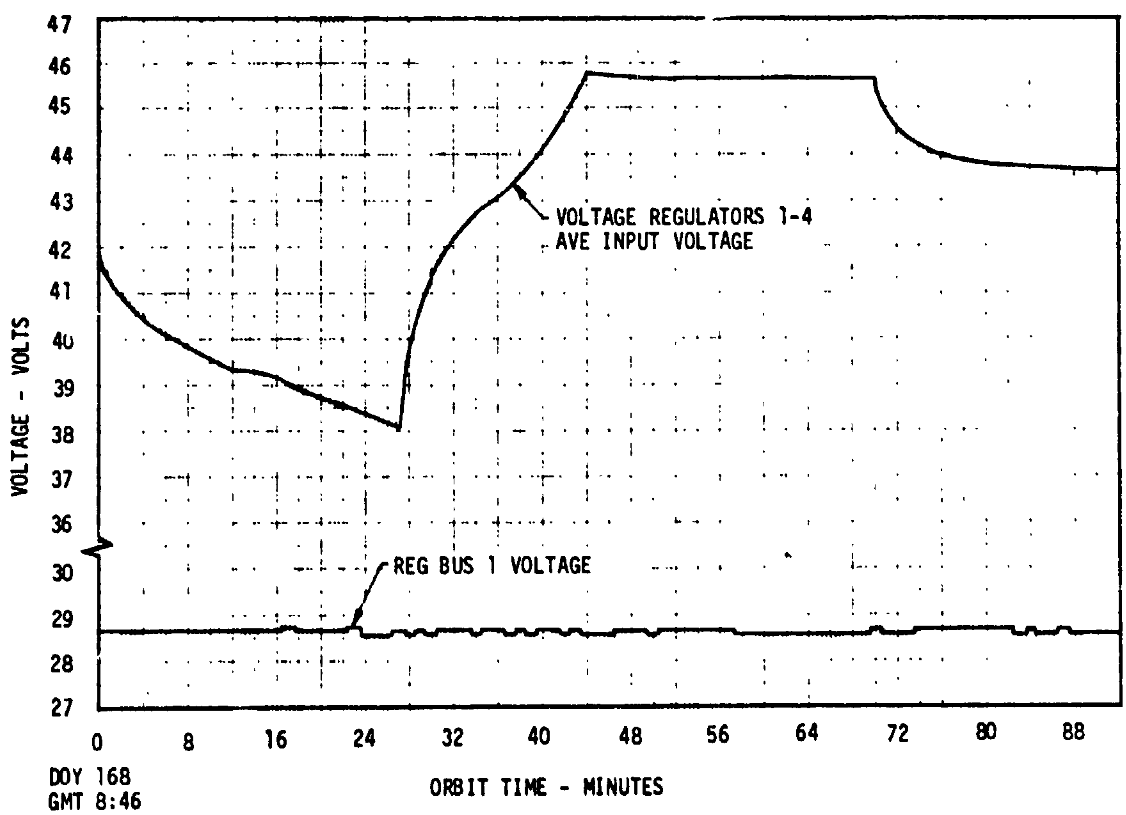


FIGURE 2.7-41 TYPICAL VOLTAGE REGULATOR INPUT AND OUTPUT VOLTAGES

voltage was not affected by the relatively large variance in input voltage. The small fluctuations in Reg bus voltage seen are attributable to bus load variations and/or telemetry data conversion accuracy limitations.

(2) V-I Output Characteristic - With four voltage regulators operating on one Reg bus, the bus voltage was expected to decrease from open circuit voltage (OCV) by 0.01 volt per ampere of load. Figure 2.7-42 shows the relationship of telemetry data points to predicted V-I curves. The curves are based on a 0.01 volt per ampere droop and Reg bus 2 OCV settings of 29.22 and 29.45 volts. These values closely approximate the desired settings for DOY 164 and DOY 170, respectively. Considering the accuracy limitations on the telemetry data, the data points compare favorably with the predicted curves and the comparison is typical for both Reg buses. The Reg bus potentiometers were adjusted several times during SL-1/SL-2 for the purpose of adjusting the AM load level or the AM/ATM load sharing. No adjustments were required because of voltage regulation drift or instability. The potentiometer adjustments were made over nearly the entire adjustment range from almost fully CCW to a 29.5 OCV setting. Computer programs which simulated the normal AM/ATM distribution system were used to calculate the amount of adjustment to be made. Each adjustment resulted in Reg bus voltages and AM/ATM load sharing which compared favorably with those predicted by the computer programs. Based on the foregoing considerations, it was concluded that the eight voltage regulators properly regulated their V-I output characteristic over a wide operating range.

(3) Efficiency - For the operating conditions encountered through the end of SL-2, the voltage regulator efficiency was expected to be better than 93%. There is no evidence that the voltage regulators did not operate at this high efficiency.

Voltage regulator temperatures in the range from 40°F to 60°F were recorded by telemetry throughout SL-2. Temperatures in this range indicated normal operation with no over-heating or efficiency problems.

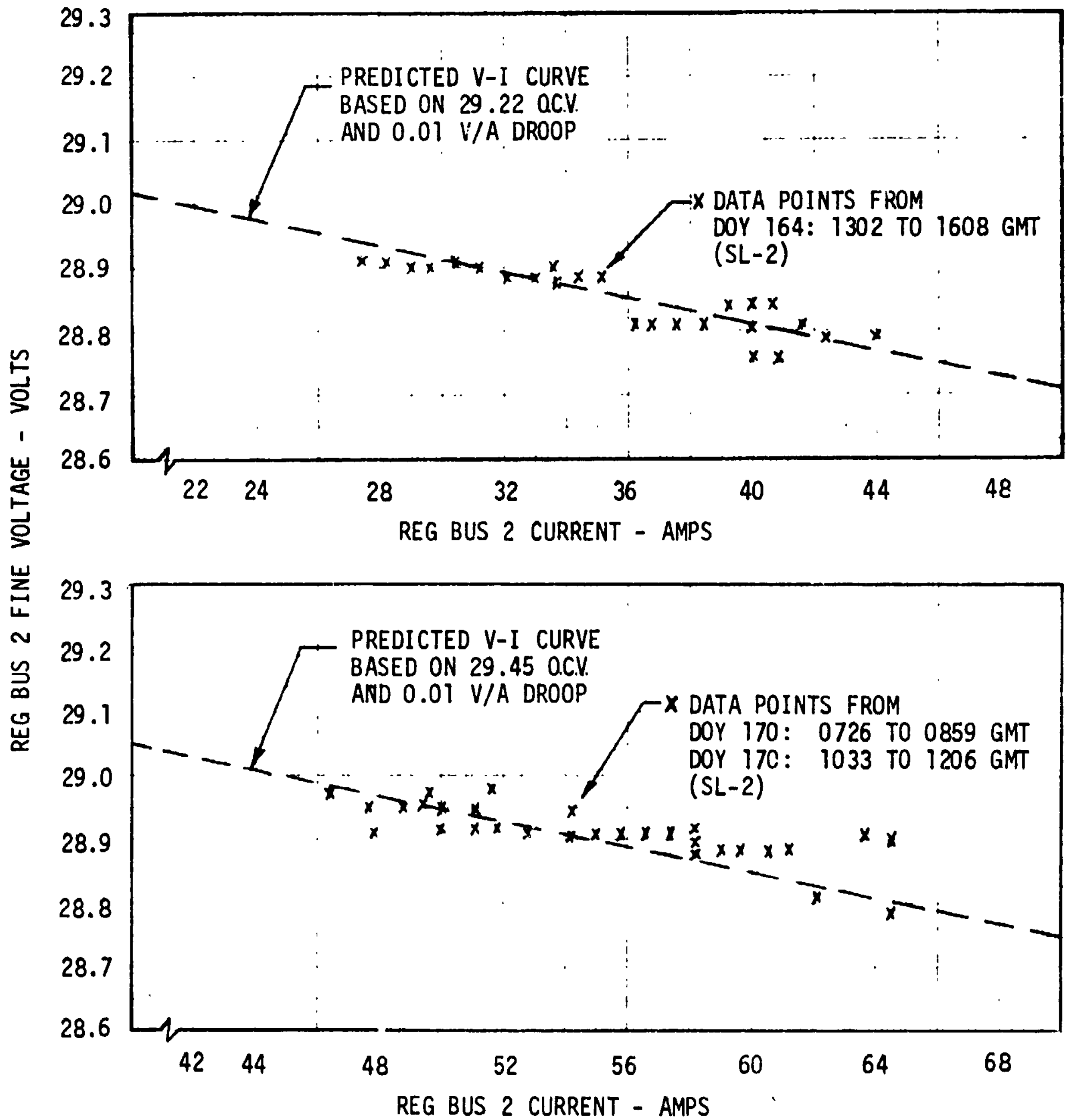


FIGURE 2.7-42 AM BUS REGULATION CURVES (TYPICAL)

(4) Power Module Operation - The AM Voltage Regulator contained five power modules which were redundant to meet the high reliability requirements. Each module operated successively as the output current demand was increased by a 13-ampere increment. During DOY 170, the Reg Bus load was great enough, approximately 15 amperes per PCG, that two power modules in each regulator were required to operate. The fact that the discharge currents for batteries associated with the same Reg Bus remained nearly equal and that no adjustment of the Fine Adjust potentiometers was required throughout SL-2 indicates that these two power modules in each voltage regulator operated satisfactorily.

C. SL-2 to SL-3 Storage Period - The AM voltage regulators operated satisfactorily through the SL-2 to SL-3 storage period. Regulated bus voltages were maintained for all input voltage levels and all bus loads. Regulator telemetry temperatures indicated no temperature or efficiency problems. Battery discharge currents indicated continued proper load sharing between regulators.

D. SL-3 Manned Phase - The eight AM voltage regulators operated satisfactorily throughout the SL-3 mission. Analysis of flight telemetry data indicated normal operation by all eight regulators with no indication of failure or operational anomalies. The required reg bus voltages were maintained for all input voltage levels, all bus loads, and all Reg Adjust potentiometer settings. Observation of all eight AM regulator temperatures throughout the SL-3 mission indicated no overheating or efficiency problems. The curves of Figures 2.7-41 and 2.7-42 are also representative of the observed SL-3 voltage regulator performance.

During most of SL-3, the AM load was such as to exercise only the first module of each voltage regulator. However, there were several instances, such as DOY 261 in which the load was sufficient to exceed 13 amperes per regulator and which required the operation of the second module in each regulator. Observation of the battery discharge current indicated proper operation of the first two modules at these times. An apparent short on the ATM TV Bus 2 on DOY 216 at 0320:21 GMT resulted in a load of greater than 200 amperes on Reg Bus 2. This meant that four and possibly five of the modules in each voltage regulator operated for a period of approximately three seconds.

E. SL-3 to SL-4 Storage Period - The AM voltage regulators operated normally through the SL-3 to SL-4 storage period. The bus voltages were adjusted approximately two volts below their normal settings per the modified SL-3 AM EPS shutdown procedure. These regulated bus voltages were maintained for all input voltage levels and all bus loads. The regulator temperature telemetry parameters indicated no temperature or efficiency problems. Battery discharge currents indicated continued proper load sharing between regulators.

F. SL-4 Manned Phase - The eight AM voltage regulators operated satisfactorily throughout the SL-4 mission. Analysis of flight telemetry data indicated normal operation by all eight regulators with no indication of failure or operational anomalies. The required Reg bus voltages were maintained for all input voltage levels, all bus loads, and all Reg adjust potentiometer settings. Observation of all eight AM regulator temperatures throughout the SL-4 mission indicated no overheating or efficiency problems. The curves of Figures 2.7-41 and 2.7-42 are also representative of the observed SL-4 voltage regulator performance. The Shunt regulator, discussed in paragraph 2.7.2.3(E), was incorporated into the Airlock design as protection against a particular voltage regulator failure mode. The AM voltage regulators were failure free throughout the Skylab mission and operation of the Shunt Regulator was never required.

2.7.4.5 PCG Controls and Monitors

The PCG controls were located on STS Panel 205 and the on-board monitors were located on STS Panel 206.

A. SL-1 Launch Through OWS SAS Deployment - Control usage during this period was by both DCS commands and crew switch actions. The low solar array power available to the PCG's was the reason for the control switching that was performed. The solar array output switches were cycled between their normal and alternate PCG's several times. This was done as a means of increasing power to a single PCG so its battery could be charged, and as a safety measure to preclude low power inputs to PCG equipments. The battery switches were used to turn the batteries off and on as required to charge when possible and preclude discharging the rest of time. The batteries were also turned on several times so the PCG's could act as backup for the ATM EPS. The charger switches were cycled in conjunction with the solar array output switches for analysis purposes and to protect the battery chargers from low solar array power operation. The PCG output switches were cycled off and on when the PCG's were acting as backup for the ATM EPS. The discharge limit switches were placed in their inhibit positions on DOY 158 and returned to auto on DOY 159. This was done as part of the OWS solar array wing deployment activities so the PCG's could supply power, if necessary, even if the battery SOC's went below 30%. All PCG telemetry signals and on-board displays provided sufficient parameter information for operation and analysis throughout this period.

B. OWS SAS Deployment through End of SL-2 - After the deployment of the OWS solar array wing, the PCG controls were used to return the PCG's to their normal configuration. No subsequent control operations were required during this mission phase. All monitors provided satisfactory information with the exception of the SAS #4 current monitor. The problem associated with the SAS #4 current monitor is discussed in detail in section 2.7.4-7-A. A work-around method was developed which allowed satisfactory evaluation of all parameters despite this problem. The SL-2 crew debriefing indicated the satisfactory design and operation of the on-board PCG displays.

C. SL-2 to SL-3 Storage Period - No PCG control operations were required during this period. All monitors performed satisfactorily.

D. SL-3 Manned Phase - All required PCG control switching during SL-3 was accomplished successfully. All PCG telemetry and on-board monitors provided satisfactory and sufficient parameter information for operation and analysis throughout the SL-3 manned mission. The SAS #4 current monitor anomaly, described for SL-2, remained the same throughout SL-3. Most of the PCG control switching was associated with the capacity discharge testing of PCG batteries 6 and 8 on DOY 238 and DOY 239 respectively. The Discharge Limit command for PCG #3 was also used several times during this period in conjunction with EREP passes. The Status Light switches and the Battery Charge selector switch (associated with the % SOC meter) were also used successfully by the crew for periodic status checks on the AM EPS power system.

The first usage of any of the eight fine adjustment potentiometers occurred during the SL-3 mission. Optimization became desirable during SL-3 because EREP passes were scheduled at the rate of one to two per day over an extended period toward the end of the SL-3 mission. This high EREP activity period also occurred over a period of low Beta angle conditions where both EPS systems, AM and ATM, had their minimum power capabilities.

Analysis of flight data showed that battery characteristics were very similar for the eight batteries. Therefore, the pot adjustments were made only to balance out the effects of array shadowing. The ATM array shadowed one module each on SAG #5 and #8 and two modules on SAG #6 (out of 15 modules which made up the SAG for each PCG). The effects of the module shadowing was that PCG's #5, 6 and 8 received less solar array power and could not recover from a DOD equal to the other 5 PCG's in the same amount of charge time. Based on the SI power capability definition, therefore, PCG #6 limited the power capability to the battery DOD it could recover from and none of the other PCG's could be operating at full capability. The amount of adjustment for the fine adjustment pots was determined by using flight data and computer simulation programs. Pot #7 was not adjusted because PCG #7 was sharing equally with PCG's #1 through #4 and had the equivalent solar array input.

Pots #5 and #8 were adjusted to cause their PCG's to supply 0.5 amperes of battery discharge current less than PCG #7 to compensate for one shadowed module and pot #6 was adjusted so that PCG #6 would supply 1.0 ampere of battery discharge current less than PCG #7 to compensate for two shadowed modules. To maintain this configuration as the two Reg bus pots were adjusted for AM/ATM load sharing at subsequent times, it was only necessary to adjust Reg bus pot #2 so that PCG #7 discharge current remained equal to PCG's #1 through #4.

As a result of the adjustments described above, all batteries returned to a fully charged state (100% SOC) at very close to the same time in a daylight period. Therefore, no one PCG, despite differences in available input power, limited the SI power capability significantly different from any other PCG. A more optimum power capability was achieved by use of the fine adjustment potentiometers.

E. SL-3 to SL-4 Storage Period - No PCG control operations were required during this period. All monitors performed satisfactorily with the exception of telemetry parameter M137, battery #5 voltage, which shifted slightly higher on DOY 298 through the end of this period.

F. SL-4 Manned Phase - All required PCG control switching during SL-4 was accomplished successfully. All PCG on-board monitors performed satisfactorily throughout SL-4. The PCG telemetry monitors performed satisfactorily with the exception of the SAS #1 current, battery #1 through #8 coarse currents, battery #1 temperature, and EPS Control Bus 1 and 2 current monitors. A T/M discrepancy caused erratic performance on these parameters from DOY 349 through the end of the mission. The SAS #4 current telemetry monitor anomaly, described for SL-2, remained the same through the end of the mission. Fine adjustment potentiometer #7 was adjusted CCW slightly to equalize the battery #7 discharge current with that of battery #5 and battery #8. End of mission battery testing required the operation of many relay circuits which had seen little prior use. No problems were experienced as a result of this activity which followed a long period of dormancy.

2.7.4.6 Power Distribution System

A. SL-1/SL-2 Mission Phase - All elements of the AM Power Distribution System functioned properly during the SL-1/SL-2 mission phase. All required switching operations were successfully accomplished. Power transfer and load sharing between EPS systems was accomplished as required with no limitations imposed by the Power Distribution System. No problems resulting in protective device operation were encountered during this period. The four major elements of the Power Distribution System are further discussed in the following paragraphs.

(1) Switching - The AM Power Distribution controls performed successfully for the following operations during SL-1/SL-2.

- Activation of Sequential buses, and activation and deactivation of Deploy buses in response to OWS-IU commands. These were one-time operations during the sequential portion of SL-1.

- Closing of Reg/Transfer bus ties in response to AM DCS commands. This operation was performed during SL-1 to parallel the AM and ATM electrical power systems for the first time in flight.
- Deactivation of sequential buses in response to manual control switching by crew. This was a one-time operation during SL-2 only.
- Changing electrical single point ground connection from AM to CSM, and from CSM to AM in response to crew manual switching of the Elec Gnd control. The change from AM SPG to CSM VGP was accomplished after CSM docking and umbilical connection and during SL-2 activation. The change back to AM SPG was accomplished during SL-2 deactivation prior to CSM undocking.
- Activation and deactivation of EREP buses in response to manual control switches located in the MDA. These switching operations were performed throughout SL-2 in conjunction with all EREP periods of operation.

(2) Protection - The Airlock Power Distribution System utilized parallel circuit breakers on the power transfer feeder wires from Airlock to CSM; between Airlock and ATM, and from Airlock to OWS. There were also two circuit breakers connecting the Reg buses together. There was only one unscheduled opening of any of these circuit breakers during SL-1/SL-2. Feeder circuit breaker 2 for OWS bus 1 was opened by an inadvertent crew action but was reclosed without any problem. Scheduled operations of Transfer/CSM feeder circuit breakers and Reg Bus tie circuit breakers were successfully accomplished. These operations were in conjunction with the procedure for paralleling and unparalleling the CSM power system and the AM/ATM combined cluster power system.

Other protective devices utilized included: circuit breakers for transfer current monitors and for power distribution controls; fuses for voltmeter circuits and Reg Bus adjustment circuits; and fusistors (fuse-resistors) in telemetry signal lines for Airlock Bus parameters. There were no unscheduled operations of any of these circuit protective devices during this mission phase. Scheduled operations of the circuit breakers for the power distribution controls during activation and deactivation periods were successful in all cases.

(3) Power Transfer - Prior to Solar Array Wing #1 deployment, the Airlock Power Conditioning Groups were unable to supply power to the AM Reg Buses because of the absence of solar array power. The Airlock power distribution system was used successfully during this period to receive and distribute ATM electrical power for all cluster loads. The Airlock EPS Control Buses were kept powered by closing selected PCG output controls to allow ATM power to each of them by way of the AM Reg Buses. Power transfer during this period was as high as 2700 watts from the ATM Buses to the AM Reg Buses. This power transfer capability contributed to the successful continuation of the SL-1/SL-2 mission until Solar Array Wing #1 could be deployed.

After Solar Array Wing #1 deployment, the AM and the ATM power systems were successfully operated in parallel and controlled throughout the SL-1/SL-2 mission to share the total cluster load. Actual power transfer values during this period were as high as: 2150 watts from the AM transfer Buses to the CSM Buses; 450 watts from the AM Transfer buses to the ATM Buses; and 1050 watts from the ATM Buses to the AM Transfer Buses.

(4) Load Sharing - Load sharing between the AM and ATM electrical power systems was controlled by the Reg Adjust Bus 1 and Bus 2 potentiometers. These potentiometers were adjusted a number of times throughout the SL-2 mission and in all cases functioned as expected to achieve the desired AM and ATM EPS load levels.

Prior to launch, both Reg Adjust potentiometers were set for an actual open circuit voltage (OCV) of 29.3V on the Reg buses. This setting was the calculated setting for the desired AM/ATM load sharing when the two systems would be paralleled by DCS commands during the SL-1 mission phase. In-flight adjustments were also referenced to OCV settings by taking the sum of the Reg bus voltage, and the PCG total current times the Reg bus voltage droop (0.01

volts per amp) as being the approximate OCV value. There were no known inadvertent operations of the Reg bus potentiometers during SL-2.

B. SL-2 to SL-3 Storage Period - Power Distribution System operations were minimal during this mission phase. No power distribution switching was accomplished and no protective devices operated during this period. Power transfer was successfully accomplished from the AM EPS to the ATM EPS during this period and the maximum transferred power value was 500 watts.

C. SL-3 Manned Phase - All elements of the AM Power Distribution System continued to function properly during this mission phase. All required switching operations were successfully accomplished. Power transfer and load sharing between EPS systems were accomplished as required with no limitations being imposed by the Power Distribution System. No problems resulting in AM EPS protective device operation were encountered.

(1) Power Transfer - Power transfer values during this mission phase were as high as 2250 watts from the AM transfer buses to the CSM buses; 350 watts from the AM transfer buses to the ATM buses; and 1550 watts from the ATM buses to the AM transfer buses. These values do not include the contingency condition which occurred on DOY 216 (SL-3 mission day 8). On this day, a short apparently occurred on the ATM load bus 2. The power provided to this short by the combined AM/ATM power systems was sufficient to clear the apparent short in approximately three seconds. Analysis of the conditions during such a limited time span was difficult and did not result in highly accurate values. The analysis, however, did indicate that approximately 9,000 watts was transferred from the AM Reg buses through the AM transfer buses to the ATM buses.

(2) Load Sharing - The Reg Adjust Bus 1 and Bus 2 potentiometers were adjusted a number of times throughout the SL-3 mission and in all cases they functioned as expected to achieve the AM and ATM EPS load levels.

During the SL-3 mission one inadvertent adjustment of a Reg Adjust potentiometer occurred when on DOY 242 an astronaut's pant cuff apparently caught on the Bus 2 potentiometer knob and resulted in a CCW rotation which increased Reg Bus 1 current to 61.3 amperes and decreased Reg Bus 2 current to 18.7 amperes. The system imbalance was quickly corrected by adjusting the Bus 2 potentiometer CW for equal PCG total currents.

D. SL-3 to SL-4 Storage Period - Power Distribution System operations were minimal during this mission phase. No power distribution switching was accomplished and no protective devices operated during this period. The Reg/Transfer Tie relays remained open throughout this period so no power was transferred between the AM and ATM electrical power system.

E. SL-4 Manned Phase - All elements of the AM Power Distribution System continued to function properly during SL-4. All required switching operations were successfully accomplished. Power transfer and load sharing between EPS systems were accomplished as required with no limitations being imposed by the Power Distribution System. No problems resulting in AM EPS protective device operation were encountered. The Reg Adjust Bus 1 and Bus 2 potentiometers were adjusted a number of times throughout the SL-4 mission, primarily in conjunction with EREP and Kohoutek passes. In all cases they functioned as expected to achieve the desired AM and ATM EPS load levels.

2.7.4.7 Anomalies - SAS #4 Current

One anomaly associated with the AM EPS occurred during the SL-1/SL-2 mission. The SAS #4 current monitors, on-board and telemetry, indicated a SAS #4 current consistently lower than the other SAS readings. This condition was discovered shortly after the AM EPS became operational with the full deployment of all SAS Wing #1 sections on DOY 159. SAS #4 current readings at this time indicated approximately 3 amps below the readings for SAS's #1, #2 and #3. Other parameter readings in PCG #4, however, indicated that PCG #4 was receiving the same amount of SAS power as the other PCG's. The initial battery charge current readings in particular indicated normal operation of PCG #4. Analysis of input and output power values in PCG #4 also indicated that the PCG was receiving a SAS input comparable to the other PCG's. At this time it was decided that only a measurement problem existed, which would continue to be monitored, but no action was required and, that there were no system or mission effects.

A major change in this condition was noted when the CSM fuel cells were shut down and the CSM became powered by the AM/ATM EPS through the Transfer buses. At this time the SAS #4 current readings started indicating approximately 6 amps below the other SAS current readings. At about this same time, a postulation was made that there might be a return path through structure causing this anomaly. One of the reasons for this postulation was that analysis also indicated that the Reg Bus 1 current reading might also be low. This was arrived at by adding all the source currents and load currents at the Reg buses and comparing them. From this comparison, it appeared that the source currents were low by a few amps in most instances. At first, the mission amps value was very small compared to the multiple amperage values being added and subtracted and so no firm conclusion could be reached. However, after the current levels increased to supply the CSM, the value by which Reg Bus 1 current was low became large enough to evaluate and it showed reasonable comparison to the amount that SAS #4 current was indicated to be low. Further analysis of available data, along with the known damage at SAS wing #2, led to the firm conclusion that a structural return path through the PCG #4 SAS return input on wing #2 did exist.

Figure 2.7-43 shows the block diagram used for an analysis of this anomaly. The return current to voltage regulator #4 can be considered to take two paths from the AM Shunt Tie Bar to the voltage regulator minus. The design path is identified as path #1 on the diagram and path #2 is the structural path caused by the damaged wiring at wing #2. A circulating current is set up in path #2 by the IR drops across the wiring resistances as represented on the diagram in the various return paths for the PCG. It can be seen from the diagram that the current through the structure path is not included in the Reg Bus #1 current reading as it should be. The diagram also shows that the same current flows through the SAS #4 current shunt in opposition to the actual SAS #4 return current. Therefore, the SAS #4 current shunt measures the resultant current and the reading is low by the amount of the structure current.

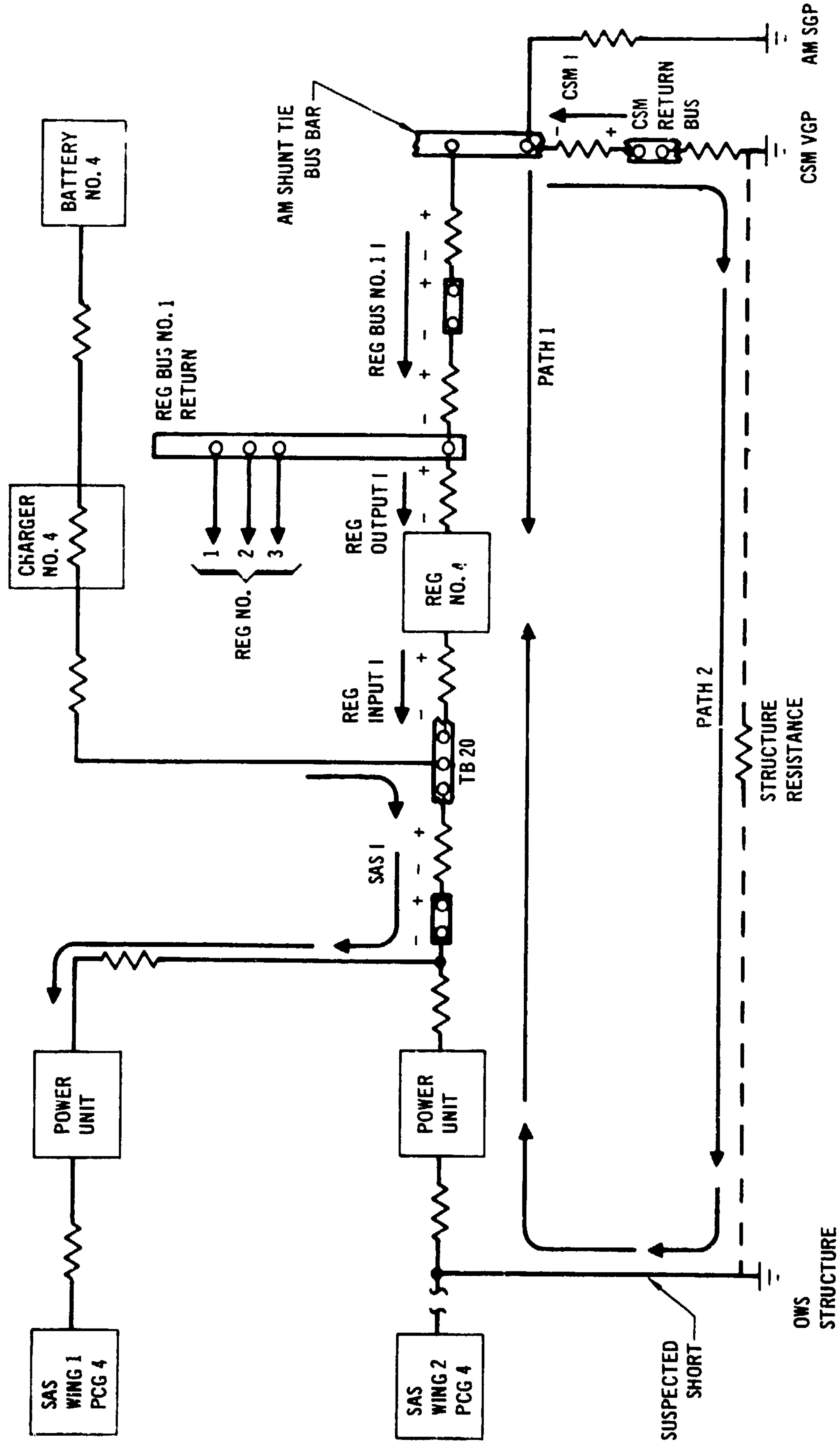


FIGURE 2.7-43 SAS NO. 4 CURRENT PATHS

Further verification of this analysis was obtained at the end of SL-2 when the CSM was switched back to internal power and the structural return was switched from the CSM VGP to the AM SPG. The SAS #4 current readings became only approximately 1.25 amps low as would be expected for the lower current levels. The analysis using the AM SPG also showed reasonable agreement with the actual measured values.

This anomaly had no significant system or mission performance effect. Procedures for evaluating SAS #4 current and Reg Bus #1 current readings were set up and used through the remainder of the SKYLAB mission. SAS #4 current was calculated as equal to the average of the SAS #1, #2 and #3 current. Reg Bus #1 current readings were assumed to be low by the difference between the SAS #4 current reading and the calculated value obtained as described above.

2.7.4.8 Special Tests

A. SL-1/SL-2: Battery Storage Test - It was established shortly after the SL-1 Launch on DOY 133 that the AM batteries would probably be stored for some indeterminate period of time. This was because of the absence of solar array input power. The battery storage conditions would be different than previously experienced because of the partially charged status (approximately 60% SOC) and the lower battery temperature (approximately 40°F). On DOY 138 it was decided to place the eight AM batteries on the Skylab Cluster Power System breadboard at MSFC in the same conditions as the flight batteries. The idea was to be able to test alternative recharge procedures on the breadboard prior to the selection and use of a procedure for the flight batteries when solar array power became available.

The breadboard test was started on DOY 138 but a facility power failure occurred on DOY 140 and the test had to be restarted on DOY 140. Each battery was discharged to the same SOC as its corresponding flight battery. The breadboard coolant loop was held at an inlet temperature of 39 $\pm$ 2°F. A set of open circuit voltage readings was taken on the batteries each day at a set time.

Analysis indicated that if the solar arrays were successfully deployed on DOY 158 there would be no need for any special battery recharge procedure. Therefore, on DOY 157 battery #4 on the breadboard was recharged per normal flight procedures. Data taken on the recharge indicated no problems and showed normal recharge characteristics. Subsequent to successful array deployment on DOY 158, all breadboard batteries were returned to 100% SOC and the breadboard was returned to normal operation.

B. SL-3: AM EPS Shutdown Procedures Test - The primary AM coolant loop was shut down during SL-3 because of a coolant fluid leak. The secondary AM coolant loop then became the only active loop. Plans were developed to reservice the primary loop at the beginning of the SL-4 manned mission. However, this left only one coolant loop operational during the SL-3 to SL-4 storage period with no back-up system. This led to an investigation of the effect of a contingency loss of all coolant flow on the AM EPS. It was determined that with only the DCS commands available it would not be possible to reconfigure the AM EPS into a shutdown configuration which would protect all AM EPS equipments from encountering thermal damage. This was based on SL-3 deactivation procedures being the same as for SL-2. It was then desired to investigate alternative procedures which could provide complete protection for AM EPS equipments in the event of such a contingency. These procedures would also have to have no adverse effect on cluster power system operations if no such contingency occurred.

Checkout of the various procedures which were proposed was done on the Skylab Cluster Power System breadboard at MSFC. This breadboard testing accomplished several goals. It allowed actual comparison of procedures as to their complexity and effectiveness on a flight equivalent cluster power system. It also established the validity and completeness of each procedure involved. The effects of other possible contingencies on the procedures were evaluated along with the effects of load levels different from the predicted mission load profiles.

The selected procedure included four flight mission procedures; a crew procedure during SL-3 deactivation; a DCS command procedure during the storage period; a DCS command procedure for SL-4; and a crew procedure for SL-4 activation. Each of these was verified by operation on the breadboard. The crew SL-3 deactivation procedure included adjusting the Reg bus pots and opening the Reg/transfer tie relays thus isolating the AM and ATM power systems. Breadboard operation evaluated the amount of pot adjustment to maintain suitable AM Reg bus voltages before a contingency coolant shutdown, and yet protect AM EPS equipments if the contingency occurred and there were less active CBRM's in the ATM EPS because of other contingencies. The pot adjustment values determined by the breadboard tests were in agreement with values obtained from computer simulations which were also run to support this investigation. Breadboard operation was also used to verify that both contingency operation and subsequent normal mission operation could still be supported if one of the Reg/transfer tie relays should fail to reclose.

C. SL-3: Voltage Regulator Thermal Test - A thermal test of the AM voltage regulator was conducted to investigate whether stabilization temperatures in excess of the redline value of 140°F would result under load as predicted by computer simulations for loss of AM coolant flow. This test program was initiated on 18 September 1973 and was completed on 23 September 1973.

The test specimen was a flight type regulator mounted on a simulated coldplate along with a flight type charger. The simulated coldplate was mounted on a temperature controlled support structure simulating the battery module structure. A simulated meteoroid structure covered the test specimen and the assembly was mounted in a vacuum chamber with a controlled temperature shroud. A solar array simulator was used to provide an input to the voltage regulator (charger bypassed) and a load was connected to the regulator output. Regulator, coldplate, and support structure temperatures were continuously monitored during the test.

Tests were conducted with a simulated solar input for $\beta = 58.5°$, solar inertial attitude condition, and at two load levels; eight watts/regulator and 250 watts/regulator. Voltage regulator temperature was below the redline value of 140°F in both test cases. For the computer simulation of the cases noted above, regulator temperatures approached the redline value for the eight watt/regulator case and were well above the redline value for the 250 watt/regulator case. It was determined that the reason for the variation between computer and test data was due to the simplified representation of background structural temperature used in the test.

The simulated support structure used in test was held at a constant temperature throughout the test, while in fact the battery module structure temperature varied with the temperature of the PCG components due to conduction and radiation of heat to structure. In the test facility, heat was removed from the regulator by the temperature controlled simulated structure. The computer simulation took into account the variation in structure temperature with regulator temperature. In development tests of the complete battery module in a vacuum chamber at MDAC-E, the variation in structure temperature with PCG component temperature was also observed. Adjustment of test results to account for variation in background structural temperature gave regulator temperatures which agreed with the computer results within 2 percent.

D. SL-3: SAS #4 Current Anomaly Test - A test was run using the Skylab Cluster Power System breadboard at MSFC to simulate the postulated cause for the SAS #4 current anomaly and to check the results. The postulated cause was a short from a SAS #4 return wire at OWS solar array wing #2 to vehicle structure. This was simulated at the breadboard by a short from the return wire at the output of the SAS #4 simulator to the cabinet structure housing the SAS simulators. All such structures were returned to simulated vehicle single point ground for the breadboard. Tests were run with and without the simulated short for several load levels and battery conditions as shown in Figure 2.7-44. The tests also were run for the ELEC GND switch on the 206 panel in both the Airlock and CSM positions. The test results were determined by subtracting the current for the short condition from the current for the no-short condition.

Comparisons of the delta currents for SAS #4 versus Reg Bus #1 showed that the two values were very close to the values indicated by analysis. The test results also showed the general trend that the value of the delta currents was affected by load levels and by the specific electrical ground in use. The breadboard simulation cannot be considered high fidelity in the area of electrical grounds and vehicle structure paths depending on which electrical ground is operational. However, the test results did show a difference depending on the electrical ground path which was also observed on flight vehicle data. In general, the test results were considered to substantiate the conclusions of the analysis.

TEST CONDITIONS			SAS #4 I	REG BUS #1 I
5700 W LOAD (BATTERY IN TRICKLE CHARGE)	AIRLOCK GND	NO SHORT	5.1	43.8
		SHORT	- 0.8	- 40.1
		ΔI	4.3	3.7
	CSM GND	NO SHORT	5.0	43.8
		SHORT	-(-1.1)	- 38.1
		ΔI	6.1	5.7
3400 W LOAD (BATTERY DRAWING PEAK POWER)	AIRLOCK GND	NO SHORT	12.6	22.1
		SHORT	- 7.2	- 16.6
		ΔI	5.4	5.5
	CSM GND	NO SHORT	12.6	22.0
		SHORT	- 7.9	- 17.2
		ΔI	4.7	4.8
5700 W LOAD (BATTERY DRAWING PEAK POWER)	AIRLOCK GND	NO SHORT	12.5	59.6
		SHORT	- 5.3	- 52.4
		ΔI	7.2	7.2
	CSM GND	NO SHORT	12.6	59.7
		SHORT	- 3.7	- 50.9
		ΔI	8.9	8.8

TEST PERFORMED ON 9-14-73 (DOY 257)

FIGURE 2.7-44 SIMULATED "SAS NO. 4 RETURN WIRE SHORT" TEST RESULTS

2.7.5 Conclusions and Recommendations

The AM Electrical Power System flight performance was completely successful in satisfying all Skylab requirements. The system design concepts were exercised early in the mission due to an unfortunate launch anomaly. The complete loss of one solar array wing and minimal deployment of the other left the entire AM EPS with practically no source of external energy for several weeks. During this time, the built-in control flexibility was utilized to maintain 3 of 8 batteries fully charged and the remaining batteries approximately 50% charge for possible contingency use, or ready for normal operation when the remaining wing could be fully deployed. Later deployment of the wing with resumption of normal EPS operations (after appropriate management of bus settings, etc.) confirmed in flight the features that had been designed and tested in the ground test programs.

AM EPS Flight Performance Milestones:

- Supplied approximately one-half cluster load despite loss of one-half of the anticipated solar input energy.
- Successfully supported extension of SL-3 and SL-4 missions (with attendant increase in number of EREP passes) with no deleterious effects.
- Successfully supported addition of Kohoutek comet observations.
- Successfully supported higher battery DOD's resulting from additional nonsolar inertial attitude maneuvers to minimize TAC usage following CMG failure.
- Flight mission rules for battery DOD's were relaxed as a result of better than expected flight performance.
- Post SL-4 battery capacity tests revealed cycle life degradation lower than predicted. Battery measured capacity ranged from 31.2 to 38.7 amper - hours as compared to a rated capacity of 33 ampere-hours.

The success of the EPS is attributed to the systematic design concepts, development and demonstration test philosophy, and the operational flexibility. These factors enabled the ground and/or crew to appropriately configure the existing equipment to support the ever changing cluster flight conditions.

Systematic Design:

- Multiple PCG concept provided system redundancy and growth capability to support increased load requirements as the design matured.
- Modularization of chargers and regulators provided internal redundancy of components and improved efficiency.
- State-of-the-art advancements in high power control were incorporated.
- High overall efficiency was achieved by usage of solar array "peak power tracking", and buck regulation throughout.
- All automatic controls were provided with override capability with the exception of charge termination for battery overtemperature. The battery overtemperature trip point was set at a point where continued charging would not result in additional stored energy due to the low charge efficiency. This control did not, however, preclude battery discharge.
- Capability for alternate or bypass operational modes was provided for contingency operation with a minimum loss in system power.
- Individual "fine tuning" of PCG regulators optimized power output by compensating for variations in solar array input power and conditioning component performance.
- Redundant distribution busses precluded major impact of hypothetical bus short circuits.
- Output voltage adjustment capability provided for load sharing control between AM, ATM, and CSM.
- Computer simulation of AM EPS enabled prediction of operation within total cluster power system under various operational configurations and permitted real-time mission planning on a reasonable time scale.

Systematic Testing:

- Early development tests integrated and optimized key hardware features.
- Thorough Qualification and Acceptance testing of all hardware established piece part capabilities.
- Operational test of one PCG including a solar array (SCST), validated major system interfaces.
- Detailed Electrical/Environmental test of one battery module (4 PCG's) and a complete distribution system including interface simulation confirmed that the various designed operational modes performed correctly and efficiently under various cluster operational conditions.

- Battery cycle tests using predicted mission duty profile demonstrated component life.
- Flight procedures and crew training were developed during system testing based on actual hardware experience.

EPS System flexibility enabled:

- Management of EPS during unexpected loss of array deployment.
- Formulation of contingency procedure for safe shutdown and reactivation of EPS in event of loss of coolant system fluid prior to SL-4 reservicing.
- Achievement of many of the flight milestones previously stated.

Upon completion of SL-4, the EPS was configured into a "dormant" mode with the batteries and all switchable busses off. The nonswitchable EPS control bus is powered during periods of sufficient solar array illumination. All 8 PCG's were operating properly at ground monitoring termination, and could have continued to supply baseline performance indefinitely.

Recommendations for improving the AM EPS for a similar space program are limited to minor improvements in the basic program such as:

- Additional parametric studies of the batteries to more fully understand the interrelationships of temperature, DOD, second plateau, discharge rates, etc.
- Additional studies of the battery/charger charge scheme for possible efficiency improvement.
- Addition of a 100 percent reset capability for the ampere-hour meter. This feature would permit resynchronization of the battery SOC and indicated SOC when ground telemetry indicated that the battery was fully charged.

2.8 SEQUENTIAL SYSTEM

The AM Sequential System provided the electrical control for conversion of the two stage Saturn V payload into the Skylab orbital configuration. This was accomplished by jettison of the payload shroud (PS), deployment of the discone antennas, deployment of the ATM, deployment of the ATM and OWS SAS, operation of various vent valves and the control of selected ATM functions. The Skylab launch sequence of events, Figure 2.8-1, was initiated by automatic and/or ground command systems.

The automatic command source for controlling the sequential system was the IU/OWS switch selector system, Figure 2.8-2, which consisted of four major components: launch vehicle digital computer (LVDC), launch vehicle data adapter (LVDA), OWS switch selector and a command and communications system (CCS). The heart of the IU automatic command system was the LVDC, located in the IU. The LVDC had the capability of remotely controlling switch selectors mounted in each stage of the vehicle.

The LVDC automatically issued stored commands at the appropriate times. These commands controlled the switch selectors located in the S-IC stage, S-II stage, IU and OWS. Parallel data bits were issued by the LVDC to the LVDA, which conditioned the data bits and transmitted them to the switch selectors. One group of data bits was considered the address bits. The address bits selected a particular switch selector to receive the command data word. After receipt the switch selector sent the data word complement back to the LVDC for verification. If valid the LVDC sent an execute command. If invalid the LVDC sent the complement which the switch selector interpreted as a valid word and the LVDC followed with an execute command.

The OWS switch selector, as well as all the other switch selectors, had a capability of activating 112 different circuits individually. The switch selector took the validated command data word and activated one channel. This channel normally operated one external relay coil, but in a few special cases two external relay coils were driven by one channel. The external relays in the OWS used either AM, Sequential, or Deploy Bus power from the AM to provide commands to the sequential system. The OWS switch selector output driver circuits were powered by redundant AM bus power. Switch selector performance was indicated by telemetry, i.e., no output, one channel active or more than one channel active.

EVENT	DOY/GMT D: H: M: S	ELAPSED MISSION TIME D: H: M: S
FIRST MOTION	134:17:30:00.28	-00:00:00:00.30
LIFTOFF (ELAPSED TIME ZERO)	134:17:30:00.58	00:00:00:00.00
*MET SHIELD TENS STRAP 2 SEP	134:17:31:02.9	00:00:01:02.32
*MET SHIELD TENS STRAP 1 & 3 SEP	134:17:31:03.0	00:00:01:02.42
*SAS WING 2 BEAM FAIRING SEP	134:17:31:03.0	00:00:01:02.42
*MET SHIELD TEMPS OFF SCALE	134:17:31:30.0	00:00:01:29.42
*MET SHIELD INDICATED PARTIAL DEPLOY	134:17:31:30.0	00:00:01:29.42
VENT OWS H/A	134:17:33:25.27	00:00:03:24.69
MDA VENT VALVES CLOSED	134:17:34:48.35	00:00:04:47.77
START TIME BASE 4	134:17:39:49.20	00:00:09:48:62
S-II/PAYLOAD SEP	134:17:39:51.2	00:00:09:50:62
TACS ACTIVATION	134:17:39:51.95	00:00:09:51.37
ACTIVATE AM SEQUENTIAL BUSES	134:17:39:54.35	00:00:09:53.77
OWS WASTE TANK VENT COMMANDED OPEN	134:17:39:55.00	00:00:09:54.42
JETTISON RS PROTECTIVE SHIELD	134:17:39:56.00	00:00:09:55.42
ORBIT INSERTION	134:17:39:58.00	00:00:09:57.42
*SAS BEAM 2 WING TEMPS WENT TO EXTREMES	134:17:40:00	00:00:09:59.42
ACTIVATE OWS RS	134:17:40:08.00	00:00:10:07.42
START TIME BASE 4A	134:17:45:09.20	00:00:15:08.62
PAYLOAD SHROUD JETTISON	134:17:45:20.99	00:00:15:20.41
ACTIVATE AM DEPLOY BUSES	134:17:45:34.22	00:00:15:33.64
INITIATE ATM DEPLOYMENT	134:17:46:37.0	00:00:16:36.42
DISCONE ANTENNA 2 DEPLOYED	134:17:46:53.09	00:00:16:52.51
DISCONE ANTENNA 1 DEPLOYED	134:17:46:54.79	00:00:16:54.21
ATM DEPLOYED AND LOCKED	134:17:50:15.47	00:00:20:14.89
INITIATE ATM SAS DEPLOY/CANISTER RELEASE	134:17:54:49.00	00:00:24:48.42
OWS SAS BEAM CMDS (IU)	134:18:11:05.90	00:00:41:05.32
TERMINATE OWS H/A VENT	134:18:11:20.00	00:00:41:19.42
OWS SAS WING CMDS (IU)	134:18:22:00.0	00:00:51:59.42
OWS SAS BEAM 1 SEP	134:18:26:00.0	00:00:55:59.42
OWS METEOROID SHIELD CMDS (IU)	134:19:06:04.10	00:01:36:03.52
ACTIVATE ATM APCS	134:19:07:00	00:01:36:59.42
BACKUP OWS SAS BEAM CMDS (AM)	134:19:08:22.0	00:01:38:21.42
BACKUP OWS SAS WING CMDS (AM)	134:19:20:56.0	00:01:50:55.42
PARALLEL ATM/AM BUSES, REG 1-XFER 1 CLOSED	134:19:27:23.0	00:01:57:22.42
REG 2-XFER 2 CLOSED	134:19:27:38.0	00:01:57:37.42
BACKUP METEOROID SHIELD CMDS (AM)	134:20:12:30.0	00:02:42:29.42
AM DEPLOY BUSES SAFE (IU CCS)	134:20:33:56.0	00:03:03:55.42
OWS SOLENOID VENT VALVES OPEN	134:21:05:55.0	00:03:35:54.42
TRANSFER ATTITUDE CONTROL FROM IU TO APCS	134:22:20:05.0	00:04:50:04.42
PNEUMATIC SPHERE DUMP	134:22:52:00	00:05:21:59.42
OWS SOLENOID VENT VALVES CLOSED	135:00:28:43.0	00:06:58:42.42
TERMINATE PNEUMATIC SPHERE DUMP	135:01:45:00	00:08:14:59.42
OWS SWITCH SELECTOR INHIBIT	135:04:00.00	00:10:29:59.42
END OF IU LIFETIME	135:12:16:00	00:18:45:59.42
START (32 MIN 45 SEC) SEVA	145:23:52:15	11:06:22:14
SEQUENTIAL BUSES OFF	146:17:02:00	11:23:32:00
PARASOL DEPLOYED & SECURED	147:01:30:00	12:08:00:00
OWS SAS BEAM 1 DEPLOYED	158:22:50:00	24:05:20:00
OWS WINGS DEPLOYED	159:00:28:30	24:06:58:30
TWIN-POLE SUN SHIELD DEPLOYED	219:00:01:00	84:06:31:00

*SEQUENCE WAS NOT COMMANDED

FIGURE 2.8-1 SL-1 AND SL-2 MAJOR SEQUENTIAL EVENTS

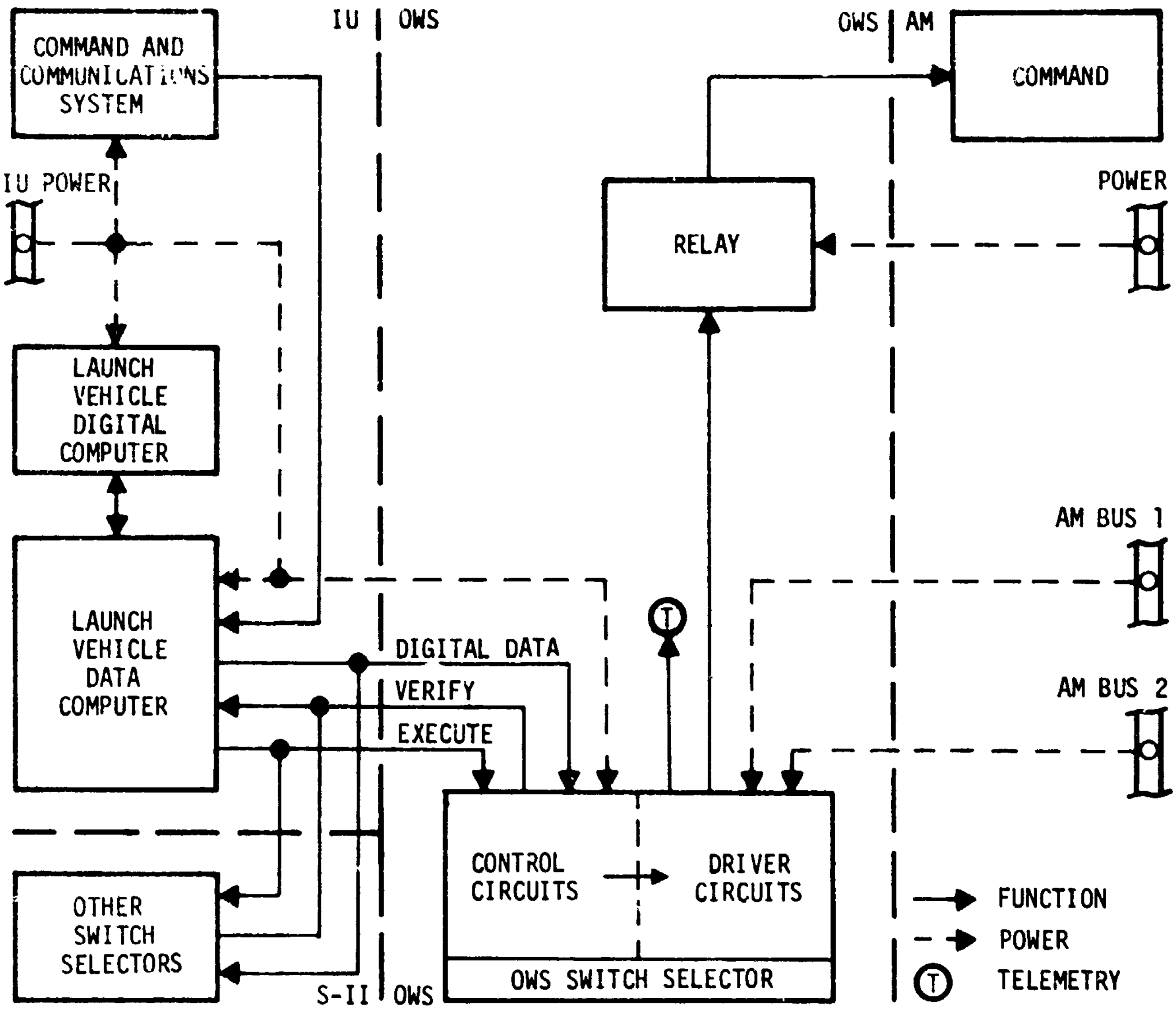


FIGURE 2.8-2 IU/OWS SWITCH SELECTOR SYSTEM

The STDN ground stations could issue backup commands to control the OWS switch selector via the IU CCS and the LVDA. The AM contained a digital command system (DCS), paragraph 2.10, to provide an alternate and/or backup method of controlling activation. The AM DCS was also controlled from STDN ground stations.

The OWS contained in addition to the switch selector, the pneumatic control system (PCS) for OWS venting and refrigeration radiator shield jettison system and the ordnance and firing units for the meteoroid shield and OWS SAS deployment systems. The OWS PCS consisted of a sphere containing high pressure gaseous nitrogen and actuation control modules. The electrically operated modules controlled the flow of nitrogen to open and close habitation area vents, to open waste tank vent and to jettison the refrigeration radiator shield. The ATM contained the ordnance and firing units for ATM SAS deployment system. The AM contained the power and control equipment for the AM Sequential System.

Verification of the design requirements was successfully completed by the test program. During the mission, the problems encountered with the OWS meteoroid shield and OWS SAS deployment required the Skylab crews to erect sunshades and complete deployment of the OWS SAS wing 1. All other sequences functioned as planned.

The sequential system consists of several separate systems related to each other only in that they each occur in the properly timed sequence as determined by the commands from the IU/OWS switch selector system or the AM DCS. For discussion purposes the sequential system will be divided into subsystems as follows:

1. Payload Shroud Jettison
2. ATM Deployment
3. Discone Antenna Deployment
4. Power Control
5. Radiator Shield Jettison/Refrigeration System Activation
6. OWS Venting
7. OWS Meteoroid Shield Deployment
8. OWS SAS Deployment
9. ATM SAS Deployment
10. ATM Activation
11. MDA Venting

Topics which will be covered under each subsystem heading are:

A. Design requirements
B. System description
C. Testing
D. Mission Results
E. Conclusions and Recommendations

2.8.1 Payload Shroud Jettison Subsystem

The electrical sequential system required to jettison the shroud interfaced with the OWS switch selector, the AM Command Relay Driver Unit (CRDU), the AM power system and the payload shroud mechanical/ordnance system. Automatic and JSC flight controller manual operating capabilities were included in the design. One deviation in contractual requirements was requested and granted for the EBW firing unit trigger circuit resistance. All other requirements were met. The hardware utilized in the design was selected from flight qualified equipment.

Detail Payload Shroud information is supplied in MDC Report G4679A.

2.8.1.1 Payload Shroud Jettison Subsystem Design Requirements

The PS jettison circuit design was initiated when the early mechanical/ordnance tradeoff studies were completed. The selected PS configuration was a radially segmented design to be jettisoned in orbit. The integrity of the PS monocoque structure was maintained during ground operations and ascent by pinned upper and lower structural rings. Discrete latch actuator pins (16 required) protruded through the links that held the structural rings together, Figure 2.8-3. Each ring joint had two discrete latch actuators to obtain redundancy in releasing the rings. Initially one EBW firing unit was allotted for each latch actuator, which was to be controlled and powered by the IU via FAS wiring to the PS. Subsequent redesign of the ordnance system resulted in the connection of the four latch actuators in each segment to a common closed tubing manifold system, Figure 2.8-4. Linear explosive contained in the tubing terminated at the detonators. A firing unit at each detonator provided the redundant means of detonating the explosive and reduced the total number of latch actuator firing units to eight. The gas generated by the burning linear explosive caused the latch actuator pins to retract. After the discrete latch actuator pins were retracted, the four shroud segments remained intact by the restraining rivets along the thrusting joints.

The linear explosive contained in the thrusting joint bellows, Figure 2.8-5, expanded after being ignited, shearing the rivets, and propelling the segments away. Each end of the linear explosive terminated in a detonator, Figure 2.8-4.

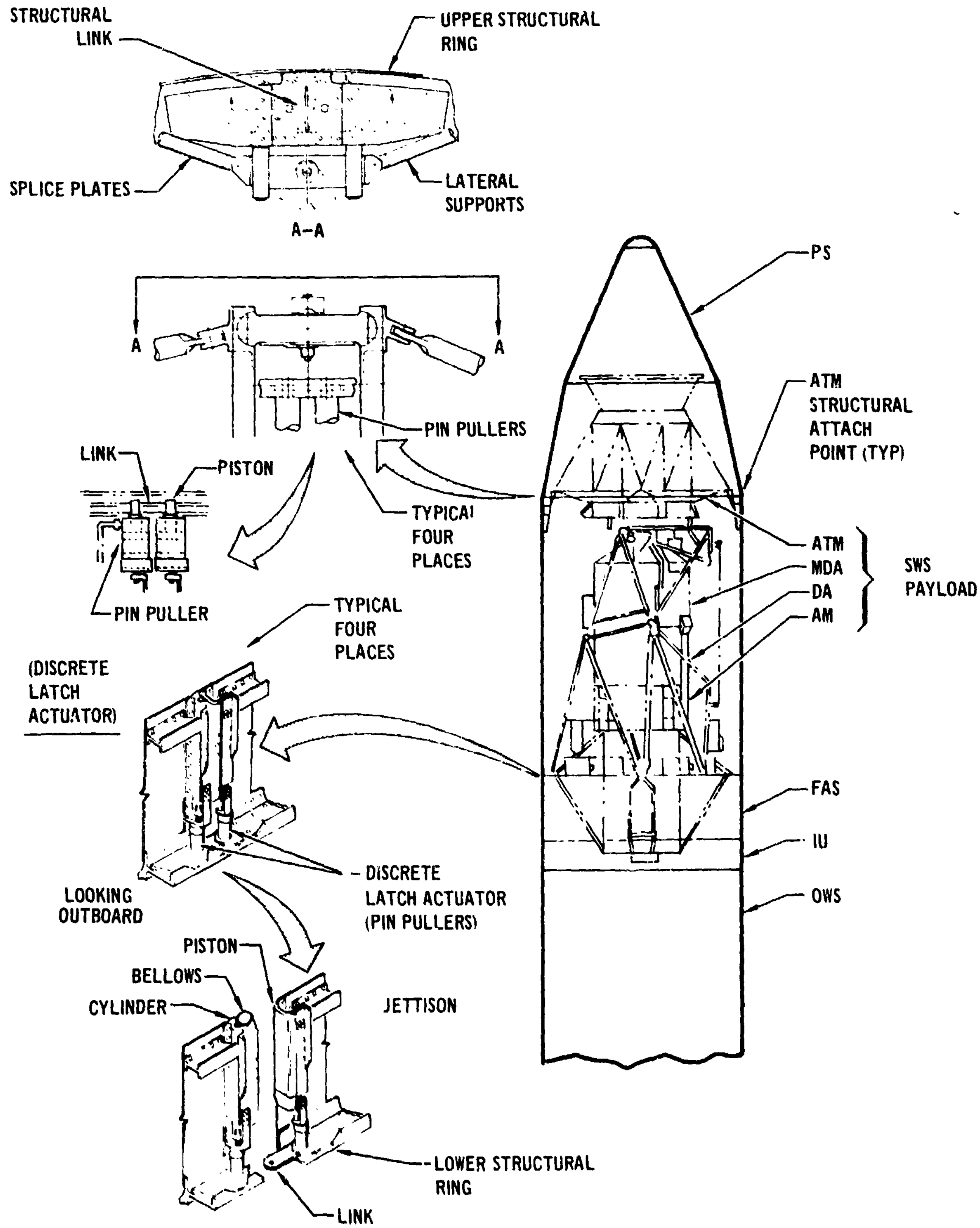


FIGURE 2.8-3 DISCRETE LATCH ACTUATOR SYSTEM

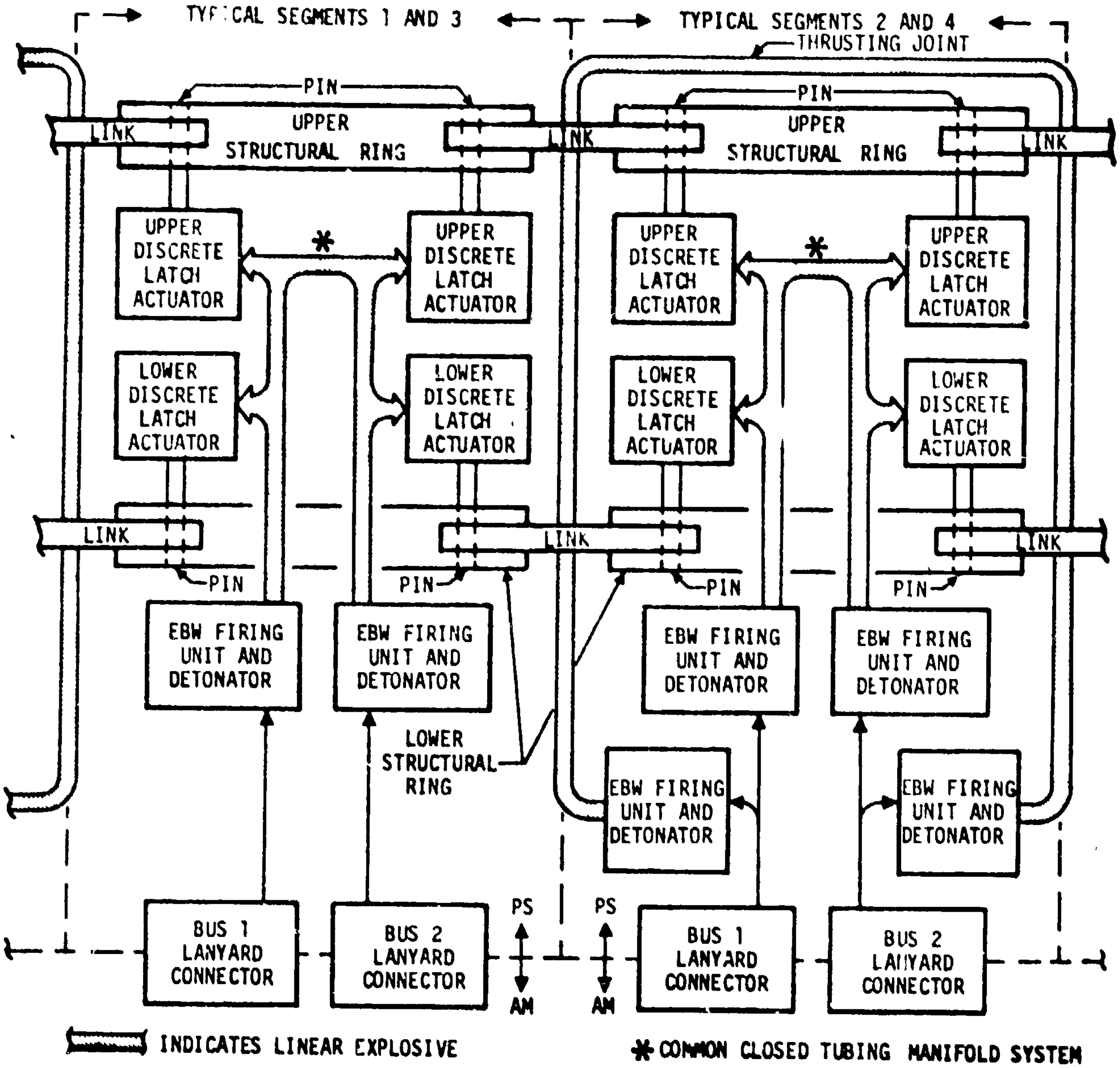


FIGURE 2.8-4 PAYLOAD SHROUD ELECTRICAL ORDNANCE

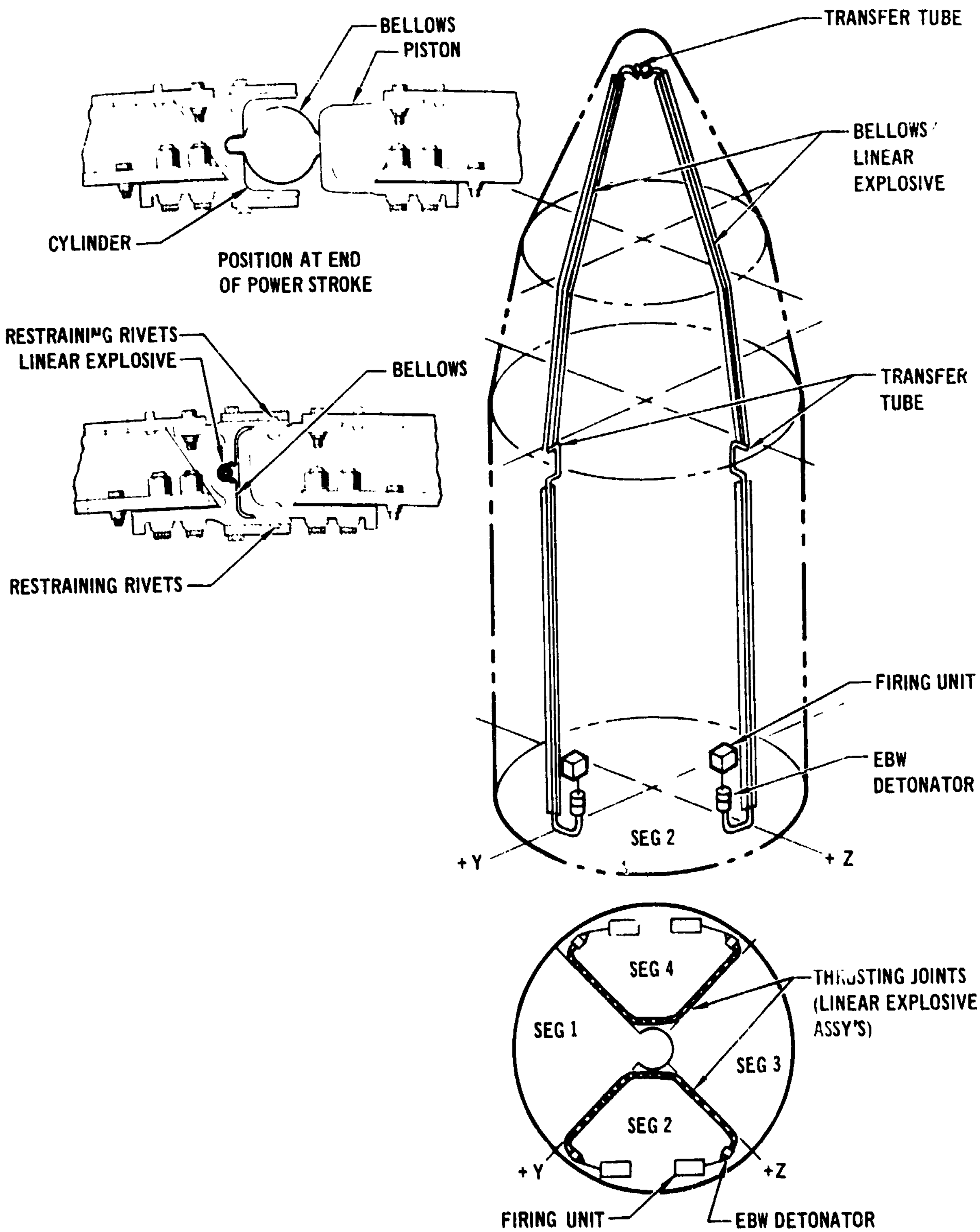


FIGURE 2.8-5 PAYLOAD SHROUD THRUSTING JOINT SYSTEM

This provided two detonators in segment 2 and two detonators in segment 4. A firing unit for each detonator provided the redundant means of igniting the thrusting joints. Since there was no ordnance ties between segments, the ignition of the two thrusting joints had to occur within a prescribed time of each other to minimize the probability of shroud contact with the payload during jettison. The electrical system was redesigned to add the OWS switch selector and the OWS relay panels powered by the AM in place of the IU switch selector and power. This provided a greater capability to maintain the command sequence. An AM relay panel was added for the jettison control logic. The OWS switch selector provided the primary and secondary jettison commands and the AM CRDU provided the backup jettison commands.

A deviation in maximum line resistance for the firing unit trigger circuit was requested and granted in the payload shroud electrical design. The integrity of the trigger circuit was not compromised since the minimum Skylab bus voltage was sufficiently high to assure a minimum trigger voltage.

2.8.1.2 Payload Shroud Jettison Subsystem Description

The payload shroud contained eight discrete latch firing units, Figure 2.8-6, and four thrusting joint firing units. The electrical signals interfaced with the shroud segments at the lanyard connectors. Receptacles for the PS lanyard connectors were mounted on the FAS. When the shroud segments moved away, the lanyard cables were pulled releasing the lanyard connectors resulting in electrical connector separation.

The OWS switch selector provided low power momentary commands, Figure 2.8-7, to the OWS relay panels. These panels utilized AM power and the commands to provide long duration commands with greater drive capability. These commands provided control for the AM control logic. The AM control logic utilized relay circuits that accepted the commands and applied power to the appropriate groups of firing units while maintaining isolation of power sources, commands, and firing units.

The PS jettison sequence had three interlocks; the power interlock, the control interlock, and the mechanical interlock. The power interlock, utilizing the

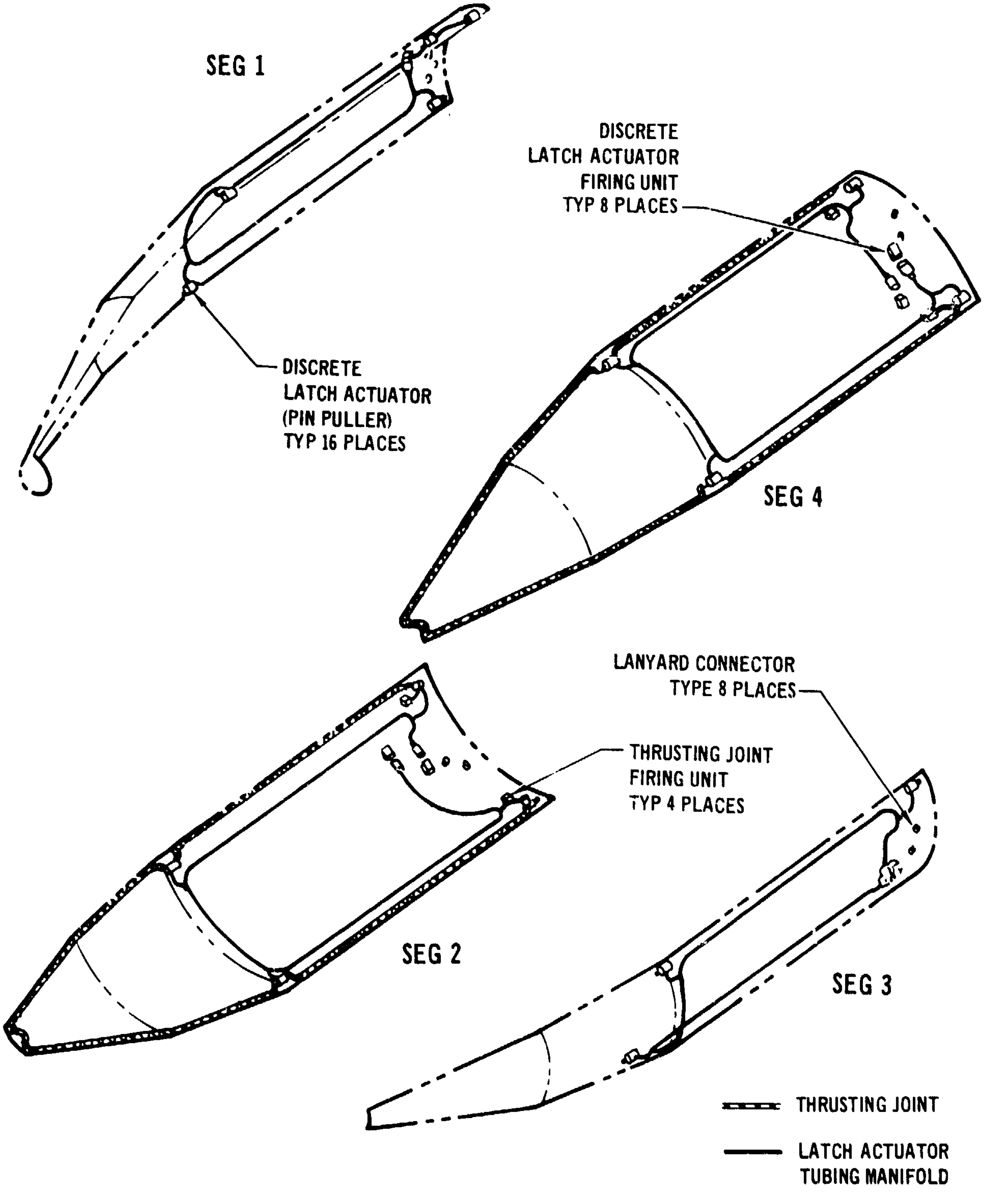


FIGURE 2.8-6 PAYLOAD SHROUD COMPONENT LOCATION

OWS SWITCH SELECTOR		AM DCS/CRDU		
PRIMARY COMMAND	SECONDARY COMMAND	BACKUP COMMAND	FUNCTIONS	AM TELEMETRY
PS ENABLE	PS ENABLE	PS ENABLE	AM-ARMS PS JETTISON CIRCUITS	NONE
PS LATCH CHARGE	NONE	PS LATCH CHARGE	AM-CHARGES FOUR BUS 1 PS LATCH EBW FIRING UNITS	8 ANALOG SIGNALS INDICATE THE CHARGE LEVEL IN THE EBW FIRING UNITS
NONE	PS LATCH CHARGE		AM-CHARGES FOUR BUS 2 PS LATCH EBW FIRING UNITS	
PS LATCH TRIGGER	NONE	PS LATCH TRIGGER	AM-FIRES FOUR BUS 1 EBW FIRING UNITS. THIS FUNCTION RELEASES THE LATCHES HOLDING THE FOUR PS SEGMENTS TOGETHER.	THIS FUNCTION IS INDICATED BY THE ABOVE ANALOG SIGNALS DROPPING TO ZERO
NONE	PS LATCH TRIGGER		AM-FIRES FOUR BUS 2 EBW FIRING UNITS. THIS FUNCTION RELEASES THE LATCHES HOLDING THE FOUR PS SEGMENTS TOGETHER.	
PS THRUSTING JOINT CHARGE	PS THRUSTING JOINT CHARGE	PS THRUSTING JOINT CHARGE	AM-CHARGES FOUR PS THRUSTING JOINT FIRING UNITS	4 ANALOG SIGNALS INDICATE THE CHARGE LEVEL IN THE EBW FIRING UNITS
PS THRUSTING JOINT TRIGGER	PS THRUSTING JOINT TRIGGER	PS THRUSTING JOINT TRIGGER	AM-FIRES THE 4 EBW FIRING UNITS CAUSING A CONFINED PYROTECHNIC CHARGE TO BURN AND EXPAND IN THE PS JOINT SEGMENTS. THIS ACTION SHEARS THE RESTRAINING HARDWARE AND PULLS LANYARD PLUGS AS THE PS SEGMENTS ARE PROPELLED AWAY FROM THE SWS.	THIS FUNCTION IS INDICATED BY THE ABOVE ANALOG SIGNALS DROPPING TO ZERO. FOUR BILEVEL SIGNALS (ONE FOR EACH SEGMENT) INDICATE THE LANYARD PLUGS HAVE SEPARATED FROM THEIR RECEPTACLES.
RESET	RESET	NONE	OWS-THESE SIGNALS RESET THE LATCHING RELAYS USED TO SEND COMMANDS TO AM.	NONE
NONE	NONE	RESET	AM-THESE SIGNALS RESET THE LATCHING RELAYS USED TO SEND THE DCS COMMANDS.	NONE

FIGURE 2.8-7 PAYLOAD SHROUD ELECTRICAL – COMMANDS/FUNCTIONS

PS jettison enable circuit, prevented premature operation of the jettison sequence. The enable commands energized the enable relays which armed the jettison circuits. The control interlock which implemented individually commanded series relay circuits prevented the sequence from continuing if a malfunction in the switch selector or OWS relays had occurred. Due to the redundancy of these relay circuits, at least two malfunctions would have been required to stop the sequence. AM CRDU commands were available to bypass the malfunctions. The mechanical interlock required the discrete latch actuators to release the upper and lower structural rings so the thrusting joints could separate the shroud segments when the joint separation commands were initiated.

The bus 1 latch control logic, Figure 2.8-8, used Sequential Bus 1 power, paragraph 2.8.4, and primary commands from the OWS switch selector and relays to charge and trigger one latch firing unit in each segment via the lanyard connectors, Figure 2.8-4. The other latch firing units were similarly charged and triggered by the bus 2 latch control logic utilizing Sequential Bus 2 power and the secondary commands. The AM CRDU, which utilizes separate sets of relays, was available as a backup method of cycling the discrete latch actuators. The bus 1 and bus 2 joint control logic used Sequential Buses 1 and 2 power, respectively, and primary commands from the OWS switch selector and relays to charge and trigger all the thrusting joint firing units, Figures 2.8-4 and 2.8-8. Upon completion of this sequence the shroud should have been jettisoned unless a malfunction had occurred. If a malfunction had occurred, the secondary commands would have cycled all the thrusting joint firing units. Also the AM CRDU provided a backup method of cycling all the joint firing units utilizing separate sets of relays.

2.8.1.3 Payload Shroud Jettison Subsystem Testing

A. Development Testing

(1) Vendor - The GFE EBW firing units had additional vibration tests performed to qualify them for use in zero gravity by verifying that no debris was present in the firing unit gap tube. The gap tube was the high voltage switching device and a piece of debris passing through the electrodes in zero gravity could have caused self-triggering of the firing unit.

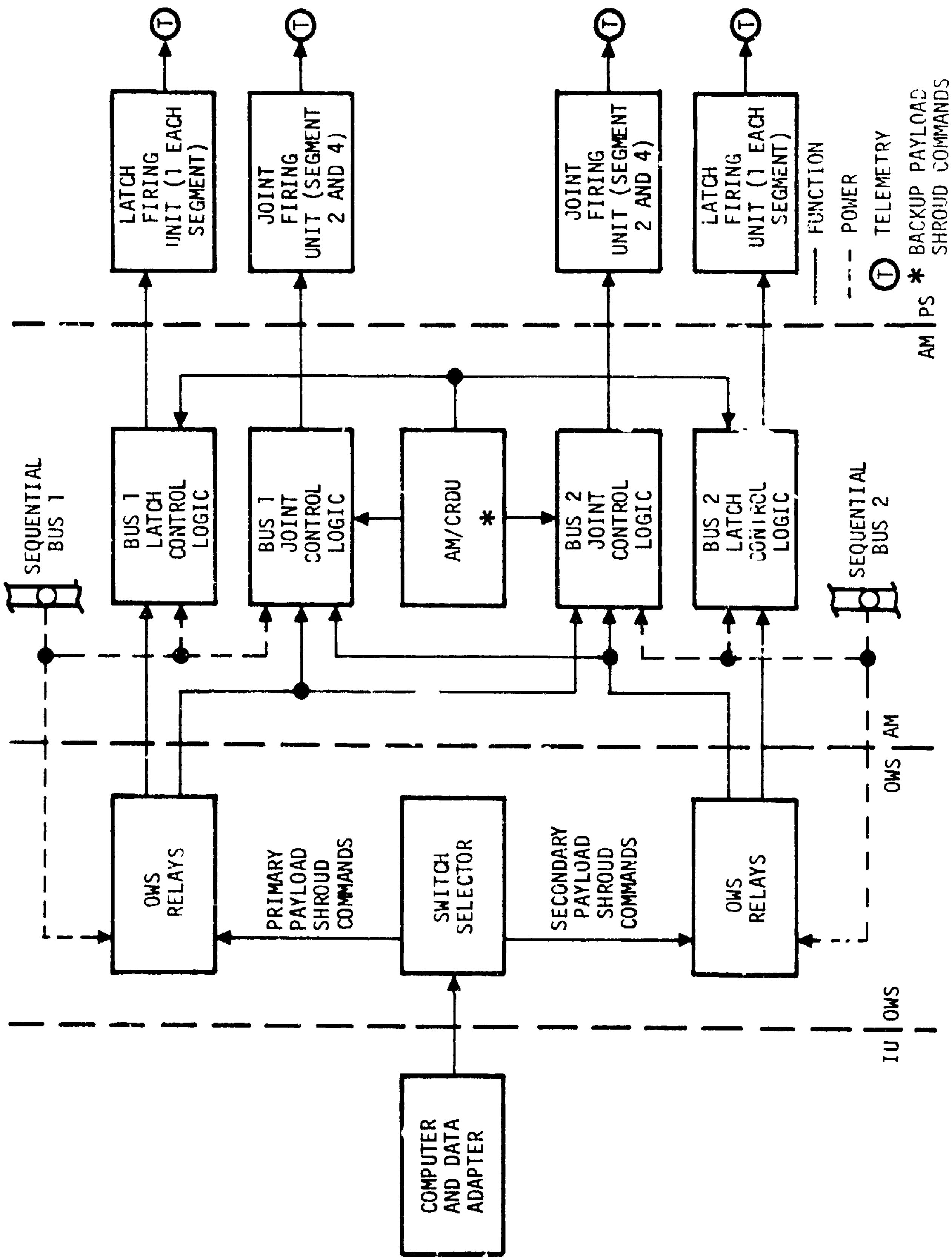


FIGURE 2.8-8 PAYLOAD SHROUD ELECTRICAL JETTISON DIAGRAM

(2) NASA Plum Brook Station - Three full scale payload shroud separation tests were performed. The electrical system in the shroud was essentially the same as the flight vehicle. The AM electrical system was abbreviated and only included flight type trigger relays for the thrusting joints. These relays were used to verify that they would ignite both thrusting joints within the prescribed time tolerance.

B. System Testing - No significant problems were encountered during these tests. Test history is shown in Figure 2.8-9.

(1) MDAC-E - The AM portion of the ordnance system was successfully verified during Payload Shroud jettison control circuitry testing. These tests verified:

- Dual and single bus operation
- Correct interlock circuitry operation
- Primary and backup control systems operation

(2) MDAC-W - An end-to-end electrical system test was successfully performed on the payload shroud to verify the integrity of the electrical circuits prior to shipping to KSC.

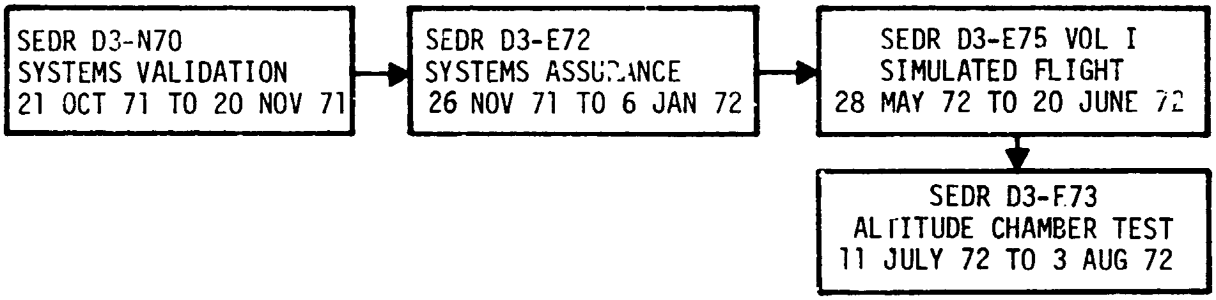


FIGURE 2.8-9 SYSTEM TESTING PAYLOAD SHROUD JETTISON SUBSYSTEM

C. Integrated Testing - KSC - Subsequent to spacecraft arrival at KSC, the AM/MDA was subjected to a series of tests including the following which were performed on the AM/MDA in the O&C building. Ordnance control circuitry verifications were performed. Interface test requirements were satisfied. The payload shroud cylinder was installed and electrically mated. The AM/MDA/FAS/DA/PS was then moved to the VAB and stacked on the launch vehicle. The ATM was then installed on the DA. The electrical interfaces between OWS and AM/MDA and between the ATM and AM/MDA were mated. OWS switch selector and DCS control of the AM system was demonstrated. A mission simulated flight test was performed during which all mission time line functions from countdown through launch sequence and activation were verified. The payload shroud nosecone was installed and the fully mated Skylab 1 vehicle was moved to the pad. While at the pad, pulse sensors were removed and live ordnance was installed. Final AM close out was accomplished. A countdown demonstration test was performed as a rehearsal for launch countdown and to obtain a timeline for countdown events. This testing culminated in a successful problem-free countdown and launch.

The ordnance control system was checked out for all redundant modes. EBW firing units were operated into pulse sensors and all interlock circuits verified. Figure 2.8-10 identifies the significant major problems.

PROBLEM	SOLUTION/ACTION
PAYLOAD SHROUD EBW FIRING UNIT (1A3A1) EXHIBITED OUT OF TOLERANCE TM CHARGING VOLTAGE.	THE FIRING UNIT WAS REPLACED AND SATISFACTORILY RETESTED. REF.: TCP KM0003, DR AM1-03-0449.
IN THE VAB, THE EXTERNAL SEQUENTIAL CIRCUIT BREAKER PANEL WAS REPLACED WITH PASS [illegible] "FUSE WIRE" PANEL DUE TO FAILURE OF THIS CONFIGURATION PANEL DURING LAUNCH VIBRATION TESTING.	ALL FUNCTIONS ROUTED THROUGH THE NEW CONFIGURATION PANEL WERE SUBSEQUENTLY RETESTED. REF.: ECP 1152, TPS AM1-03-0186.

FIGURE 2.8-10 SUMMARY OF LAUNCH SITE SIGNIFICANT ORDNANCE AND DEPLOYMENT PROBLEMS

2.8.1.4 Payload Shroud Jettison Subsystem Mission Results

The payload shroud jettison sequence was initiated after orbital insertion during the pitch maneuver to gravity gradient attitude, Figure 2.8-11. At 13 minutes 13.9 seconds after lift-off the discrete latch actuator firing units charged and remained charged for about 5 seconds at which time they were triggered, Figure 2.8-12. All eight latch actuator firing units functioned normally indicating that the latch ordnance system was redundantly initiated from the firing units and that the firing unit circuits functioned as planned. When gravity gradient attitude was attained two minutes later at 15 minutes 18.8 seconds, the four thrusting joint EBW firing units were charged. About 1.5 seconds later, at 15 minutes 20.348 $^{+.000}_{-.0125}$ seconds the OWS switch selector issued the thrusting joint trigger command. At 15 minutes 20.41 $^{+.000}_{-.099}$ seconds the payload shroud lanyard

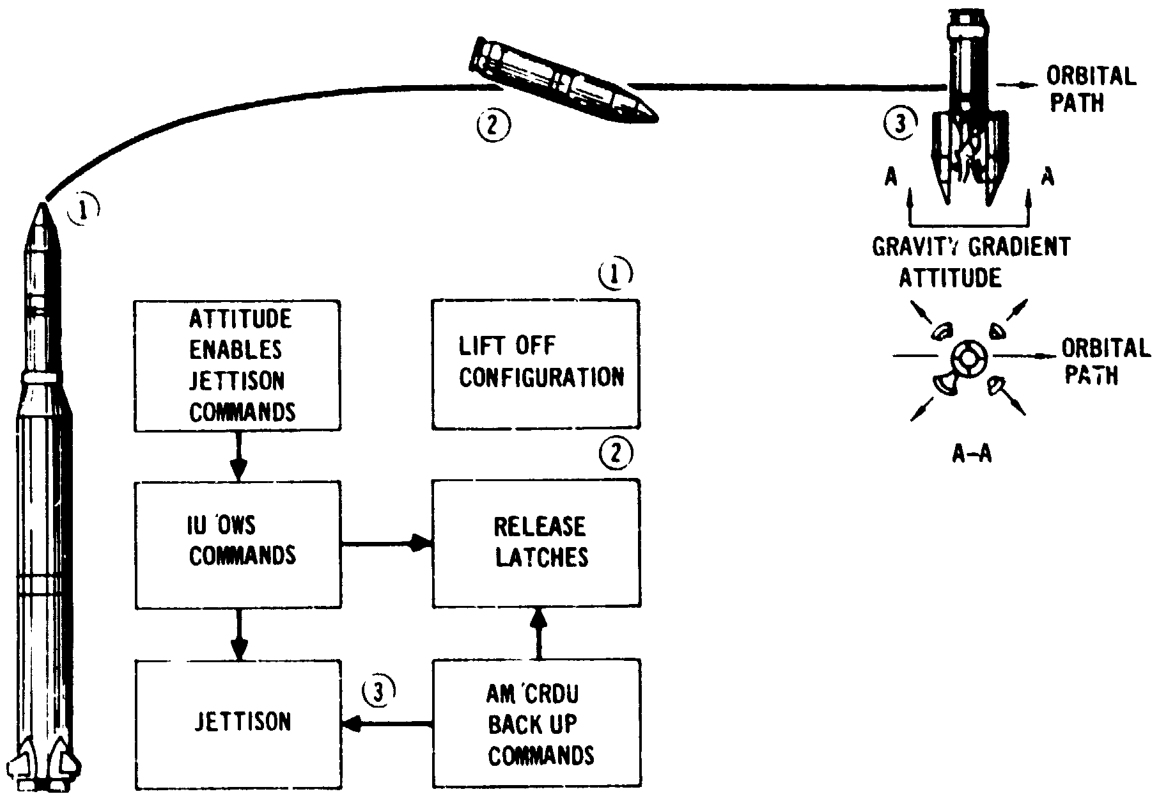


FIGURE 2.8-11 PAYLOAD SHROUD JETTISON

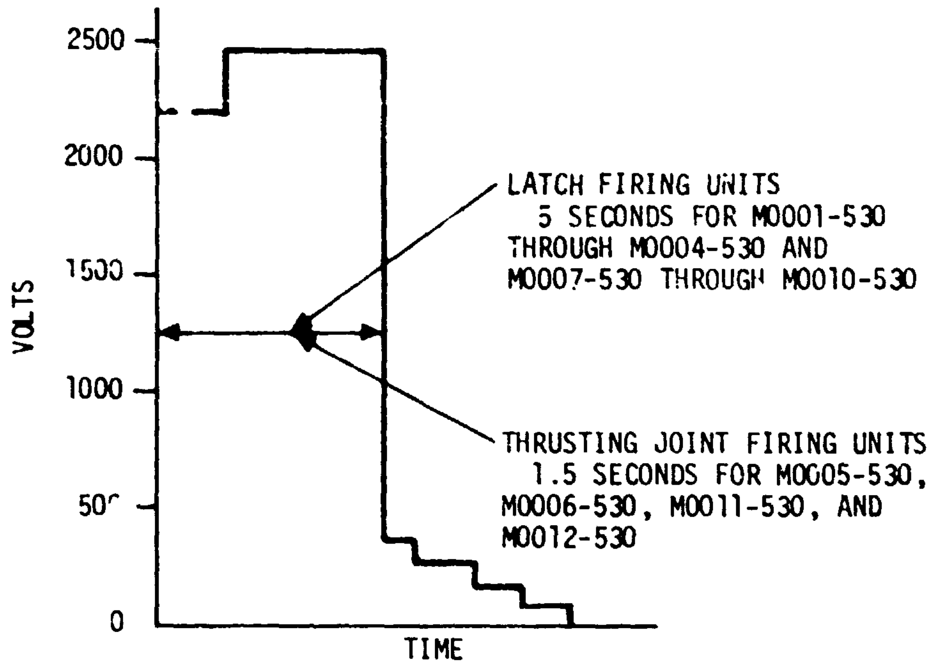


FIGURE 2.8-12 TYPICAL EBW FIRING UNIT CHARGE/TRIGGER CURVE (TELEMETRY DATA)

connectors telemetry indications verified separation. Calculated jettison time was 15 minutes 20.38 $^{+.008}_{-.004}$ seconds which was arrived at by starting with the OWS switch selector trigger command and adding the OWS and AM relay and the average PS (Plum Brook test data) operate times together. Figure 2.8-13 indicated the predicted jettison sequence which was the same as the actual within the tolerance of the telemetry system. The electrical sequential system operated as planned with no hardware failures. The AM CRDU backup jettison circuits were not required.

2.8.1.5 Payload Shroud Jettison Subsystem Conclusions and Recommendations

A. Conclusions - The payload shroud jettison circuits were proven to be adequate during the SL-1 flight. The system operated as planned with no anomalies. The major design problem encountered was the detonation of the two linear explosives within a prescribed time tolerance. A premature firing of one thrusting joint firing unit could have caused a recontact problem between the shroud and payload during separation.

B. Recommendations - The payload shroud jettison subsystem performed satisfactorily during testing and flight and no modifications are recommended.

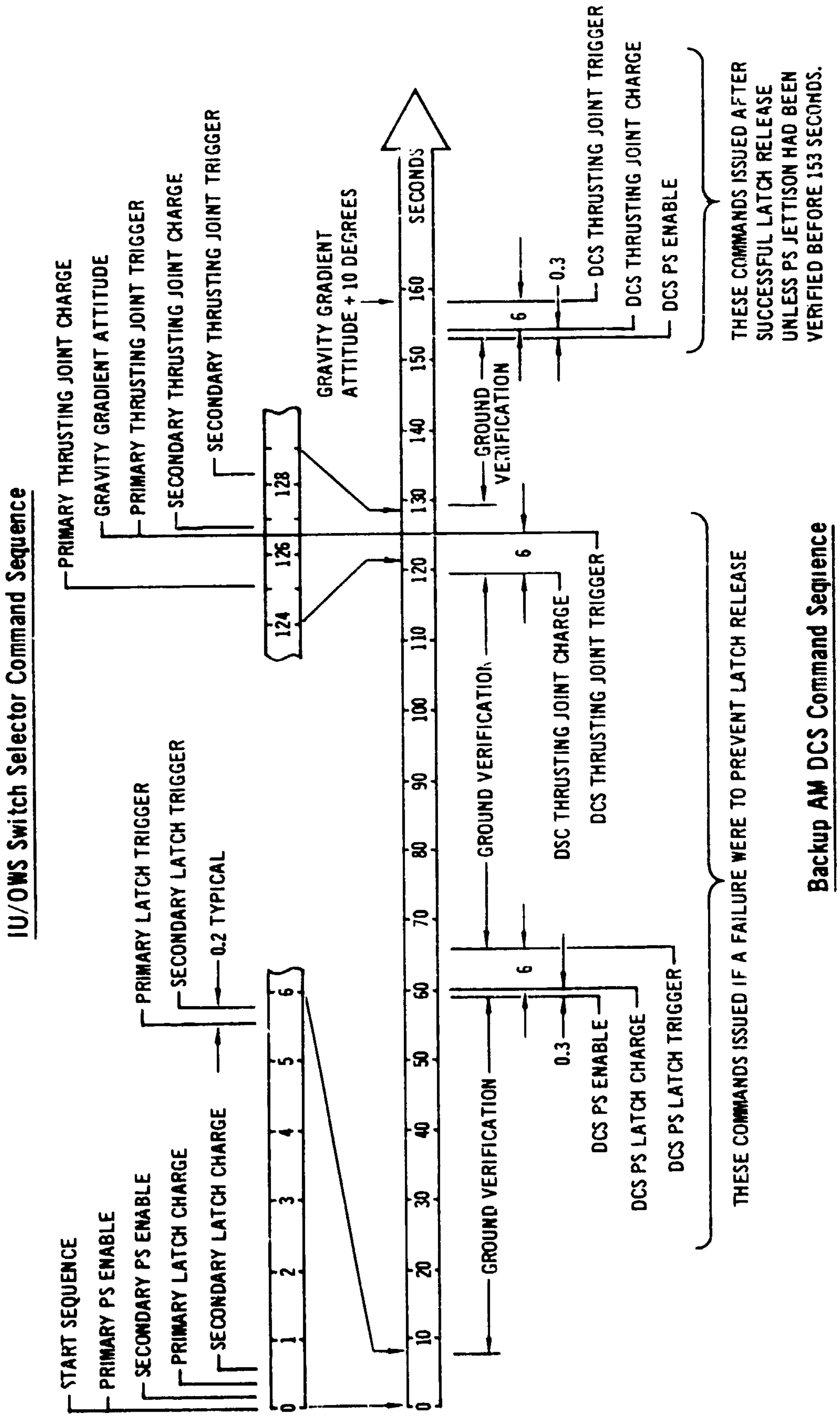


FIGURE 2.8-13 PAYLOAD SHROUD JETTISON SEQUENCE

2.8.2 ATM Deployment Subsystem

The ATM required deployment from a launch position to a mission position to clear the primary docking port and orient the ATM so its systems could function as designed. The Deployment Assembly (DA) structure contained the mechanisms required for deployment. Refer to Section 2.2 for details of the DA. Electrical sequential subsystem deployment equipment included the OWS switch selector, the AM CRDU, the AM power system, the release mechanisms, and motor reel assemblies. Automatic and JSC flight controller manual operating capabilities were included in the design. Flight qualified equipment was utilized in the design.

2.8.2.1 ATM Deployment Subsystem Design Requirements

The ATM deployment circuit design was initiated when the early mechanical tradeoff studies were completed. Initially, the DA design consisted of spring loaded trunnion joints which were to rotate the upper DA around to the deployed position. Redundant motors were to control reels which payed out cable to allow rotation. Scissor linkages and pin pullers which were not ordnance actuated were to be used to release the launch latches to allow rotation. Discrete signals were to be provided by the IU to the DA for initiation of the ATM deployment sequence. The primary signals were to be provided by IU programmed commands and the backup signals were to be provided by the IU via CCS commands. All DA electrical power requirements were to be provided by the IU. Subsequent redesign changed the payout reel to a deployment reel which pulled the ATM around during deployment. The negator springs at the trunnion joints were reversed to retard rotation of the ATM DA. The reversing of the system made more positive the cycling of the latch mechanism allowing for higher spring forces in the latch (see Section 2.2). The release mechanism (launch latch) changed to an ordnance initiated system using first the payload shroud discrete latch actuators and finally using smaller pin pullers. The ordnance system was initiated by redundant GFE EBW firing units. The electrical system was also redesigned. The OWS switch selector and the OWS relay panels powered by the AM were utilized to provide a greater command driving capability and to maintain the command sequence. AM relay panels were added for the ATM deployment control logic. The IU switch selector and power were deleted. The OWS switch selector provided the primary deployment commands. The AM CRDU provided a backup set of commands.

2.8.2.2 ATM Deployment Subsystem Description

The ATM deployment assembly (DA) contained two release mechanism EBW firing units, and two motor reel assemblies. The OWS switch selector provided low power momentary commands, Figure 2.8-14, to the OWS relay panels. These panels utilized AM power and the momentary commands to provide long duration commands with greater drive capability. The AM control logic utilized relay circuits that accepted the commands and applied power to the appropriate firing unit or motor reel assembly while maintaining isolation of power sources, commands, firing units and the motor reel assemblies.

One of the relay circuit functions was to provide ATM deployment inhibit control. The ATM deployment inhibit circuits prevented electrical initiation of the ATM deployment sequence before PS jettison. The inhibit circuit was initialized by the PS jettison enable function. This action energized, via the PS lanyard connectors, the ATM deploy inhibit relays. These relays were designed to inhibit the ATM enable and motor on commands from initiating the ATM deployment sequence until the PS was successfully jettisoned. When the PS was jettisoned the lanyard connectors disconnected the circuit to the inhibit relays. The inhibit relays deenergized and allowed the deployment commands to proceed. The deployment sequence consisted of several interlocks; the power interlock, the control interlock, and the mechanical interlock. The power interlock, utilizing the ATM DA enable circuit, prevented premature operation of the deployment sequence. The enable commands energized the enable relays which armed the deployment circuits. The control interlock which implemented individually commanded series relay circuits prevented the sequence from continuing if a malfunction in the switch selector or OWS relays had occurred. Due to the redundancy of these relay circuits, at least two malfunctions would have been required to stop the sequence. AM CRDU commands were available to bypass the malfunctions. The mechanical interlock required the release mechanism to free the upper DA so the deployment motor reel assemblies could deploy the ATM.

The primary deployment relay logic, Figure 2.8-15, used Sequential Bus 1 power, Paragraph 2.8.4 and primary commands from the OWS switch selector and relays to charge and trigger one release mechanism firing unit. The redundant firing unit was charged and triggered in a similar fashion using Sequential Bus 2 power, the secondary deployment relay logic and the primary commands from the OWS

OWS SWITCH SELECTOR OUTPUT (PRIMARY)	AM DCS/CRDU OUTPUT (BACKUP)	FUNCTION	TELEMETRY FUNCTION
ATM DA ENABLE COMMAND	ATM DA ENABLE COMMAND	AM-ARMS ATM DA LATCH RELEASE CIRCUITS.	NONE
NONE	DCS SYSTEM SELECT COMMAND	AM-SELECTS AM DCS COMMAND SOURCE.	NONE
ATM DA LATCH RELEASE CHARGE COMMAND	ATM DA LATCH RELEASE CHARGE COMMAND	AM-CHARGES THE TWO LATCH RELEASE EBW FIRING UNITS LOCATED ON THE DA TRUSS.	TWO ANALOG SIGNALS INDICATE THE CHARGE LEVEL IN THE EBW FIRING UNITS.
ATM DA LATCH RELEASE TRIGGER COMMAND 1	ATM DA LATCH RELEASE TRIGGER COMMAND	AM-FIRES THE BUS 1 EBW FIRING UNIT. THIS FUNCTION RELEASES THE MECHANISMS HOLDING THE STABILIZATION TRUSS.	THIS FUNCTION IS INDICATED BY THE ABOVE TWO ANALOG SIGNALS DROPPING TO ZERO.
ATM DA LATCH RELEASE TRIGGER COMMAND 2		AM-FIRES THE BUS 2 EBW FIRING UNIT. THIS FUNCTION RELEASES THE MECHANISMS HOLDING THE STABILIZATION TRUSS.	
ATM DEPLOYMENT MOTORS ON COMMAND	ATM DEPLOYMENT MOTORS ON COMMAND	AM-ACTIVATES THE TWO DEPLOYMENT MOTORS. THIS FUNCTION ROTATES THE ATM INTO THE 90° DEPLOYED POSITION.	TWO BILEVEL SIGNALS INDICATE POWER APPLIED TO DEPLOYMENT MOTORS.
			TWO BILEVEL SIGNALS INDICATE DOWN LIMIT SW'S HAVE BEEN TRIPPED, INDICATING ATM DEPLOY.
NONE (PRIMARY COMMAND MODE SELECTED)	NONE	AM-DEACTIVATES THE TWO DEPLOYMENT MOTORS AUTOMATICALLY BY THE LIMIT SWITCH CIRCUITS.	THIS FUNCTION IS INDICATED BY THE TWO BILEVEL SIGNALS INDICATING POWER APPLIED TO ATM DEPLOYMENT MOTORS DROPPING TO ZERO.
NONE	ATM DEPLOYMENT MOTORS OFF COMMAND	AM-DEACTIVATES THE TWO DEPLOYMENT MOTORS.	
RESET COMMAND	NONE	OWS-THESE SIGNALS RESET THE LATCHING RELAYS USED TO SEND COMMANDS TO AM.	NONE
NONE	RESET COMMAND	AM-THESE SIGNALS RESET THE AM DCS LATCHING RELAYS AND SELECT THE IU/OWS COMMAND SOURCE.	NONE

FIGURE 2.8-14 ATM DEPLOYMENT ELECTRICAL – COMMANDS/FUNCTIONS

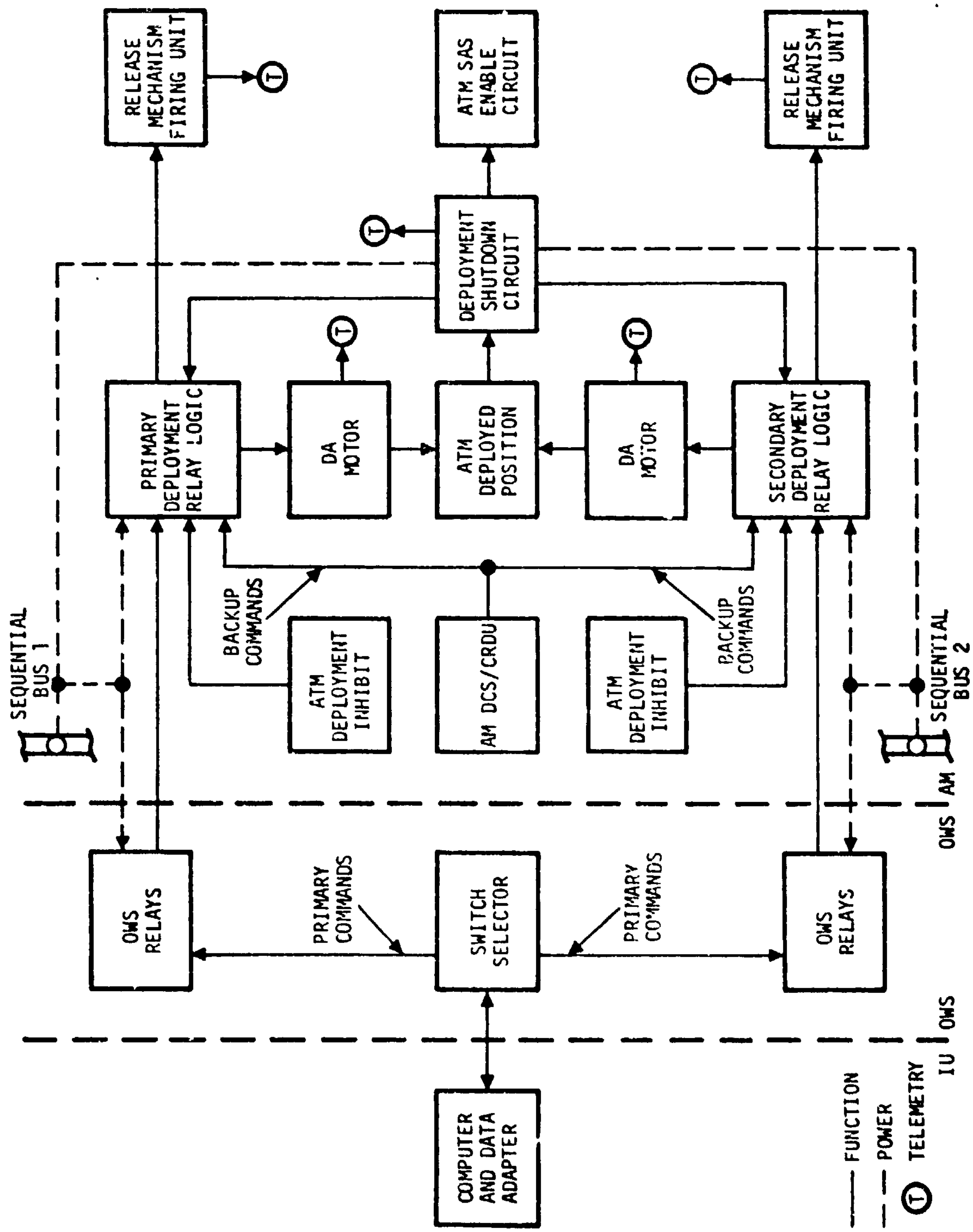


FIGURE 2.8-15 ATM DEPLOYMENT DIAGRAM

switch selector. The AM CRDU provided a backup method of cycling the two firing units. The primary and secondary deployment relay logic used Sequential Bus power and the primary command from the OWS switch selector to start the deployment motor reel assemblies. The AM CRDU provided a backup method for starting the motors. When the ATM rotated to the fully deployed position the latch mechanism captured the upper DA and actuated the down limit switches which turned the motors off after a short time delay allowing the latch to cinch up tight. The limit switches also enabled the command circuits to allow ATM SAS deployment. The AM CRDU provided a backup method of stopping the motors and deploying the ATM SAS.

2.8.2.3 <u>ATM Deployment Subsystem Testing</u>

A. <u>Development Testing</u>

1. <u>Vendor</u> - The GFE EBW firing units had additional vibration tests performed to qualify them for use in zero gravity by verifying that no debris was present in the firing unit gap tube. The gap tube was the high voltage switching device and a piece of debris passing through the electrodes in zero gravity could have caused self-triggering of the firing unit.

   Additional EMI testing was performed on the time delay relays to verify that they conformed to Skylab requirements.

2. <u>MDAC-E</u> - Vibration testing was performed on the time delay relays to verify that they would survive the Skylab environment.

B <u>System Testing</u> - No major problems were encountered during these tests. Test history is shown in Figure 2.8-16.

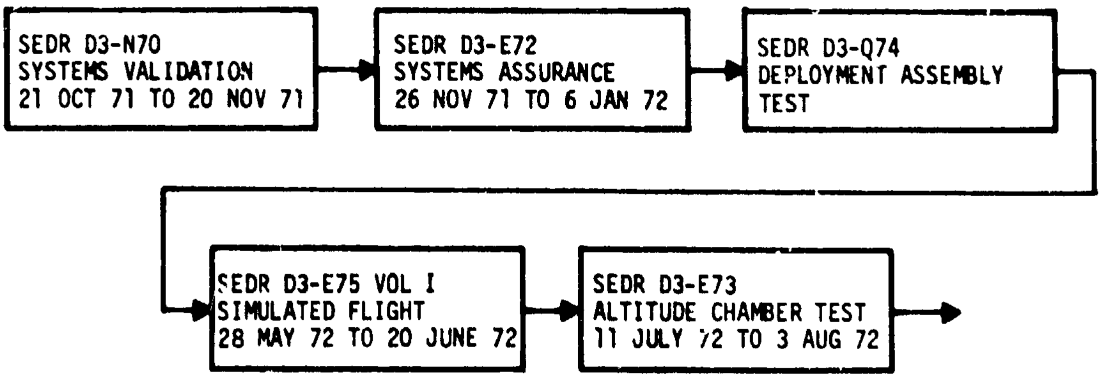


FIGURE 2.8-16 SYSTEM TESTING – ATM DEPLOYMENT SUBSYSTEM

MDAC-E - The ordnance and deployment system was successfully verified during DA deployment control circuitry testing. These tests verified:

- Dual and single bus operation
- Correct interlock circuitry operation
- Limit switch operation
- Time delay timing
- Primary and backup control systems operation

In an actual ATM deployment test, dual and single motor/reel mode operation was verified. No major problems were encountered during these tests.

C. Integrated Testing - KSC - Subsequent to spacecraft arrival at KSC the AM/MDA was subjected to a series of tests including the following which were performed on the AM/MDA in the O&C building. Ordnance control circuitry verifications were performed. Interface test requirements were satisfied. The AM/MDA was hardmated to the FAS and DA and all electrical interfaces in this configuration were flight mated. A powered DA deployment was performed. The AM/MDA/FAS/DA/PS was then moved to the VAB and stacked on the launch vehicle. The ATM was then installed on the DA. The electrical interfaces between OWS and AM/MDA and between the ATM and AM/MDA were mated. A comprehensive AM/OWS/ATM electrical interface test verified all systems operational function. OWS switch selector and DCS control of the AM system was demonstrated. A mission simulated flight test was performed during which the ATM deployment sequence was verified. While at the pad, pulse sensors were removed and live ordnance was installed. Final AM close out was accomplished. A countdown demonstration test was performed as a rehearsal for launch countdown and to obtain a timeline for countdown events. This testing culminated in a successful problem free countdown and launch. The ordnance and deployment control system was checked out for all redundant modes. DA motors were operated, EBW firing units operated into pulse sensors and all interlock circuits verified.

2.8.2.4 ATM Deployment Subsystem Mission Results

The ATM deployment sequence was initiated after payload shroud jettison, at 16 minutes 36.42 seconds, Figure 2.8-17. First, ATM DA firing units charged and remained charged for about three seconds at which time they were triggered, Figure 2.8-18. Both release mechanism firing units functioned normally indicating that the release mechanism ordnance system was redundantly initiated from the firing units and that the firing unit circuits functioned as planned. About 10 seconds later the motor reel assemblies were commanded "ON". Both motors started and deployed the ATM in 3 minutes 9 seconds. Full deployment of the ATM was indicated by the actuation of the ATM DA down limit switches. These switches enabled the ATM SAS command circuits and started time delay relays which, after about 15 seconds, turned the motors "OFF". Figure 2.8-19 indicated the predicted deployment sequence which was the same as the actual within the tolerance of the telemetry system except for the motors which performed better than predicted. The system operated as planned with no hardware failures and the backup deployment circuits were not utilized.

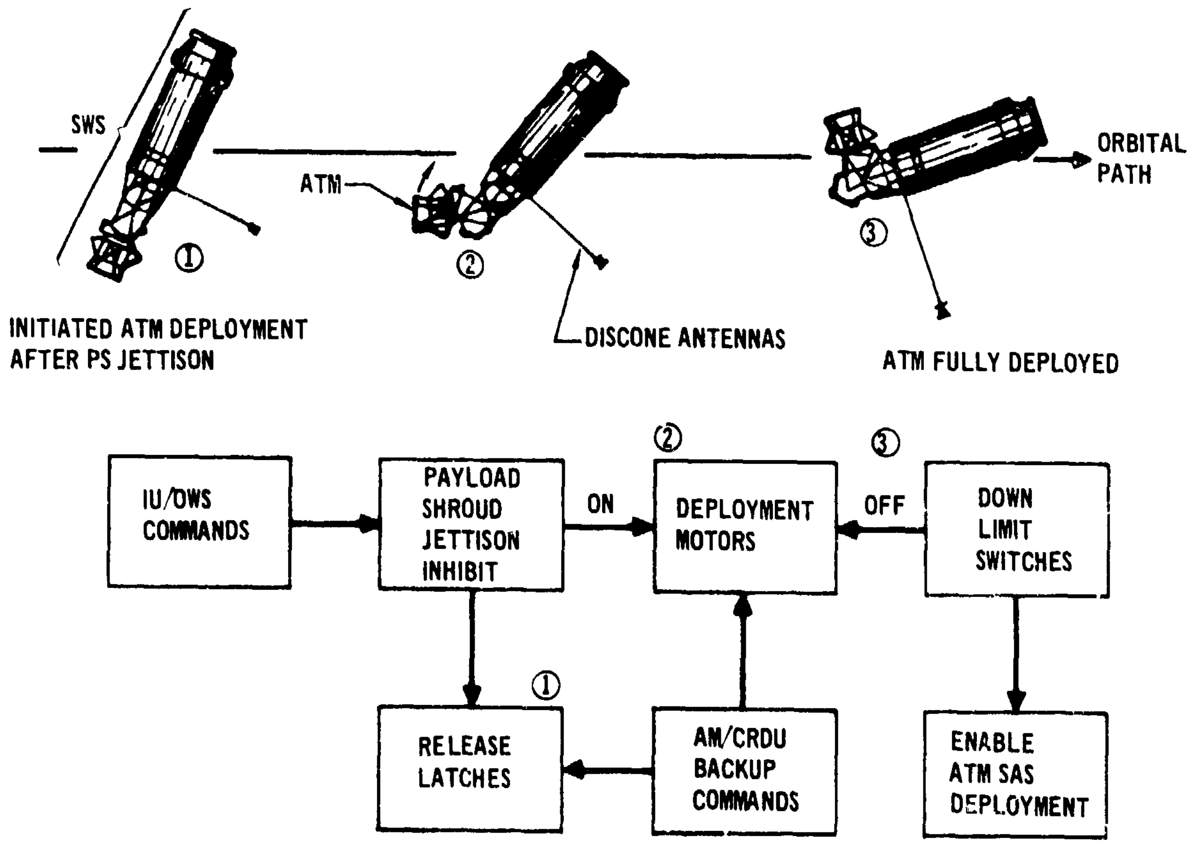


FIGURE 2.8-17 ATM DEPLOYMENT

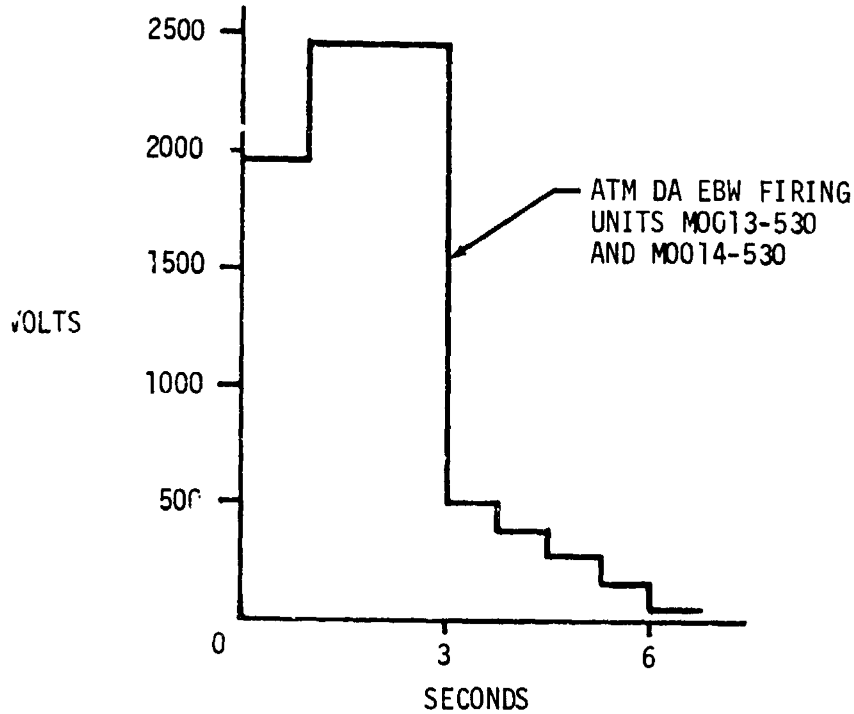


FIGURE 2.8-18 TYPICAL EBW FIRING UNIT CHARGE/TRIGGER CYRVE (TELEMETRY DATA)

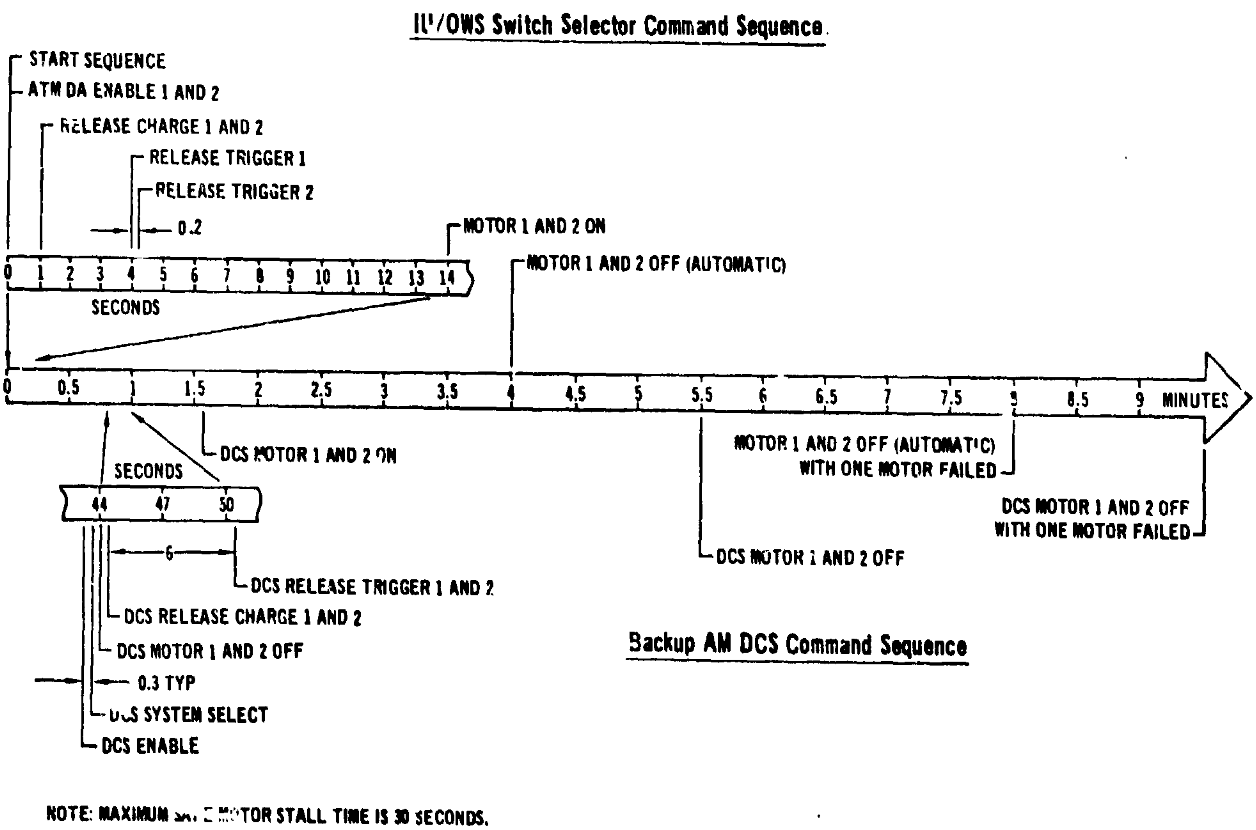


FIGURE 2.8-19 ATM DEPLOYMENT SEQUENCE

2.8.2.5 ATM Deployment Subsystem Conclusions and Recommendations

A. Conclusions - The ATM deployment circuits were proven to be adequate during the SL-1 flight. The system operated as planned with no anomalies. An additional telemetry monitor which indicated the position of the ATM during deployment would have aided the JSC flight controllers in identifying the status of ATM position during deployment.

B. Recommendations - The ATM deployment subsystem performed satisfactorily during tests and the mission. Added telemetry parameters providing ATM deployment position data would be an aid.

2.8.3 Discone Antenna Deployment Subsystem

The electrical sequential system was required to release the discone antennas from the launch secured position, allowing the mechanical system to deploy the antennas. JSC flight controller manual operating mode was selected as the primary and backup method of commanding antenna deployment.

2.8.3.1 Discone Antenna Deployment Subsystem Design Requirements

The discone antenna circuit design originally utilized an ordnance actuated guillotine to release the antenna booms. Eventually the ordnance system was eliminated and hot wire actuators were added. Two AM CRDU commands were added to allow the JSC flight controllers to deploy the antennas. The original flight plan called for the ATM to be launched separately from the SWS and by using LM, the LM/ATM would be docked with the SWS. The crew would then deploy the antennas. After the SWS went through its final evolution to a dry workshop, the ATM was launched with the SWS and the antennas were to be deployed by JSC flight controllers. The electrical circuit went through a minimum of redesign retaining the crew deployment capability. The design used hot wire actuators which utilized the principle of heating a wire until it fused to allow a spring loaded plunger (pin) to retract. The pin retraction permitted the scissor mechanism to open freeing the cables and allowing the spring loaded retainers to release the strap assemblies. Once the straps were released the spring loaded rotary joints rotated deploying the antennas.

2.8.3.2 Discone Antenna Deployment Subsystem Description

The discone antenna deployment circuit, Figure 2.8-20, consisted of the ground command mode and the crew control mode. The antenna deploy switch, when placed in the command position, would allow each AM CRDU command to pass through the switch and energize a relay. The relay would apply power to two hot wire actuators for a sufficient length of time to fuse the wire resulting in the release of the actuator pins and the deployment of the antennas. An alternate or crew control method of deploying the antennas was available by having the crew place the antenna switch in the deploy position. This energized both relays and actuated all the hot wire actuators releasing the antennas.

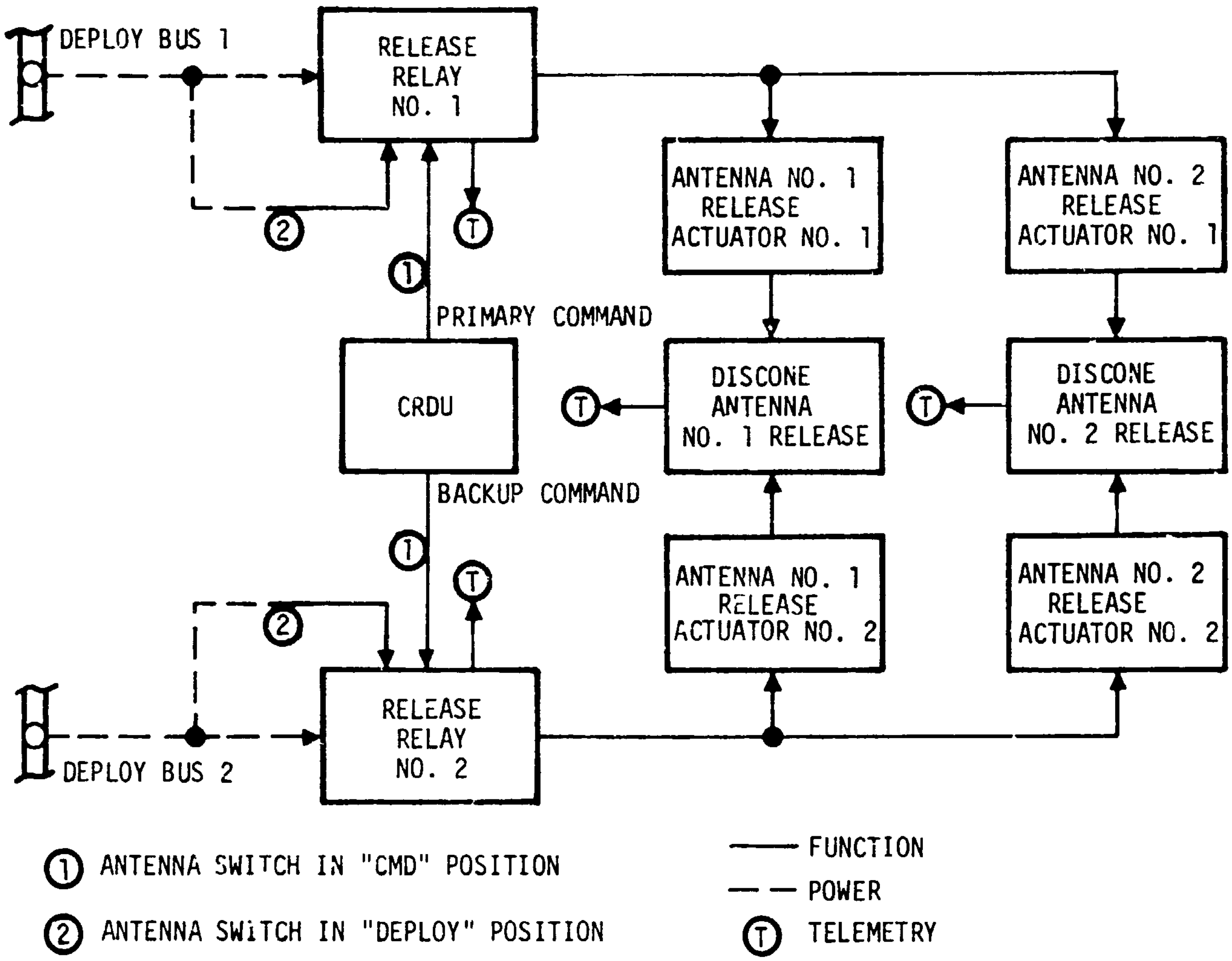


FIGURE 2.8-20 DISCONE ANTENNA DEPLOYMENT DIAGRAM

2.8.3.3 Discone Antenna Deployment Subsystem Testing

A. Development Testing

1. Vendor - The hot wire actuators were originally qualified for the Apollo Program. Additional vibration and high and low temperature testing was performed at the vendor to meet Skylab requirements.

B. System Testing

1. MDAC-E - The discone antenna deployment system was successfully verified during control circuitry testing. These tests verified:
   - Dual and single bus operation
   - Primary and backup control system operation

   No major problems were encountered during these tests.

C. Integrated Testing

1. KSC - Subsequent to spacecraft arrival at KSC the AM/MDA was subjected to a series of tests including the following which were performed on the AM/MDA in the O&C building. The AM/MDA was hard-mated to the FAS and DA and all electrical interfaces in this configuration were flight mated. At the launch pad DCS control of the system was demonstrated. The hot wire actuators were installed and checked out. Final AM close out was accomplished including photographs of all internal panel switch and circuit breaker positions.

2.8.3.4 Discone Antenna Deployment Subsystem Mission Results

The discone antenna deployment sequence was initiated after PS jettison and the arming of the Deploy Buses. Telemetry indicated actuation of the deploy circuits at 16 minutes 10.1 seconds and at 16 minutes 34.1 seconds. The first actuation released both antennas. Discone antennas number 1 and 2, Figure 2.8-21, were fully deployed at 16 minutes 54.2 seconds and 16 minutes 52.5 seconds, respectively. The circuits operated as planned with no hardware failures.

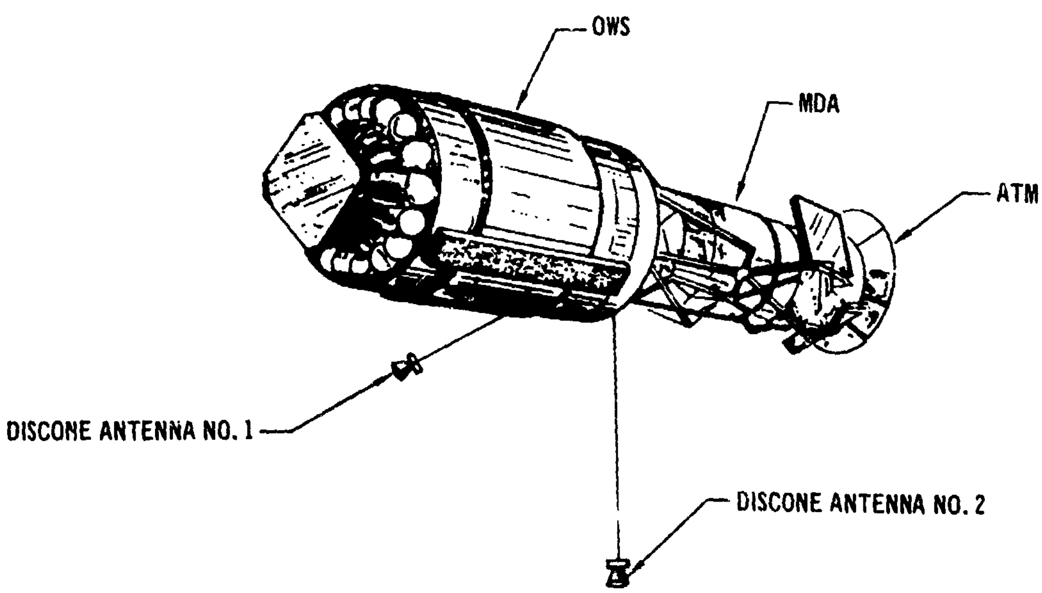


FIGURE 2.8-21 DISCONE ANTENNAS

2.8.3.5 Discone Antenna Deployment Subsystem Conclusions and Recommendations

A. Conclusions - The discone antenna deployment circuits were proven to be adequate during the SL-1 flight. The system operated as planned with no anomalies.

B. Recommendations - No recommended design changes should be implemented.

2.8.4 Power Control Subsystem

Electrical power was controlled to prevent premature initiation of an activation sequence, especially during ascent vibration. The power buses associated with the sequential system were armed when required, and then safed after functions were completed. The primary method of arming the buses was automatic, with the JSC flight controllers providing backup arming capability. The power control system was successfully tested at MDAC-E and at KSC.

During flight the power control system armed the buses as planned. Due to problems encountered with the deployment of the OWS SAS and meteoroid shield, the Deploy Buses were disarmed early by the flight controllers and the Sequential Buses were disarmed by the SL-2 crew during the activation period which was about 10 days late due to the launch delay of SL-2. The power control system functioned with no failures.

2.8.4.1 Power Control Subsystem Design Requirements

In the initial design, Squib Buses were created to control the Am cryogenic system squibs. This system was to supply the cryogenics for the CSM fuel cells. When discone antenna deployment circuits were added to the buses, the name was changed from Squib to Deploy. Mechanical/ordnance tradeoff studies resulted in mounting of the SAS on the OWS using an EBW ordnance system for deployment. This deployment system was added to the Deploy Buses and the cryogenic system was deleted. Subsequently the meteoroid shield deployment circuits and selected ATM activation sequences were added to the Deploy Buses. These buses were to be armed late in the first orbital revolution. The PS jettison and ATM deployment circuits were planned to be activated shortly after orbital insertion. The Sequential Buses were created for this purpose. Later the OWS refrigeration radiator shield jettison and selected ATM activation circuits were added to the Sequential Buses. Subsequent changes in the flight plan resulted in deployment of the discone antennas just after PS jettison resulting in Deploy Bus arming early in the first orbital revolution. At this time the two buses could have been combined or the antenna circuits could have been moved to the Sequential Buses. Since the circuits were already built and installed and a change would have delayed the checkout of the AM, the two buses remained intact.

2.8.4.2 Power Control Subsystem Description

The OWS switch selector provided the primary means of arming or disarming the Deploy Buses, Figure 2.8-22. AM Bus 2 power was supplied to the OWS relay via the command position of the Deploy Bus Arm switch. When the switch selector issued the primary arm command, the OWS relay latched in, latching the No. 2 Deploy Bus Arm Relay which then connected AM Buses 1 and 2 to deploy Buses 1 and 2, respectively. When the switch selector issued the primary disarm command, the OWS relay reset, resetting the No. 2 Deploy Bus Arm relay which disconnected AM Buses 1 and 2 from Deploy Buses 1 and 2. The backup method of arming or disarming the Deploy Buses supplied AM Bus 1 power via the command position of the Deploy Bus Arm switch to the relay module in the DCS. When the permission and arm commands were issued by the DCS, the relays in the DCS relay module latched, latching the No. 1 Deploy Bus Arm Relay connecting AM Buses 1 and 2 to Deploy Buses 1 and 2, respectively. When the permission or arm reset commands were issued by the DCS, the relays in the DCS relay modules reset, resetting the No. 1 Deploy Bus Arm relay which disconnected AM Buses 1 and 2 from Deploy Buses 1 and 2.

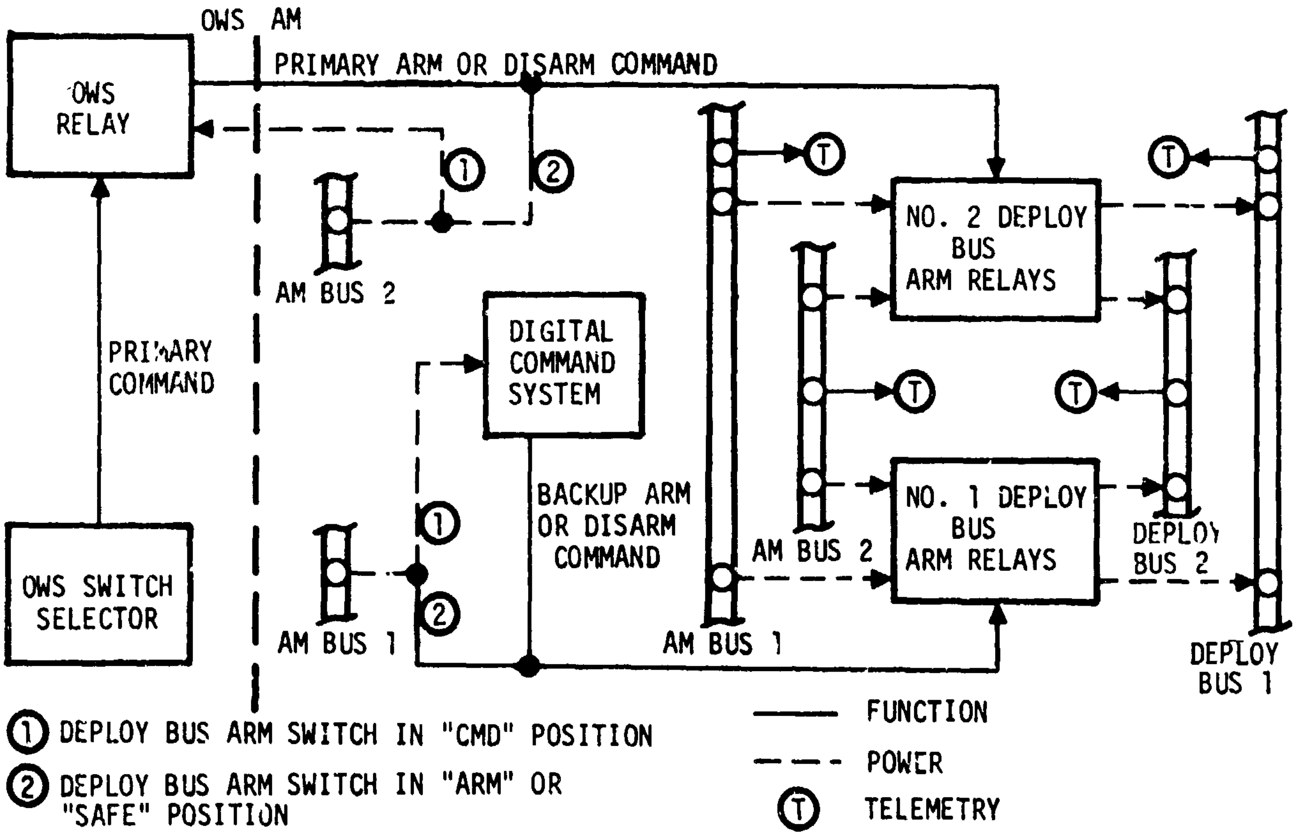


FIGURE 2.8-22 DEPLOY BUS CONTROL DIAGRAM

The deploy bus arm switch could have been manually positioned to the arm or safe position which would have armed or disarmed the Deploy Buses, using both sets of arming relays.

The OWS switch selector, Figure 2.8-23, provided two commands to arm the Sequential Buses. The primary command latched in an OWS relay which switched AM Bus 1 power to energize the No. 1 Sequential Bus Arm Relay connecting AM Buses 1 and 2 to Sequential Buses 1 and 2, respectively. The secondary command repeated this process using AM Bus 2 power and the No. 2 Sequential Bus Arm Relay. The CRDU was capable of issuing a backup command which energized both the No. 1 and No. 2 Sequential Bus Arm Relays. Disarming of the Sequential Buses was accomplished by the crew placing the Sequential Bus switch in the safe position. This reset the arm relays and disconnected AM Buses 1 and 2 from Sequential Buses 1 and 2.

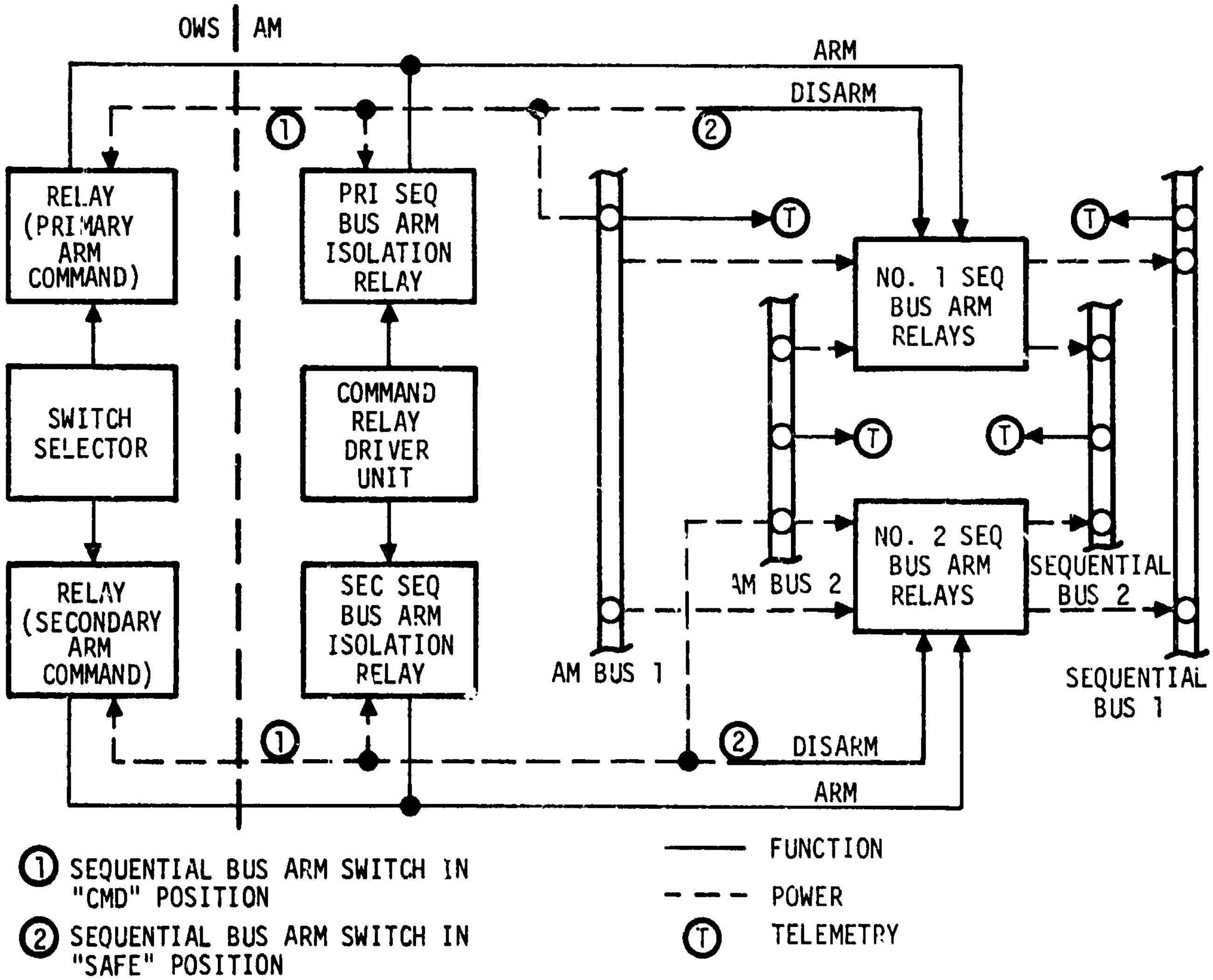


FIGURE 2.8-23 SEQUENTIAL BUS CONTROL DIAGRAM

2.8.4.3 Power Control Subsystem Testing

Testing of this subsystem was accomplished during system and integrated testing of the other subsystems of the Sequential System.

2.8.4.4 Power Control Subsystem Mission Results

At 9 minutes, 53.77 seconds both Sequential Buses were armed as a result of OWS switch selector action. Since the relays were in parallel, redundant operation could not be verified. At 11 days, 23 hours, 32 minutes, the Sequential Buses were turned off by the crew. The Sequential Bus arming circuits operated as planned with no hardware failures. Normally the buses were to be turned off on day two by the crew, but since the meteoroid shield problem delayed the launch of SL-2 until 10 days later, the Sequential buses were left on for an extended period of time. Sequential bus loads were designed so that no degradation of any component would occur if the buses remained on. No degradation was detected.

The Deploy Buses were armed at 15 minutes, 33.64 seconds by the primary OWS switch selector command. The buses were planned to be disarmed at 3 hours, 11 minutes, 28.59 seconds. They were actually disarmed at 3 hours, 3 minutes, 55.42 seconds to prevent the switch selector commands from activating the ATM Thermal System. The circuit operated as planned with no hardware failures. A backup arming circuit was available but was not used.

2.8.4.5 Power Control Subsystem Conclusions and Recommendations

A. Conclusions - The power control circuits were proven to be adequate during the SL-1 flight. The system operated with no anomalies.
B. Recommendations - No changes are recommended for this subsystem.

2.8.5 Radiator Shield Jettison/Refrigeration Subsystem Activation

The purpose of the OWS Refrigeration System (RS) was to provide freezers for food and urine, and chillers for food, urine and water. The RS was a low temperature thermal control system using a refrigerant fluid in a closed loop circuit which dissipated heat through an externally mounted radiator. The first of two sequential events required to activate the system was to expose the externally mounted radiator which had been covered to prevent damage from Stage II retro rocket plumes during separation. The second event was to enable the RS.

2.8.5.1 OWS Radiator Shield Jettison/Refrigeration Subsystem Activation Design Requirements

The original design which remained basically unchanged - except for the addition of AM CRDU isolation relays - required an enable and disable control for both the primary and secondary RS. The initial primary activation control was provided by the IU/OWS switch selector system, while the AM CRDU provided initial activation backup control and primary control after the useful life of the switch selector system had expired. The addition of the RS radiator protective shield required a jettison system which used a command from the IU/OWS switch selector system and Sequential Bus power to control the PCS actuation control modules. Cycling of these modules was to allow gaseous nitrogen to release a pip pin type mechanism which would cause a preloaded spring to release and jettison the shield. AM CRDU provided a backup jettison command.

2.8.5.2 OWS Radiator Shield Jettison/Refrigeration Subsystem Activation Description

The OWS relays converted the momentary IU/OWS switch selector command into a long duration signal which cycled the actuation control modules, Figure 2.8-24. Actuation of these modules permitted gaseous nitrogen to jettison the radiator shield, Figure 2.8-25. The AM CRDU provided a backup jettison command.

The initial activation of the RS was provided by the IU/OWS switch selector system, Figure 2.8-26. The AM CRDU provided backup activation control and primary enable/disable control of the RS after the useful life of the switch selector had expired.

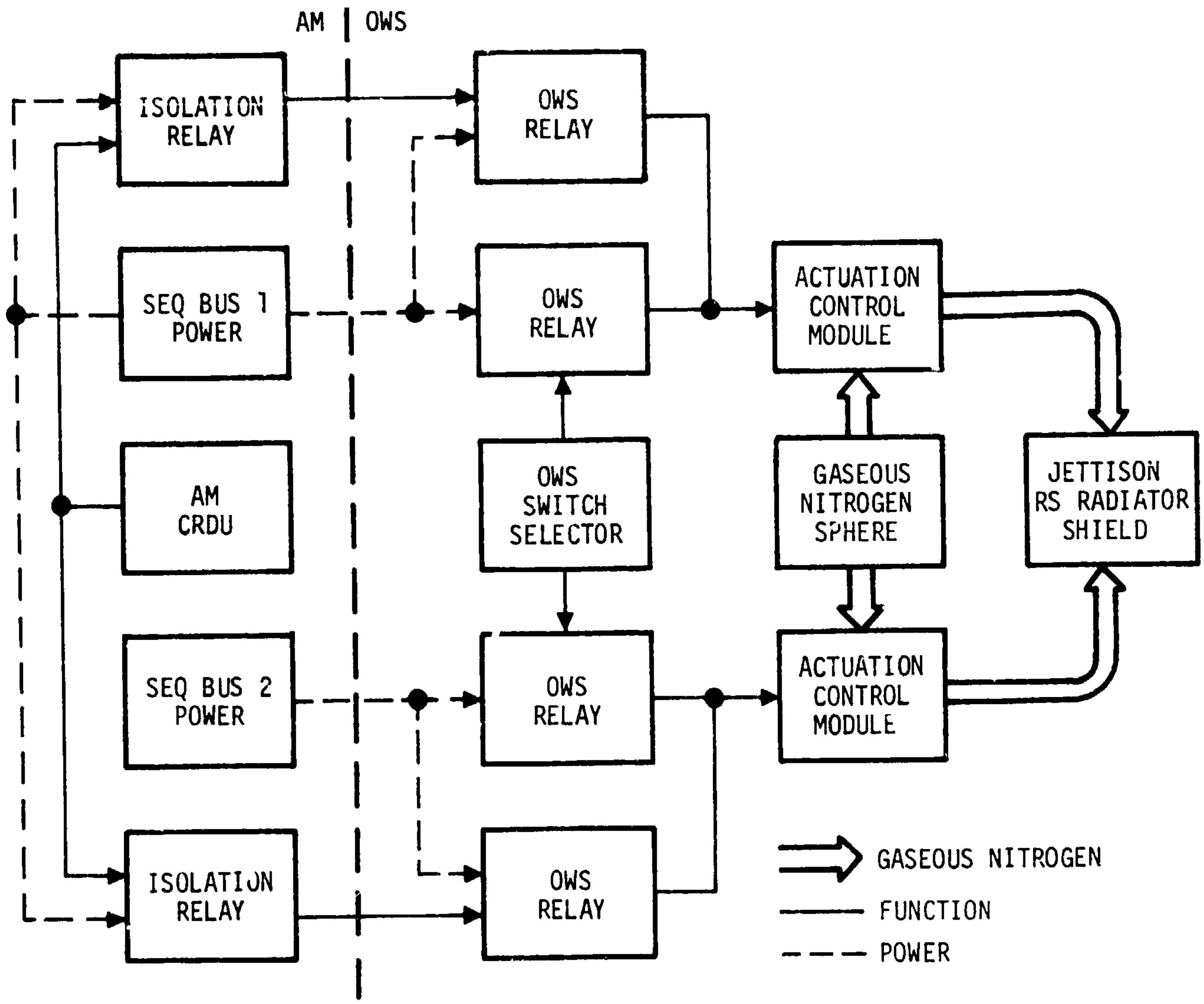


FIGURE 2.8-24 REFRIGERATION SYSTEM RADIATOR SHIELD JETTISON DIAGRAM

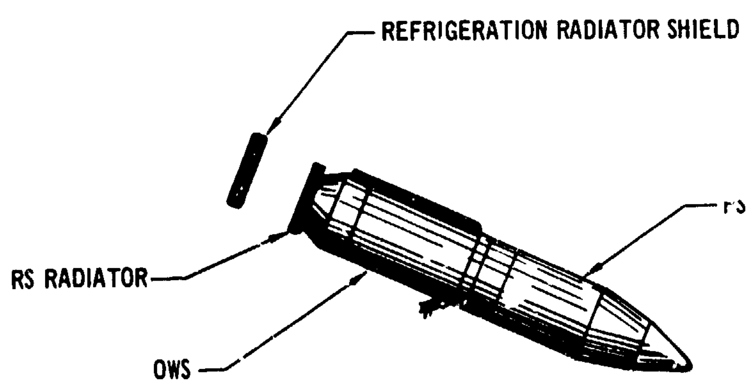


FIGURE 2.8-25 REFRIGERATION SYSTEM RADIATOR SHIELD JETTISON

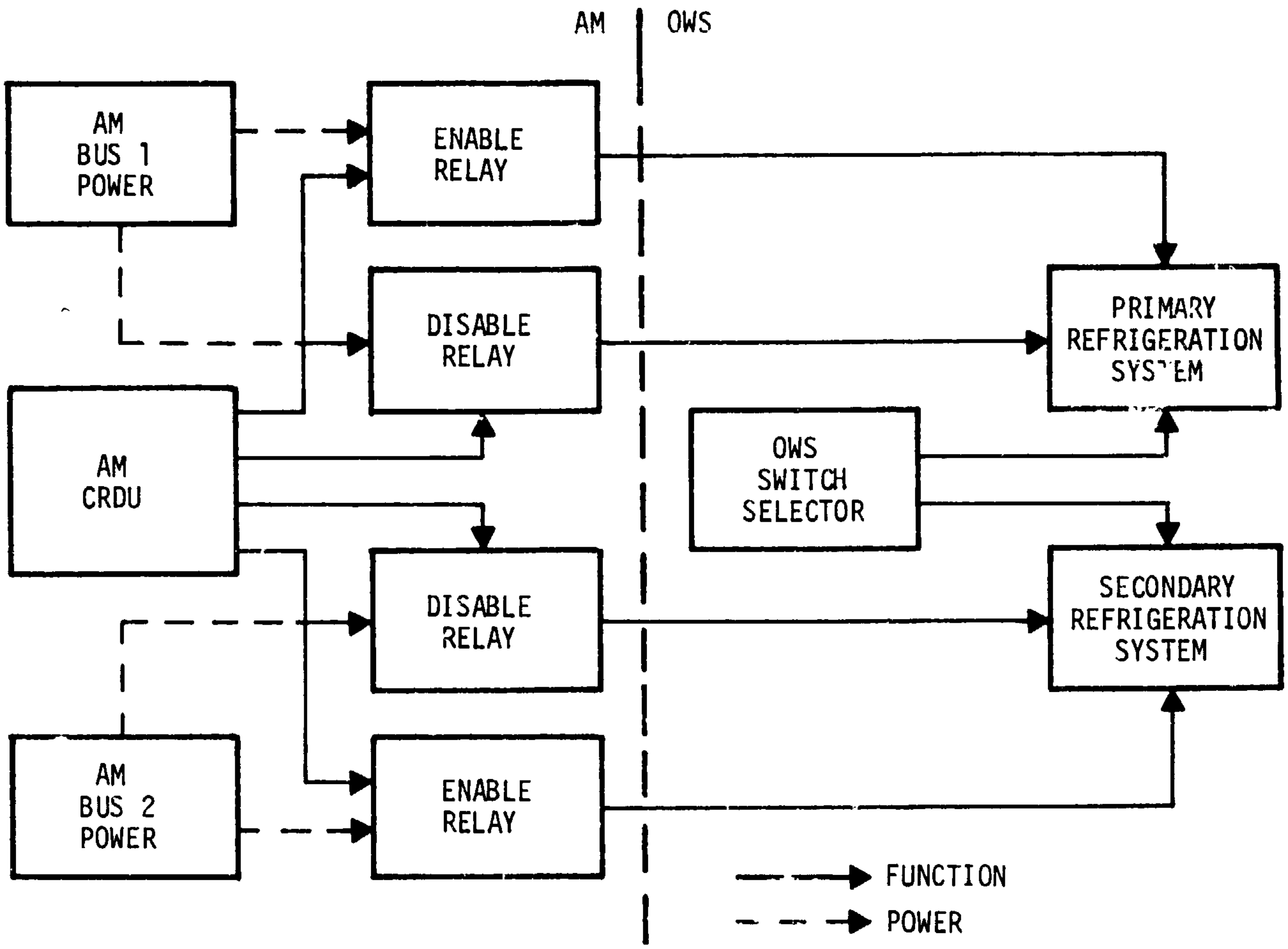


FIGURE 2.8-26 REFRIGERATION SYSTEM CONTROL DIAGRAM

2.8.5.3 <u>OWS Radiator Shield Jettison/Refrigeration Subsystem Activation Testing</u>

Testing of this subsystem was accomplished during the integrated testing with the OWS at KSC.

2.8.5.4 <u>OWS Radiator Shield Jettison/Refrigeration Subsystem Mission Results</u>

The OWS refrigeration radiator cover jettison circuits reacted to the OWS switch selector command which was issued at 9 minutes 55.42 seconds and jettisoned the radiator shield. No direct telemetry readout for shield jettison existed so the gradual decaying temperature of the radiator surface, Figure 2.8-27, served as the initial indication that the shield was jettisoned. The AM CRDU jettison command was issued as a precautionary measure but was not necessary as the electrical jettison system operated as planned. At 10 minutes 7.42 seconds the

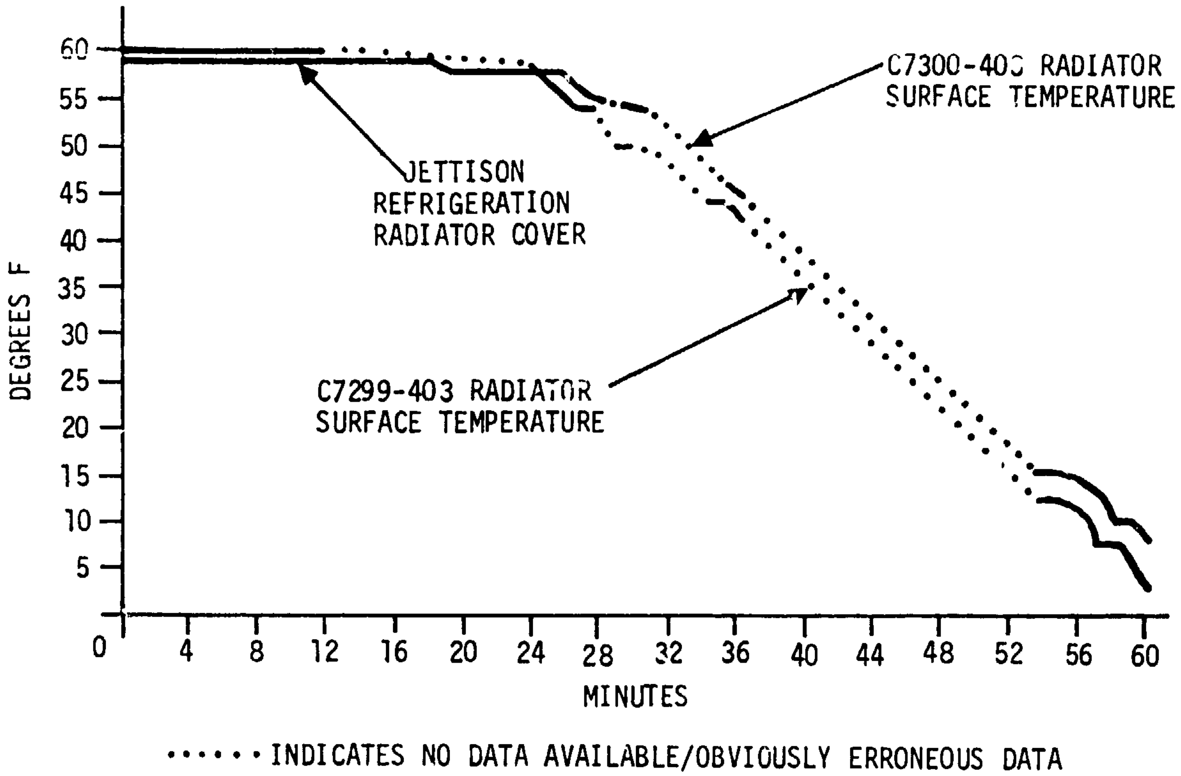


FIGURE 2.8-27 OWS REFRIGERATION RADIATOR TEMPERATURE

OWS refrigeration system was activated by the OWS switch selector. This was accomplished by utilizing the primary system which was contained in the OWS. The backup AM activation system was not utilized.

2.8.5.5 Radiator Shield Jettison/Refrigeration Subsystem Activation Conclusions and Recommendations

A. Conclusions - The OWS refrigeration shield jettison configuration was proven to be adequate during the SL-1 flight. The system operated as planned with no anomalies.

B. Recommendations - On SL-1 it took approximately 15 minutes to initially determine that the shield was jettisoned, Figure 2.8-27, and an additional 15 minutes before it could be reasonably verified. The telemetry data was erratic and caused sufficient uncertainty among the flight controllers to warrant the issuing of the backup jettison commands. The addition of telemetry indication of the shield jettison would have eliminated this uncertainty and would have afforded the flight controllers with a more reliable measurement of the shield's status.

2.8.6 OWS Venting Subsystem

The OWS was pressurized to aid in maintaining structural integrity during the initial powered flight phase. As the ascent loads diminished the pressurization loads were reduced by venting the OWS habitation area and the waste tank. The depressurized habitation area allowed a controlled habitable atmosphere to be added. The continuously vented waste tank allowed various waste products to outgass without pressurizing the tank. After the PCS usefulness had expired, the pneumatic (GN_2) sphere was depressurized. Automatic control of the vents was provided except for the solenoid vent valves which had JSC flight controller manual control capability only.

2.8.6.1 OWS Venting Subsystem Design Requirements

The OWS habitation area and waste tank were pressurized to about 23 psia prior to lift off to aid in maintaining structural integrity through launch and ascent. The PCS actuated the vents used during initial activation. The actuation control modules in the PCS were controlled by the electrical system. One set of modules when powered, opened the parallel habitation area vent valves. When power was removed the vent valves closed. The waste tank in a similar fashion utilized the PCS to open the waste tank vents. The vents once opened remained opened. The final sequence in the PCS was to vent the GN_2 pneumatic sphere. The electrical system operated a pneumatic dump valve which depressurized the sphere. The solenoid vent valves, arranged in a series-parallel combination and controlled by the AM-CRDU, provided a backup habitation area venting system during initial activation and they provided the primary venting system after initial activation.

2.8.6.2 OWS Venting Subsystem Description

A. OWS Habitation Area Vents - The IU/OWS switch selector system utilized AM Bus power to control the PCS actuation control modules, Figure 2.8-28. The open commands cycled the actuation control modules to allow the pneumatic pressure to open the vent valves. Issuing the closed commands effected the removal of pneumatic pressure and the closure of the valves. Parallel vent valves were used so both valves had to cycle to stop venting. The crew had the capability of capping the vent if the valves failed to close.

B. OWS Waste Tank Vents - The IU/OWS switch selector system utilized AM Bus power to control the PCS actuation control modules, Figure 2.8-29.

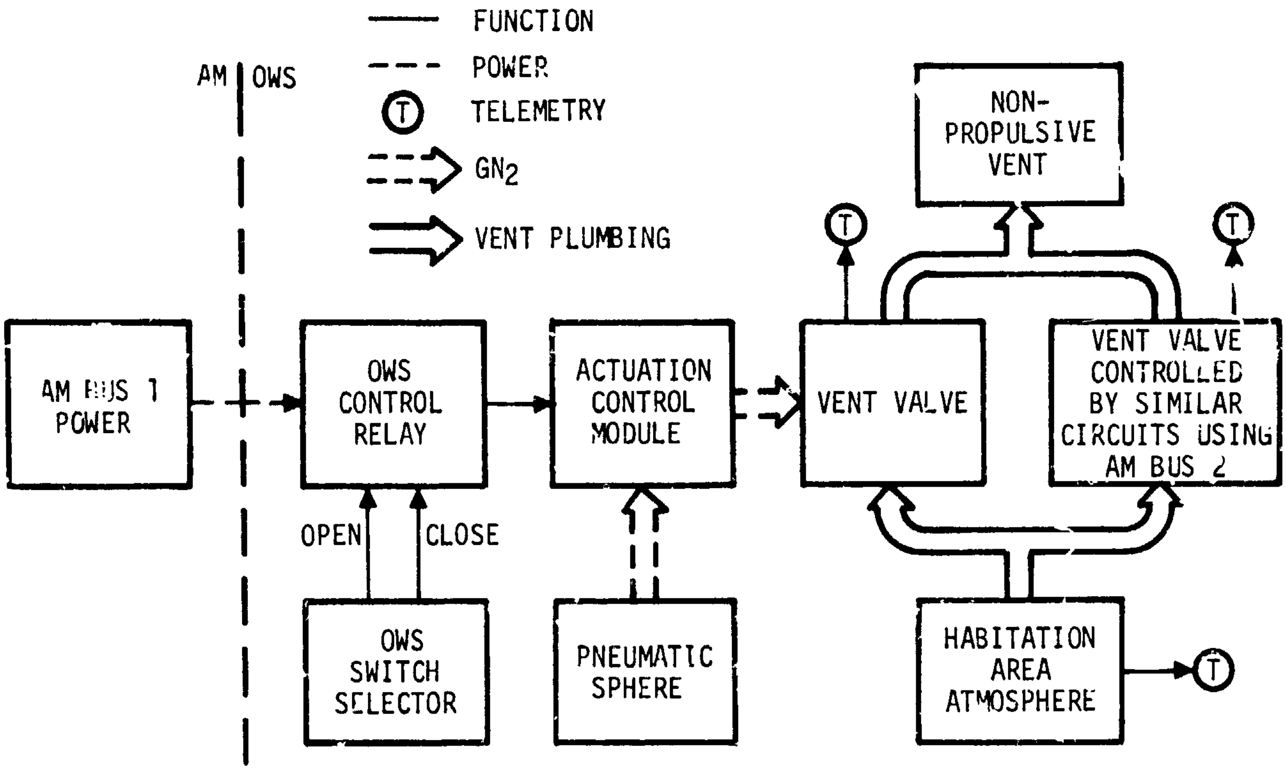


FIGURE 2.8-28 OWS HABITATION AREA VENT VALVES

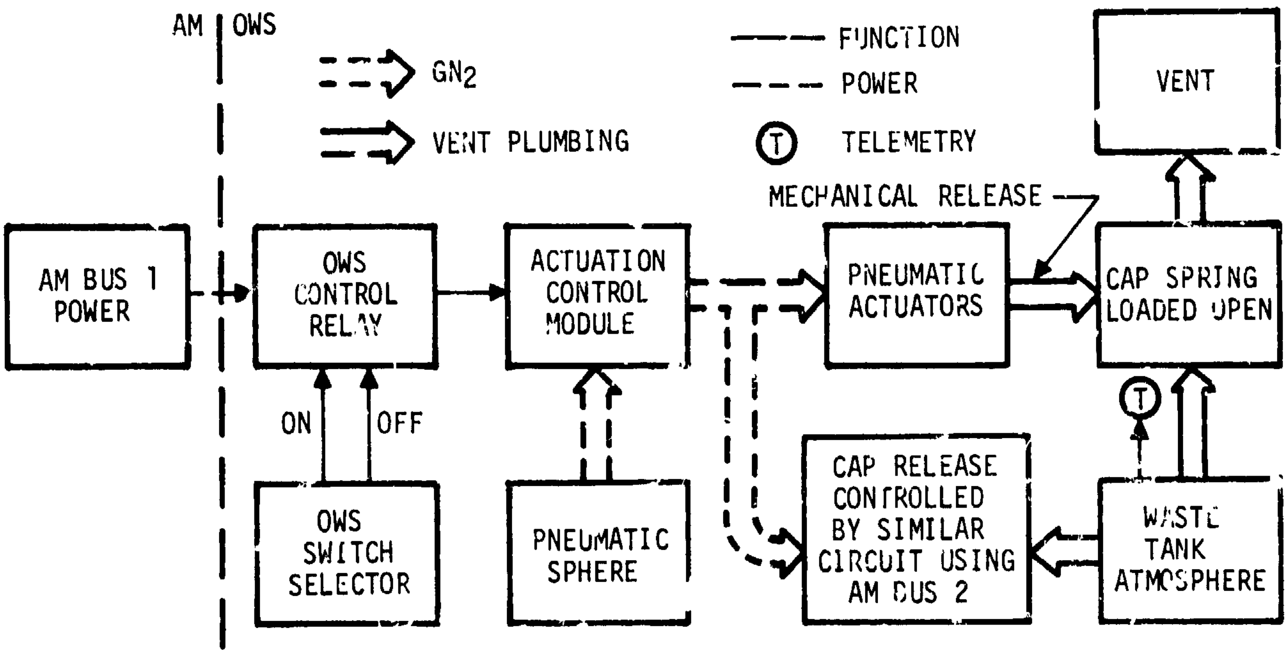


FIGURE 2.8-29 OWS WASTE TANK VENTS

The on commands cycled the actuation control modules, allowing the pneumatic pressure to cycle actuators which released the spring loaded caps opening the vents. The off commands closed the actuation control modules, removing pneumatic pressure from the actuators. The vents remained open. There were two sets of commands, relays and actuation control modules. The actuation control modules were in parallel requiring one module operation to cycle the redundant actuators releasing both vent caps.

C. OWS Pneumatic Sphere Dump - The switch selector system utilized AM Bus 2 power to open the pneumatic dump valve, which depressurized the pneumatic sphere, Figure 2.8-30. After the sphere was depressurized to less than 50 psia, a closed command was issued stopping the depressurization.

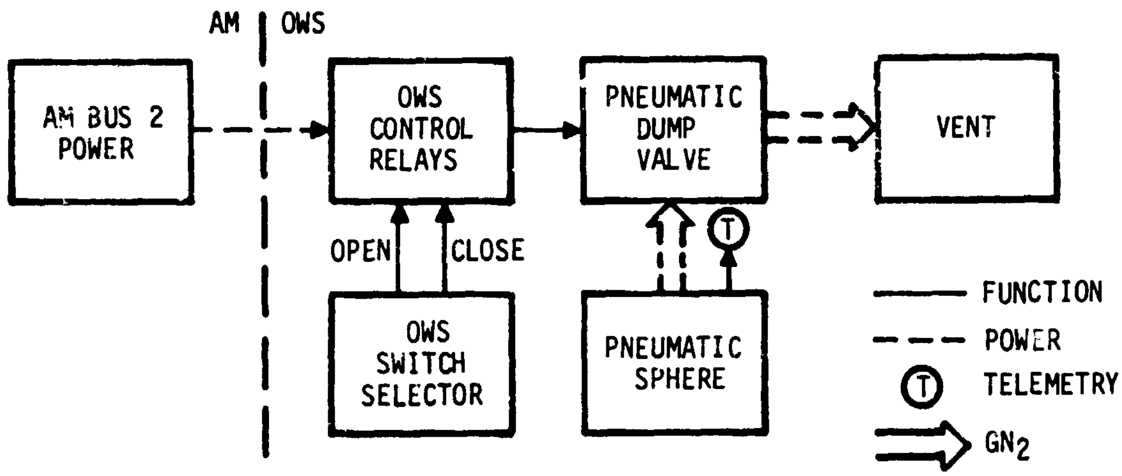


FIGURE 2.8-30 OWS PNEUMATIC SPHERE DUMP

D. OWS Solenoid Vent Valves - The AM CRDU provided two sets of commands to open and close the solenoid vent valves, Figure 2.8-31. One command set utilizing AM Bus 1 power controlled the open function of valves 1 and 2 and the closed function of valves 1 and 3. The other command set utilizing AM Bus 2 power controlled the corresponding functions of valves 2, 3 and 4. Either command set was capable of opening and closing a sufficient number of valves to start and stop venting. The crew had the capability of capping the vent if the valves failed to close.

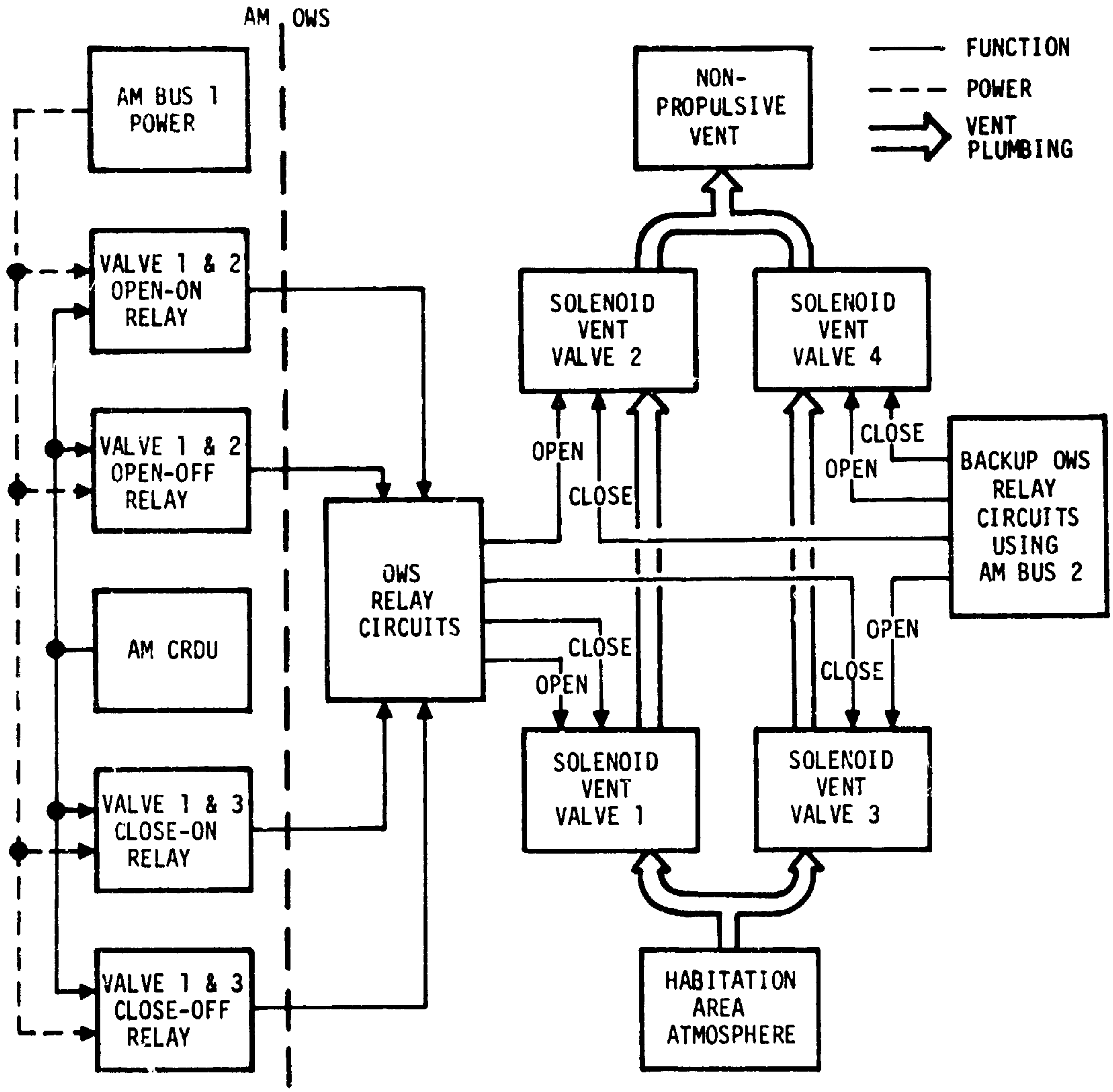


FIGURE 2.8-31 OWS SOLENOID VENT VALUES (HABITATION AREA)

2.8.6.3 OWS Venting Subsystem Testing

Testing of this subsystem was accomplished during the integrated testing with the OWS at KSC.

2.8.6.4 OWS Venting Subsystem Mission Results

A. OWS Habitation Area Vents - Prior to lift off the OWS habitation area was pressurized to 23 psia. The vent valves reacted to OWS switch selector commands at 3 minutes 24.69 seconds and at 3 minutes 24.89 seconds respectively by changing state from fully closed to fully closed-off. At 3 minutes 24.79 seconds and 3 minutes 25.09 seconds the valves indicated open. OWS habitation area pressure decay started at this time, Figure 2.8-32. At 41 minutes 19.42 seconds the vent valves closed and the pressure decay stopped. This indicated that the redundant AM circuits functioned normally with no hardware anomalies.

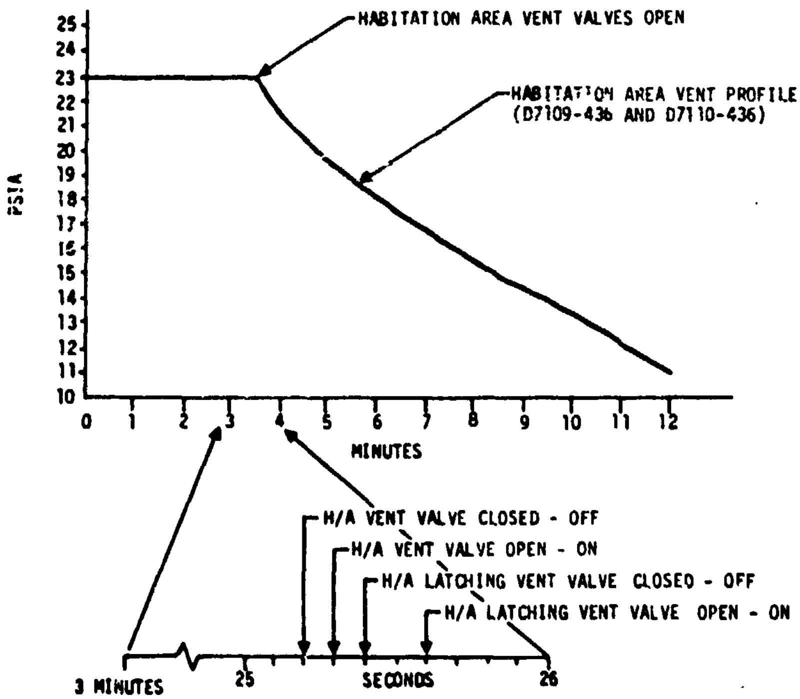


FIGURE 2.8-32 OWS HABITATION AREA VENT

B. OWS Waste Tank Vents - Prior to lift off the OWS waste tank was pressurized to about 23 psia. The vents were commanded open at 9 minutes 54.42 seconds. The OWS waste tank pressure started to decay just prior to 10 minutes, Figure 2.8-33. The redundant electrical operation of the pneumatic vent valves could not be verified since the valves did not have individual telemetry indicators and the pneumatic system paralleled the electrical operated pneumatic controls. Apparently the electrical system operated as planned.

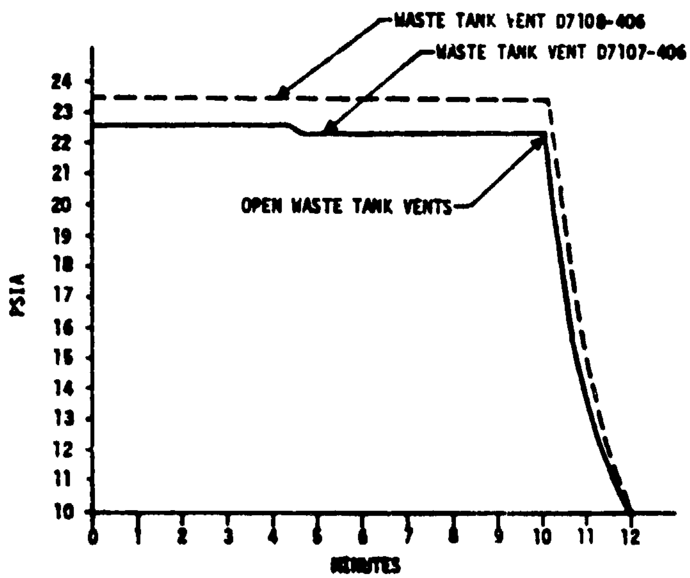


FIGURE 2.8-33 OWS WASTE TANK VENT

C. Pneumatic Sphere Dump - Prior to lift-off the OWS pneumatic sphere was pressurized to about 500 psia. At 5 hours 21 minutes 59.42 seconds the OWS switch selector commanded the pneumatic sphere dump. The pressure decayed indicating that the event had occurred, Figure 2.8-34. At 8 hours 14 minutes 59.42 seconds the pneumatic dump was terminated. Pneumatic residual pressure was about 40 psia.

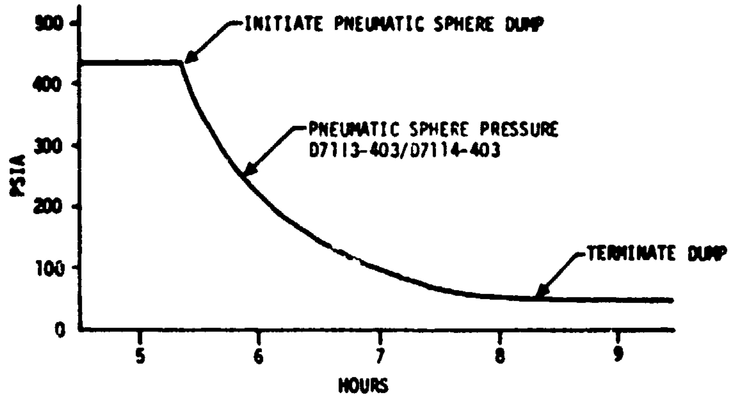


FIGURE 2.8-34 PNEUMATIC SPHERE DUMP

2.8.6.5 <u>OWS Venting Subsystem Conclusions and Recommendations</u>

A. Conclusions - The OWS venting circuits were proven to be adequate during the SL-1 flight. The system operated as planned with no anomalies.

B. Recommendations - No changes are recommended for this subsystem.

2.8.7 OWS Meteoroid Shield Deployment Subsystem

The purpose of deploying the OWS meteoroid shield was to reduce the probability of micrometeoroid penetration of the OWS habitation area. The shield also was to provide thermal protection for the habitation area. The AM electrical sequential system was required to provide a backup deployment control system and to provide power for the primary and backup deployment systems. Automatic and JSC flight controller manual operating capabilities were included in the design. The AM hardware utilized in the design was selected from flight qualified equipment.

2.8.7.1 OWS Meteoroid Shield Deployment Subsystem Design Requirements

The OWS meteoroid shield deployment circuit design was initiated when the early mechanical/ordnance trade off studies were completed. The IU/OWS switch selector system was chosen to cycle an EBW firing unit/EBW detonator/confined detonating fuse (CDF) which ignited a mild detonating fuse (MDF) in an expandable tube. This expanded tube sheared the straps and allowed the preloaded torsion bars to rotate and deploy the shield. The AM CRDU had the capability of cycling a separate EBW firing unit, MDF and expandable tube to provide a backup method of shearing the straps and deploying the shield.

2.8.7.2 OWS Meteoriod Shield Deployment Subsystem Description

The OWS meteoroid shield deployment circuit utilized Sequential Bus 2 power and the IU/OWS switch selector system to charge and trigger the primary EBW firing unit, Figure 2.8-35. This action resulted in the deployment of the shield. A backup method of deploying the meteoroid shield was provided by using Sequential Bus 1 power and the AM CRDU command system to cycle the backup EBW firing unit.

2.8.7.3 OWS Meteoroid Shield Deployment Subsystem Testing

Electrical testing of this subsystem was accomplished during the integrated testing with the OWS at KSC.

2.8.7.4 OWS Meteoroid Shield Deployment Subsystem Mission Results

Approximately 63 seconds after lift off, telemetry readouts indicated that the meteoroid shield structure failed and the shield was torn away from the OWS. The primary and backup deployment commands were issued at 1 hour, 36 minutes

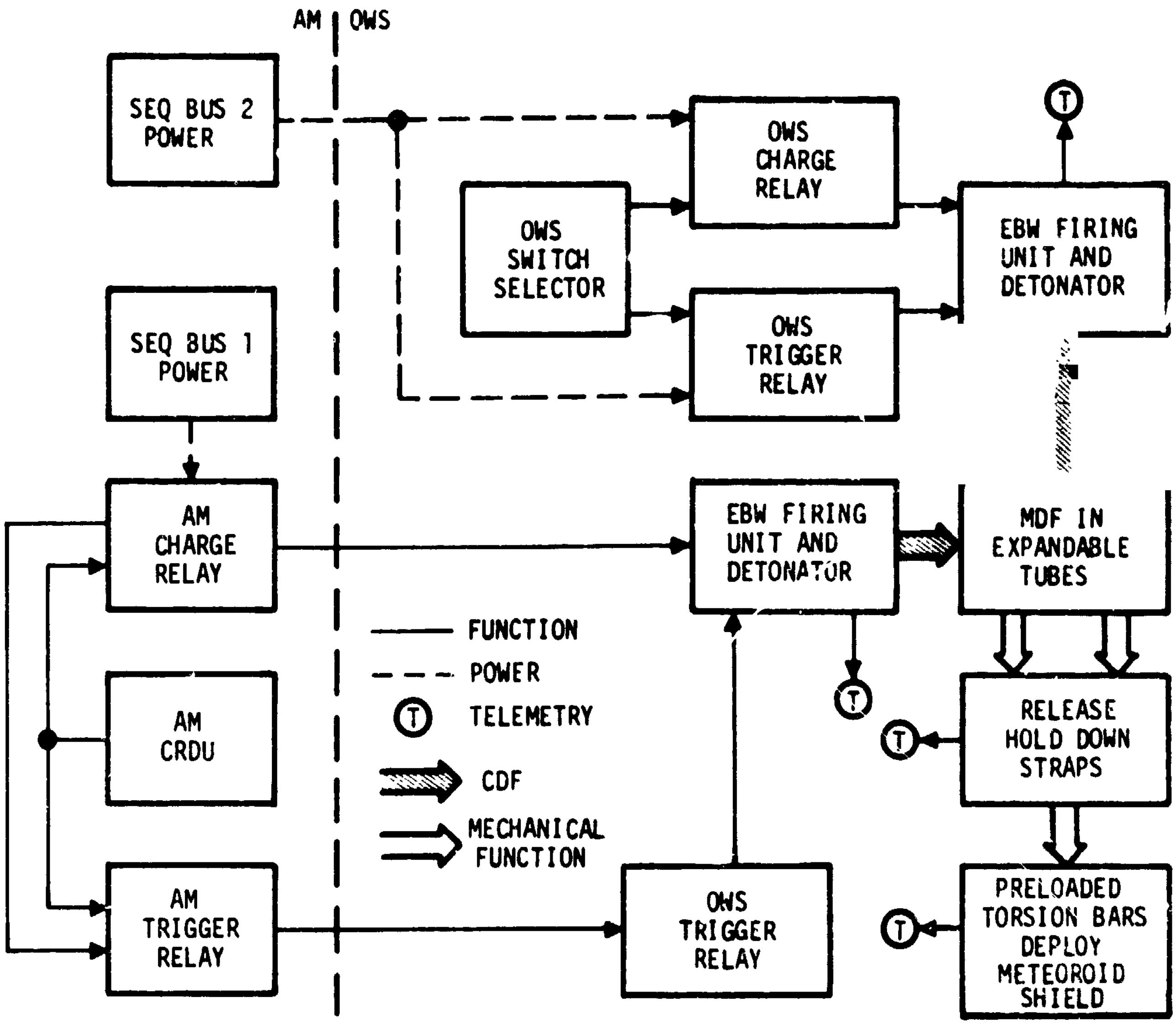


FIGURE 2.8-35 METEOROID SHIELD DEPLOYMENT

3.52 seconds and at 2 hours, 42 minutes, 29.42 seconds respectively. The electrical sequential system responded as expected to these commands even with a missing meteoroid shield. Subsequent analysis indicated that the shield was actually gone and cycling the electrical system only affirmed that the electrical system had cycled the firing units.

2.8.7.5 <u>OWS Meteoroid Shield Deployment Subsystem Conclusions and Recommendations</u>

A. Conclusions - Although the shield had been torn away, the OWS meteoroid shield deployment circuits were shown to be adequate.

B. Recommendations - No electrical changes are recommended on this subsystem.

2.8.8 OWS SAS Deployment Subsystem

The OWS solar array system (SAS) was folded and stored on opposite sides of the OWS exterior during ascent. The SAS consisted of two wing assemblies which unfolded upon receiving automatic or JSC flight controller backup commands. The OWS SAS was the power source for the AM electrical power system (EPS), Section 2.7. The AM hardware utilized in the design was selected from flight qualified equipment.

2.8.8.1 OWS SAS Deployment Subsystem Design Requirements

The initial mechanical/ordnance trade off studies resulted in OWS mounting of the SAS using an EBW ordnance system for initiating deployment. The first of two sequential events required for OWS SAS deployment was the release of the beam fairings which protected the solar arrays during ascent and separation. The electrical sequential system cycled an EBW firing unit which detonated an EBW detonator, and caused a confined detonating fuse (CDF) to burn resulting in the igniting of a mild detonating fuse (MDF). As it burned, the MDF caused the tube in which it was contained to expand and shear the beam fairing hold down straps. Preloaded springs in the hinge joints caused the beam fairings to rotate to the deployed position (90 degrees from the X axis). The second event - releasing the folded solar wings - used an ordnance system similar to the one described above except that tension straps were employed in place of the hold down straps. Preloaded springs in the hinge joints caused the wings to deploy. The IU/OWS switch selector system was designated as the primary command system with the AM CRDU providing the backup commands.

2.8.8.2 OWS SAS Deployment Subsystem Description

The OWS beam fairing deployment circuits utilized Deploy Bus 2 power and the IU/OWS switch selector system to charge and trigger the primary EBW firing unit, Figure 2.8-36. This combined action resulted in the deployment of the OWS beam fairings. A backup method of deployment, using Deploy Bus 1 power and the AM CRDU command system to cycle the backup EBW firing unit, was also available. The OWS wing deployment circuit cycled the primary wing EBW firing unit ordnance, Figure 2.8-37, in a fashion similar to the beam fairing deployment circuit. In this case, however, the beam fairing trigger relay acted as an interlock to prevent the wings from being released prior to the beam fairing deployment.

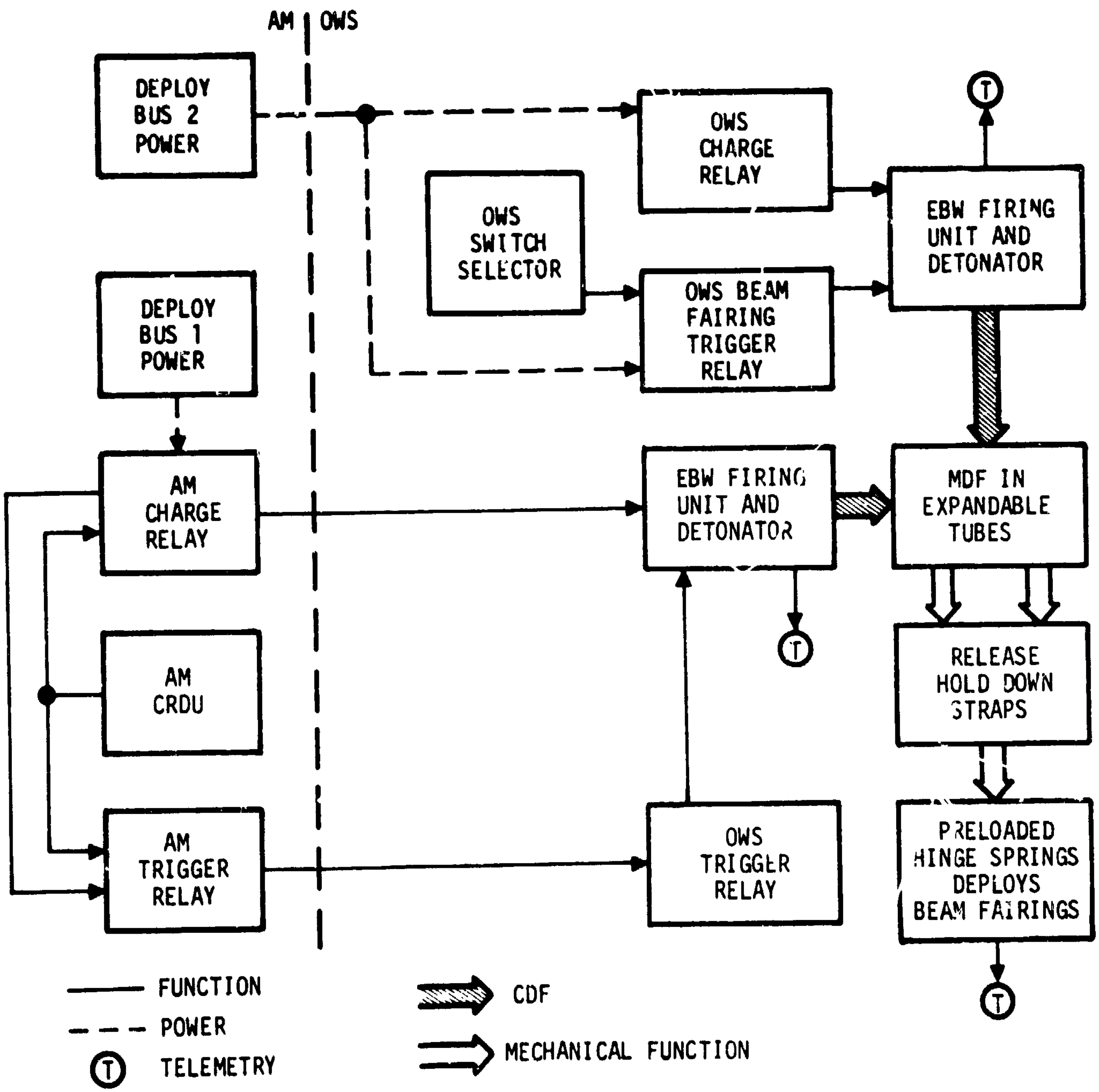


FIGURE 2.8-36 OWS BEAM FAIRING DEPLOYMENT

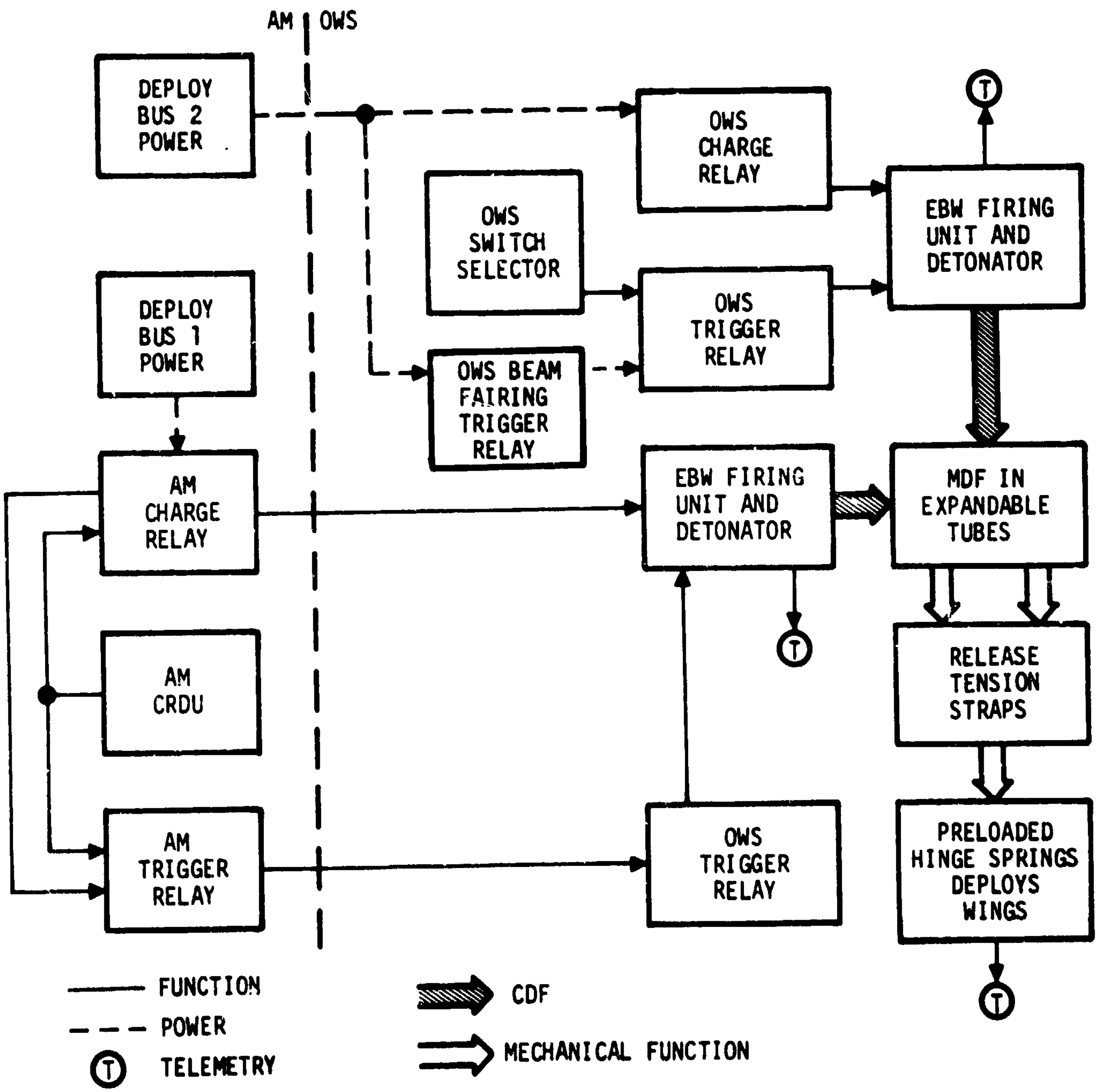


FIGURE 2.8-37 OWS WING DEPLOYMENT

The AM CRDU bypassed the beam fairing interlock and provided backup wing deployment commands.

2.8.8.3 OWS SAS Deployment Subsystem Testing

Electrical testing of this subsystem was accomplished during the integrated testing with the OWS at KSC.

2.8.8.4 OWS SAS Deployment Subsystem Mission Results

During the ascent phase of the flight the OWS meteoroid shield was torn off. This caused the OWS beam fairing number 2 hold down straps to break and thus free the beam fairing which was forced to deploy during retro rocket firing during stage II separation. The beam fairing evidently rotated to the deployed position with a force that was sufficient to shear the hinge joint and tear the beam fairing off. When the meteoroid shield tore off, a piece of the shield structure embedded into beam fairing number 1 preventing it from deploying. The primary OWS SAS beam fairing and OWS wing deployment commands were issued at 41 minutes, 5.32 seconds and at 51 minutes 59.42 seconds respectively. At 55 minutes 59.42 seconds telemetry indications showed that the hold downs released. Due to the restriction of the meteoroid shield structure, full beam fairing deployment was not obtained. The AM CRDU backup OWS SAS beam fairing and OWS wing deployment commands were issued at 1 hour 38 minutes 21.42 seconds and at 1 hour 50 minutes 55.42 seconds respectively. The backup system released the wings but was still unsuccessful in deploying the beam fairing. The cycling of the primary and backup systems did prove that the electrical sequential system functioned as planned with no hardware failures. The SL-2 crew performed a stand up extra-vehiclar activity (SEVA) from the CSM to deploy beam fairing number 1. This attempt was also unsuccessful. Twenty four days after SL-1 liftoff, the SL-2 crew performed an EVA and successfully deployed the beam fairing. Subsequently beam fairing No. 1 wings were fully deployed.

2.8.8.5 OWS SAS Deployment Subsystem Conclusions and Recommendations

A. Conclusions - The OWS SAS electrical deployment circuits functioned as planned during the SL-1 flight with no anomalies.

B. Recommendations - No change to the electrical design is recommended.

2.8.9 ATM SAS Deployment/Canister Release Subsystem

The ATM solar array system (SAS) and the ATM canister were restrained to protect them from the structural loading generated during the powered flight phase and the deployment of the ATM. After the ATM deployment was complete, the ATM SAS was deployed, providing the power source for the ATM electrical power system. The SAS deployment circuits also released the canister. Automatic and JSC flight controller manual operating capabilities were included in the requirement.

2.8.9.1 ATM SAS Deployment/Canister Release Subsystem Design Requirements

The ATM SAS deployment initially used the IU/OWS switch selector system to provide a decinching function to release the wings and a motor control function to deploy the wings. The decinching function utilized EBW firing units and detonators to ignite CDF's. The CDF's ignited pressure cartridges - actuating the thruster assemblies - resulting in the rotation of torque tubes allowing ball end rods to slip out of key hole slots in the torque tubes. This action freed the wings. The deployment motors were cycled on and the wings which were folded in scissors style, were pushed out as the motors reeled in cables closing the scissors mechanisms. When the wings reached their fully deployed position, the latches secured the wings and the limit switches turned the motors off. The AM CRDU was available as a backup command system. An inhibit circuit was added to the automatic system to prevent SAS deployment before the ATM was fully deployed. Isolation relays were added to the AM CRDU circuits. A launch pad monitor was added to indicate inadvertent arming of the ATM EBW circuits. An AM CRDU command inhibit circuit was added to prevent igniting the ATM ordnance on the launch pad. The ATM canister release function was paralleled off the ATM SAS decinching relay circuits.

2.8.9.2 ATM SAS Deployment/Canister Release Subsystem Description

The ATM deployment limit switches, section 2.8.2.2, sensed full deployment of the ATM and latched in the SAS enable relays. These relays applied Deploy Bus power to the OWS relays in the IU/OWS switch selector command system, Figure 2.8-38. The switch selector commands, Figure 2.8-39, charged and triggered the SAS decinching and canister release EBW firing units. This action freed the wings and released the canister. Subsequent commands energized

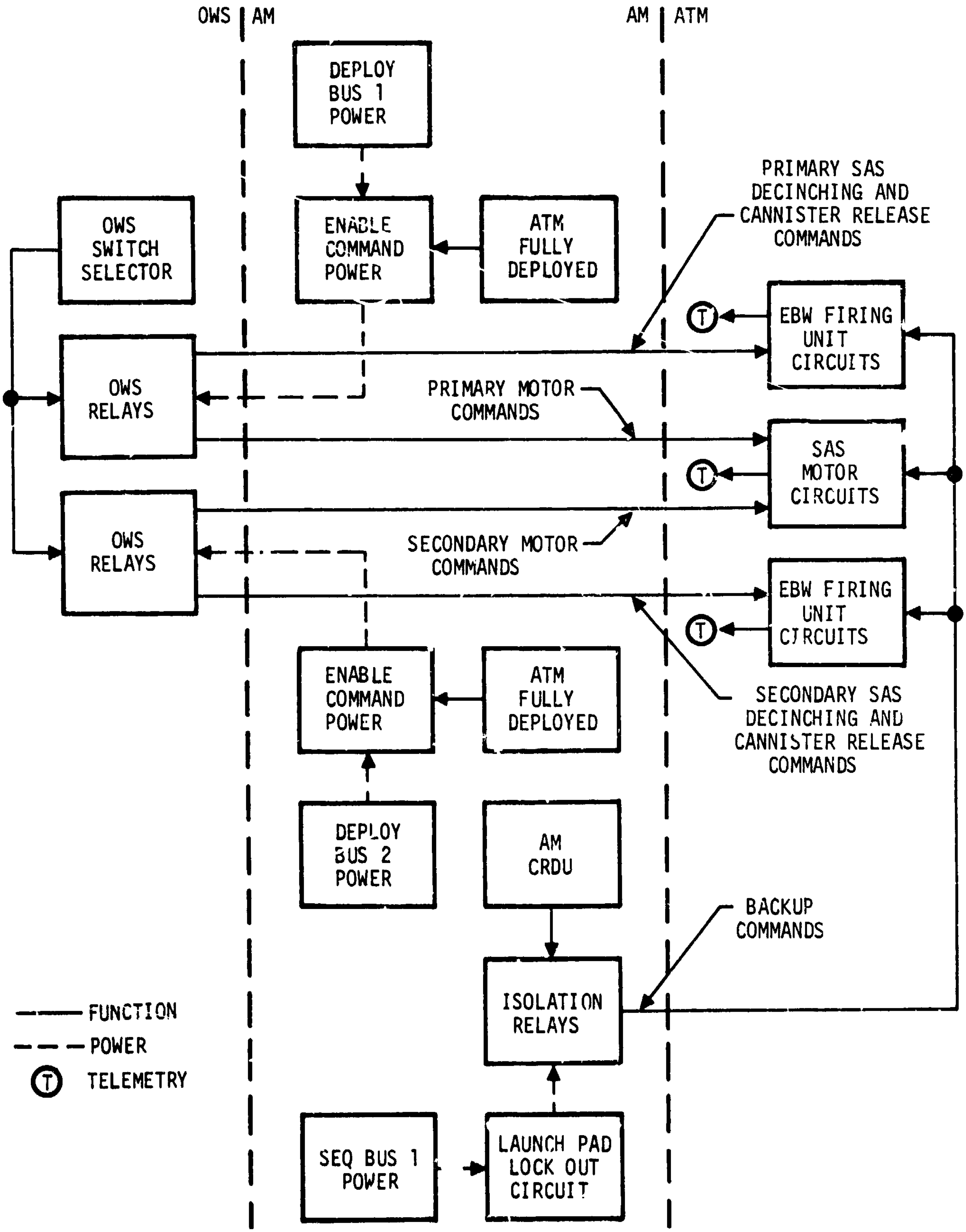


FIGURE 2.8-38 ATM SAS DEPLOYMENT/ATM CANISTER RELEASE

OWS SWITCH SELECTOR		AM CRDU	FUNCTION	TELEMETRY
PRIMARY COMMAND	SECONDARY COMMAND	BACKUP COMMAND		
CHARGE EBW SYSTEM "A"	NONE	CHARGE EBW SYSTEM "A" AND "B"	ATM-SAS DECINCHING AND CANISTER RELEASE FIRING UNITS "A" CHARGE	AM-SAS DECINCHING TWO ANALOG SIGNALS INDICATES CHARGE LEVEL
NONE	CHARGE EBW SYSTEM "B"		SAME AS ABOVE FOR SYSTEM "B"	
FIRE EBW SYSTEM "A"	NONE	FIRE EBW SYSTEM "A"	ATM-FIRES SYSTEM "A" FIRING UNITS RELEASING THE WINGS AND CANISTER	AM-ABOVE SIGNALS DROP TO ZERO MOMENTARILY
NONE	FIRE EBW SYSTEM "B"	FIRE EBW SYSTEM "B"	SAME AS ABOVE FOR SYSTEM "B"	
WINGS 1 AND 3 DEPLOY	WINGS 1 AND 3 DEPLOY	WINGS 1 AND 3 DEPLOY	ATM-ENERGIZES WINGS 1 AND 3 MOTORS AND DEPLOYS THE WINGS	AM-EIGHT BILEVELS INDICATE DEVELOPMENT STATUS
WINGS 2 AND 4 DEPLOY	WINGS 2 AND 4 DEPLOY	WINGS 2 AND 4 DEPLOY	SAME AS ABOVE FOR WINGS 2 AND 4	
SYSTEM "A" AND WING DEPLOY 1 RESET	SYSTEM "A" AND WING DEPLOY 1 RESET	NONE	ATM-RESETS SWITCH SELECTOR ACTIVATED SYSTEM "A" AND ONE SET OF WING DEPLOYMENT CIRCUITS	AM-ABOVE ANALOG SIGNALS DROP TO ZERO. BILEVELS INDICATE ONE.
SYSTEM "B" AND WING DEPLOY 2 RESET	SYSTEM "B" AND WING DEPLOY 2 RESET	NONE	SAME AS ABOVE FOR SYSTEM "B" AND THE OTHER SET OF CIRCUITS	
NONE	NONE	RESET SYSTEM "A" AND "B"	ATM-RESETS AM CRDU ACTIVATED SYSTEM "A" AND "B".	AM-ABOVE ANALOG SIGNALS DROP TO ZERO

FIGURE 2.8-39 ATM SAS/CANISTER – COMMANDS/FUNCTIONS

the motor circuits which deployed the SAS wings. The AM CRDU bypassed the ATM deployment limit switches and provided backup commands for decinching, release and deployment functions.

2.8.9.3 ATM SAS Deployment/Canister Release Subsystem Testing

Electrical testing of this subsystem was accomplished during the integrated testing with the ATM at KSC.

2.8.9.4 ATM SAS Deployment/Canister Release Subsystem Mission Results

At 24 minutes, 48.42 seconds the command sequence was initiated resulting in the decinching of the SAS, release of the canister and deployment of the wings. The system operated as planned with no hardware failures. The AM CRDU backup command system was not utilized.

2.8.9.5 ATM SAS Deployment/Canister Release Subsystem Conclusions & Recommendations

A. Conclusions - The ATM SAS deployment and canister release circuits were proven to be adequate during the SL-1 flight. The system operated as planned with no failures.

B. Recommendations - No electrical changes are recommended on this subsystem.

2.8.10 ATM Activation Subsystem

ATM activation circuits were to initially activate various ATM systems, and after activation, provide control of selected ATM functions. The sequential system was required to operate in conjunction with the IU/OWS switch selector system, the AM CRDU and the AM power system. Automatic and JSC flight controller manual operating capabilities were included in the requirement. The hardware utilized in the design was selected from flight qualified equipment.

2.8.10.1 ATM Activation Subsystem Design Requirements

Initially the IU switch selector utilizing IU power was to provide the ATM activation commands with the AM providing interconnecting circuits between the IU and ATM. At completion of the early electrical/command system trade off studies, this design was altered to utilize AM power with the IU/OWS switch selector system for ATM activation. The AM CRDU was selected as a backup command source and the isolation relays were added to provide a greater command driving capability for the CRDU commands.

2.8.10.2 ATM Activation Subsystem Description

Figure 2.8-40 identifies the functions supplied by the sequential system for ATM activation and control. The IU/OWS switch selector system, Figure 2.8-2, was utilized except the AM provided an interconnecting circuit between the OWS and ATM so that commands were transfered directly to the ATM. This system provided the primary and secondary activation commands. The AM CRDU provided backup activation and control commands for the ATM, Figure 2.8-41.

2.8.10.3 ATM Activation Subsystem Testing

Electrical testing of this subsystem was accomplished during the integrated testing with the ATM at KSC.

2.8.10.4 ATM Activation Subsystem Mission Results

The command/sequential system performed as expected providing selected ATM control functions.

COMMAND SYSTEM		ATM FUNCTION
SWITCH SELECTOR	AM CRDU	
		ACTIVATION
OWS	NONE	THERMAL SYSTEM ON
OWS	BACKUP	APCS ON
OWS	NONE	TELEMETRY SYSTEM ON
		TRANSMITTER CONTROL
ATM	BACKUP	NO. 1 FORWARD ANTENNA SELECT
ATM	BACKUP	NO. 1 AFT ANTENNA SELECT
ATM	BACKUP	NO. 2 FORWARD ANTENNA SELECT
ATM	BACKUP	NO. 2 AFT ANTENNA SELECT
		TM MODULATION MODE SELECT
ATM	BACKUP	NO. 1
ATM	BACKUP	NO. 2
ATM	BACKUP	NO. 3
ATM	BACKUP	NO. 4
		DECODER AND RECEIVER CONTROL
ATM	BACKUP	NO. 1 POWER ON
ATM	BACKUP	NO. 1 POWER OFF
ATM	BACKUP	NO. 2 POWER ON
ATM	BACKUP	NO. 2 POWER OFF
		TAPE RECORDER CONTROL
ATM	BACKUP	NO. 1 RECORD
ATM	BACKUP	NO. 1 PLAYBACK
ATM	BACKUP	NO. 1 STOP
ATM	BACKUP	NO. 2 RECORD
ATM	BACKUP	NO. 2 PLAYBACK
ATM	BACKUP	NO. 2 STOP

FIGURE 2.8-40 ARM ACTIVATION/CONTROL

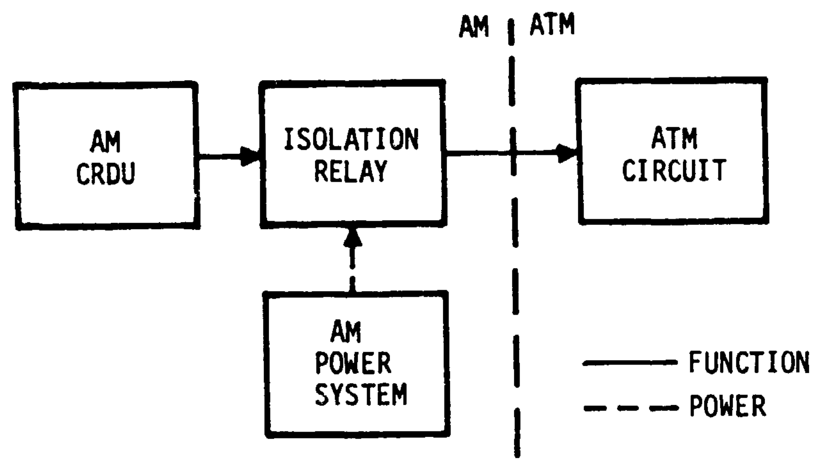


FIGURE 2.8-41 TYPICAL AM CRDU CIRCUIT

2.8.10.5 ATM Activation Subsystem Conclusions and Recommendations

A. Conclusions - The AM portion of the ATM activation circuits was proven to be adequate during the SL-1 flight. The system operated as planned with no anomalies.

B. Recommendations - No electrical changes are recommended on this subsystem.

2.8.11 MDA Venting Subsystem

MDA venting was required in order to dump the AM/MDA atmosphere in flight so a controlled oxygen/nitrogen gas mixture could be added before crew arrival. The electrical sequential system provided MDA vent valve control using the OWS switch selector command and AM power systems. JSC flight controllers had a partial backup capability by utilizing the IU CCS to re-issue switch selector commands.

2.8.11.1 MDA Venting Subsystem Design Requirements

Initially the venting of the MDA was to be accomplished by launching with the vent valves closed and then opening the valves after lift-off. The valves were in parallel to assure venting. When the pressure decayed to an acceptable level the valves were to be closed. The OWS switch selector provided primary commands to control the valves with AM CRDU providing backup commands. Later the operational procedure was changed to open the valves prior to lift-off via AM CRDU with the switch selector commanding the valves closed after the pressure decayed to an acceptable level. This operational mode required both valves to close to preclude a hard vacuum in the AM/MDA. Subsequent redesign put the valves in series with the valves being opened and verified before lift-off. This configuration required only one of the redundant switch selector commands to cycle one valve in flight to terminate venting. The AM CRDU commands were deleted.

2.8.11.2 MDA Venting Subsystem Description

The commands, Figure 2.8-42, for controlling the AM relays originated from the OWS switch selector. The switch selector issued these momentary commands to latch in OWS relays. These relays provided continuous commands to energize AM relays, using AM power, to cycle the MDA vent valves, Figure 2.8-43. The OWS and AM relay circuits were configured so that two commands were required to initiate the cycling of the valves. The MDA vent valves remained closed during launch pad operations to reduce the probability of contaminating the AM/MDA atmosphere. Just before lift-off the valves were commanded open to allow venting during ascent. An electrical solenoid in the valve released a mechanical brake allowing the motor to run and cycle the valve. When the cycle was completed limit switches removed power from the solenoid applying the brake

OWS SWITCH SELECTOR		FUNCTION	TELEMETRY
PRIMARY CMD	SECONDARY CMD		INDICATION
CLOSE ENABLE	CLOSE ENABLE	SELECTS VENT VALVE CLOSE CIRCUIT	NONE
EXECUTE	EXECUTE	APPLIES POWER TO THE OPEN/CLOSE CIRCUIT	CLOSED=ON OPEN=OFF *
EXECUTE RESET	EXECUTE RESET	REMOVES POWER FROM THE OPEN/CLOSE CIRCUIT	NONE
OPEN ENABLE	OPEN ENABLE	GROUND FUNCTION - SELECTS VENT VALVE OPEN CIRCUIT	NONE

* IF OPEN ENABLE WAS SELECTED THEN CLOSED = OFF OPEN = ON

FIGURE 2.8-42 MDA VENT VALVE FUNCTION

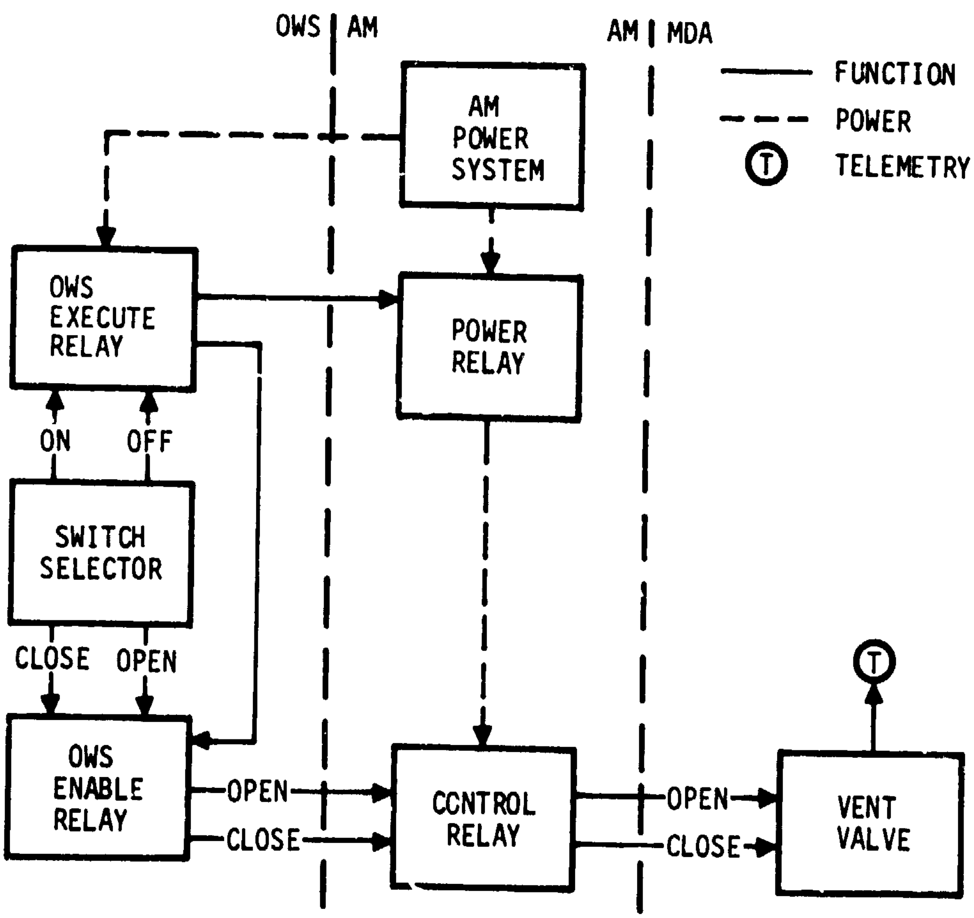


FIGURE 2.8-43 TYPICAL VENT VALVE CONTROL CIRCUIT

stopping the motor. The switch selector, after allowing more than adequate time to cycle the valve, commanded power off. Valve position was verified by telemetry before lift-off. After lift-off AM/MDA pressure was to decay to about 1 psia. At this time closed commands were issued cycling the valves and terminating venting. Valve position and MDA pressure was verified by telemetry.

2.8.11.3 MDA Venting Subsystem Testing

Electrical testing of this subsystem was accomplished during the integrated testing with the MDA and OWS at KSC.

2.8.11.4 MDA Venting Subsystem Mission Results

The MDA pressure was ambient (14.7 psia) when the MDA vent valves were opened prior to lift-off at KSC. During ascent MDA pressure decayed through the open vent valves, Figure 2.8-44. MDA vent valves 1 and 2 reacted to OWS switch selector commands at 4 minutes 39.97 seconds and 4 minutes 40.17 seconds respectively by changing state from fully open-on to fully opened-off. Valves 1 and 2 closed, 7.8 and 7.6 seconds later, respectively. MDA pressure stabilized indicating valve closures. This indicated that the redundant AM circuits functioned normally with no hardware failures.

2.8.11.5 MDA Venting Subsystem Conclusions and Recommendations

A. Conclusions - The AM/MDA venting configuration was proven to be adequate during the SL-1 flight.
B. Recommendations - No changes to the electrical design are recommended.

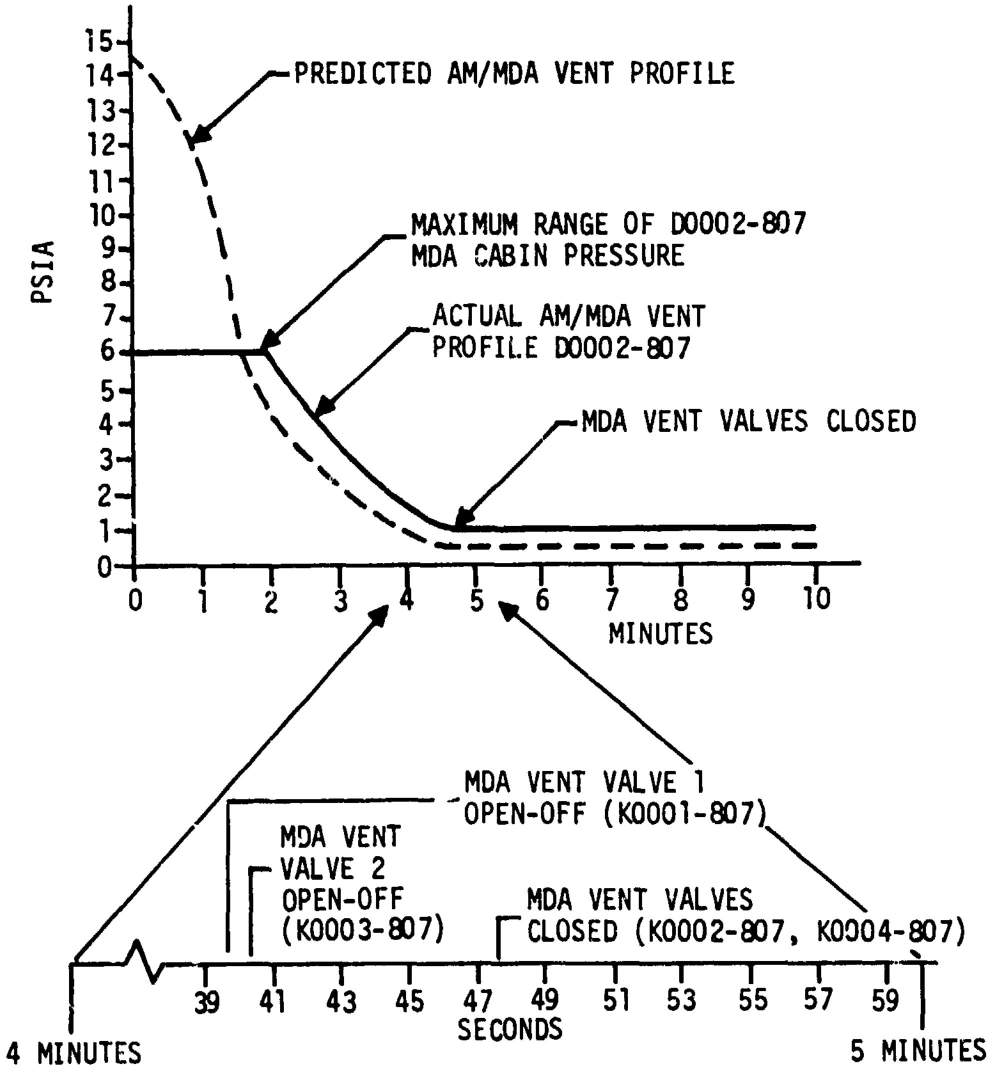


FIGURE 2.8-44 MDA VENT VALVE OPERATION

2.9 INSTRUMENTATION SYSTEM

The initial Saturn Workshop Instrumentation System utilized Gemini Program hardware. It was expanded to its present form during the change from the wet to dry workshop concept. This expansion resulted in equipment modifications, additional hardware, relocation of components and accommodation of MDA, OWS, and selected ATM measurements by this system. Subsequent design changes during the program only added selected sensors. The final system consisted of:

- Sensors/Signal Conditioners
- Regulated Power Converters
- PCM Multiplexers/Programmer/Interface Box
- Tape Recorder/Reproducers

This equipment was used to sense, condition, multiplex and encode vehicle systems, experiment and biomedical data for downlink to the Spaceflight Tracking and Data Network (STDN). Telemetry data was backed up by selected crew displays and by PCM hardline capability for prelaunch checkout. Real time data was supplemented from on-board recordings played back for downlink in delayed time. A total of 1076 telemetry channels, 566 in the AM, 10 in the ATM, 416 in the OWS, and 84 in the MDA, were monitored by this system. Figure 2.9-1 depicts the system in block diagram form.

2.9.1 Design Requirements

The prime requirements of the Instrumentation System were to acquire, multiplex and encode data from the AM, OWS, and MDA and to provide the data as follows:

- Via telemetry for real time coverage.
- Via tape recordings for continuous coverage.
- Via panel displays for crewmen.
- Via hardline for prelaunch operation.

Some of these requirements were established during the course of the design program; all were implemented prior to test and delivery of the AM. Detailed requirements, specific to equipment design and performance are defined where applicable in the system description section. No equipment requirements imposed upon the equipment vendors are discussed except where necessary to describe the equipment or its function. Specific requirements governing system design, operation, use, and documentation included the following:

- Make maximum use of existing flight qualified hardware.
- Make maximum use of common equipment between vehicles.

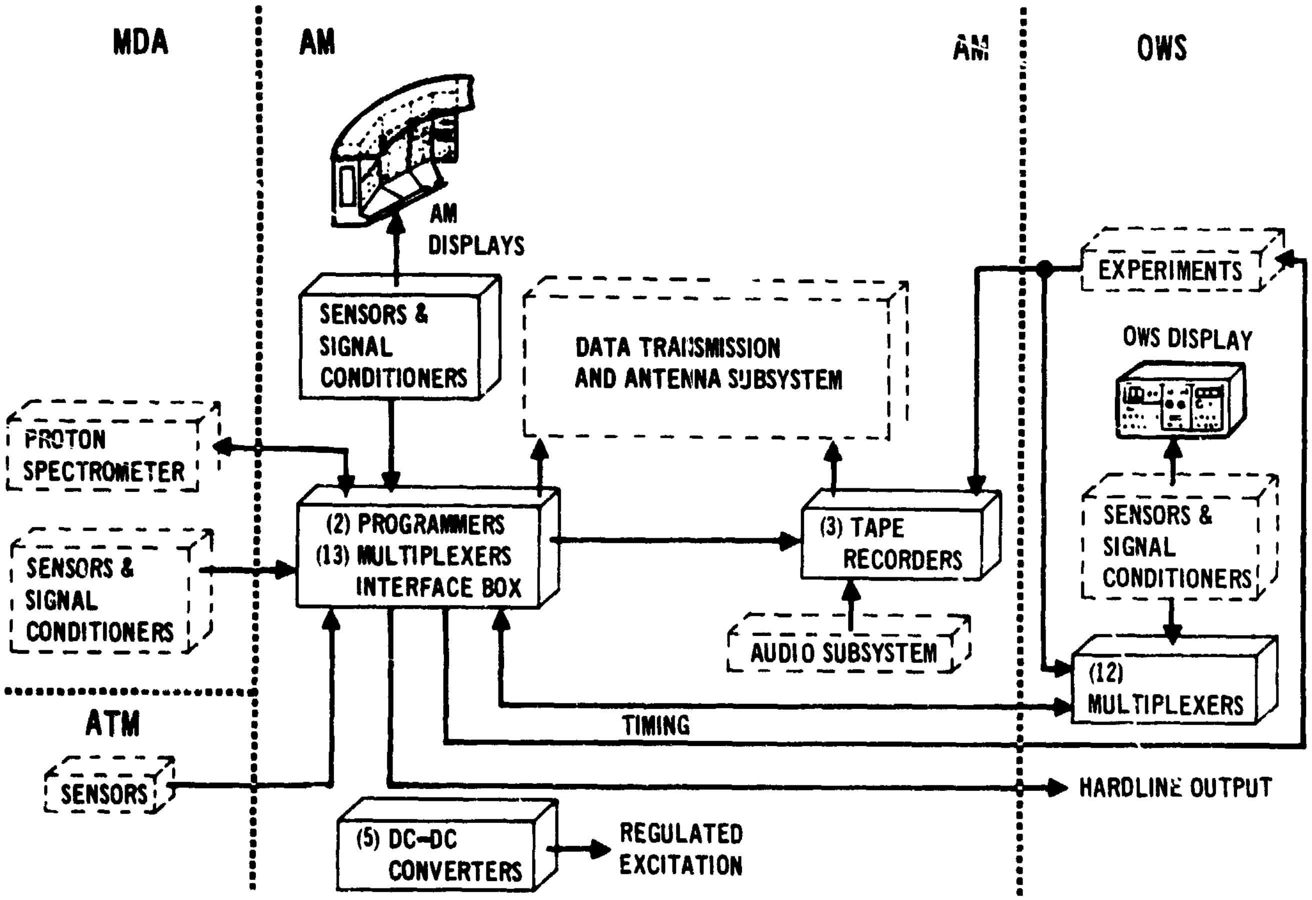


FIGURE 2.9-1 SATURN WORKSHOP INSTRUMENTATION SYSTEM

- Provide for in-flight replacement capability of selected hardware.
- Provide redundancy to meet mission requirements except signal conditioners, transducers, and multiplexers.
- Design system for compatibility with the STDN.
- Provide timing to OWS experiments from PCM Interface Box.
- Provide scientific experiments support.
- Monitor all parameters during manned missions and selected measurements during storage.
- Provide isolated outputs on transducers which supply signals to more than one system.
- Use Pulse Code Modulation telemetry.
- Provide ground control over equipment selection and functions with crew control backup.
- Provide crew control over experiment and voice recording.
- Provide ground control over data downlink.
- Provide timing to EREP.
- Design system to a goal that equipment be neither source of nor susceptible to EMI.

2.9.2 System Description

The Instrumentation System was assembled by utilizing existing Gemini Program designs where applicable and/or by modifying these and other designs to accommodate AM requirements. New designs were used only where available hardware did not satisfy needs.

The initial system consisted of 238 channels of PCM telemetry with single tape recorder capability. Program evolution and mission redefinitions resulted in a series of studies to determine the best methods to accommodate the data from other Skylab vehicles; downlink and redundancy alternatives were also considered. It was concluded that an expansion of the PCM Multiplexer/Encoder equipment in the AM downlinked via VHF transmitters would be the most efficient method to satisfy the new requirements. An interface box was added to the PCM equipment which allowed use of an increased quantity of low sample rate channels via added multiplexers. A total of 37 multiplexers could be accommodated. The interface box also provided for three additional separate portions of the real time data output to be available for recording; these allowed excess housekeeping and experiment data to be available via delayed time. Two additional tape recorder/reproducers and an additional DC-DC converter required for their excitation were added. Redundancy for selected equipment was provided. The evolution of the vehicle system design dictated some new sensors, range changes on existing sensors, an increase in the nominal sensor types and some additional signal conditioning be provided to satisfy the added functional monitoring requirements. This baseline system provided an increase in telemetry channel capacity to 428 channels with 342 used; multiplexers were located in the AM and OWS.

During the change from a wet to dry workshop concept, the major alteration to this baseline was an increase in the quantity of multiplexers. The measurement capability in the AM was now 629 channels; 535 channels were allocated.

Subsequent changes to the Instrumentation System included reallocation of multiplexers among the Skylab modules to optimize mission data acquisition and operations. Nominal measurement changes resulting from vehicle and experiment system evolution were also experienced during this program period. The final flight system provided 1297 telemetry channels of which 1076 were used; remote multiplexers were only located in the AM and OWS. Data signals from the MDA and selected measurements from other modules were wired across the appropriate vehicle interface and accommodated by the multiplexing and encoding hardware in the AM.

System control was primarily ground command with crew backup. The Airlock Module Instrumentation System provided a portion of either sensing, multiplexing and encoding, or recording functions for the following total parameters from the AM, MDA, OWS, and ATM:

363 Temperatures
106 Pressures
15 Flows
536 Events
284 Voltages/Currents
117 Miscellaneous

The end result was a system with maximum data monitoring flexibility which still maintained efficient operational ease for crew members as well as ground controllers. The subsequent paragraphs describe the individual components contained in this system.

2.9.2.1 Sensors and Signal Conditioners

The devices used to provide life support, physical environment and systems housekeeping data in the AM are described below. The temperature, pressure and CO_2 partial pressure sensors were basic Gemini Program designs. New AM designs included the gas flowmeter, rapid pressure loss and fire detectors. The remaining units, acoustic noise, dew point temperature, O_2 partial pressure and quartz crystal microbalance contamination sensors were essentially existing designs modified for AM needs. The signal conditioners fit into all three categories, the new designs being mainly those used in the Caution and Warning System. The description of the rapid pressure loss detector and ultraviolet fire detector are presented in Section 2.11. These devices were primarily used to supply emergency inputs to the Caution and Warning System. Similar data from these was also monitored via the telemetry system.

A. Temperature Sensors - Twenty-two different configuration resistive-element temperature sensors were provided in the AM to sense various temperatures, and to convert these temperatures into proportional electrical outputs for telemetry and for crew displays. Sensor outputs to the C&W System were signal conditioned prior to use. The sensors consisted of surface-mounted and probe sensing elements with integral bridges. The sensing elements were made from fully annealed pure platinum wire encased in ceramic insulation in a strain-free manner to provide maximum stability.

   Surface-mounted sensors were used to measure the skin temperatures of components and spacecraft structures. Air probe temperature sensors were positioned upstream and downstream of the mole sieve heat exchanger to provide cabin temperature data and for evaluation of the heat exchanger performance. An additional air probe sensor was located in the aft compartment of the AM. This sensor indication was a function of the OWS return air temperature and the OWS cooling module performance. Sensor outputs to telemetry ranged from 0 to 20 millivolts DC while 0 to 0.4 volts DC were provided to panel displays.

B. Dew Point Temperature - The dew point sensor utilized the cold mirror technique in which a reflective surface was cooled to the temperature at which condensation began to form on the mirror surface. This temperature was the dew point. The presence of moisture on the mirror surface was detected by the change in intensity of a light beam reflected off the mirror onto a photocell. The dew point sensor provided a 0 to 5 VDC output for telemetry corresponding to a dew point temperature of 20 to 80°F and a 0 to 0.4 VDC output for a crew display. Two dew point sensors were used in AM, one at the inlet of each mole sieve.

C. Pressure Transducers/Switches - Ten different absolute and differential pressure transducers were provided to sense O_2, N_2, water and coolant pressures and to convert these pressures into proportional electrical outputs for telemetry and crew displays. Six different pressure switches were used for telemetry, display, control and C&W. The mechanical portion of eight of the transducers was a bellows or a capsule which varied the wiper position of a potentiometer proportionally with input pressure variation. Two potentiometers were used in the dual output units to insure that the indicator circuit did not create a loading error on the PCM output. The pressure switches used a mechanical switch in lieu of the potentiometer as the output. Redundant units were provided for critical functions. The transducers converted the pressure into proportional electrical outputs ranging from 0 to 5 VDC for telemetry and 0 to 0.4 VDC for crew displays.

Low pressures were sensed by the other two transducer types by a unit consisting of a diaphragm mounted to a slug in a transformer. Small deflections of a diaphragm changed the reluctance in the transformer coils, which were connected as legs of an AC Bridge. The output from this bridge was suitably conditioned to provide a 0 to 5 VDC output for telemetry and a 0 to 0.4 VDC signal for crew displays.

D. Carbon Dioxide Partial Pressure Detector - The $PPCO_2$ transducer ionized filtered gas to obtain an output signal which was proportional to the partial pressure of CO_2 present at the point of measurement. The transducer consisted of two in-flight replaceable filters, two ion chambers and a bridge circuit. An inlet gas stream was divided into two substreams. One substream was filtered for CO_2 and H_2O removal; the other was filtered for H_2O removal only. The output gas substreams were ionized by the chambers containing approximately 400 microcuries of Americium 241 each. The resulting ion currents from each substream were compared in a bridge circuit to obtain the measurement.

Six $PPCO_2$ detectors were used; one at the inlet of each mole sieve and two at the output of each mole sieve. The inlet detectors provided the cabin level indication and the outlet detectors monitored the mole sieve performance and provided a caution and warning signal if the mole sieve performance became marginal. Each unit provided two 0.2 to 5.2 VDC outputs for telemetry and C&W and a 16 to 416 mV DC output for crew display; both outputs were proportional to 0 to 20 mmHg CO_2 partial pressure input. Provisions were made for thirty-two filter changes during the three Skylab missions.

E. Oxygen Partial Pressure Sensing System - The PPO_2 transducer was composed of two subassemblies, an in-flight replaceable life limited O_2 sensor and an amplifier. The sensor consisted of a diffusion barrier, gold-plated stainless steel catalytic electrode, potassium hydroxide electrolyte and a metal counterelectrode made from copper. These were physically joined together in a housing and electrically connected externally through a load resistor. Selection of the diffusion barrier provided an oxygen flow which was directly proportional to the oxygen partial pressure.

The output was a current flow through the external load resistor. The voltage drop across the load resistor was amplified and conditioned to supply 0 to 5 VDC outputs for C&W, telemetry, and O_2/N_2 control and a 0 to 400 mV DC for crew display. Both outputs were proportional to 0 to 6 psi oxygen partial pressure. Each output was isolated from the other so that mutual interference would not occur. Three transducers were provided in the AM; one was used for monitoring, the second provided the control signals for the O_2/N_2 control system and the third was an installed spare selectable by the crew for either of the other two. Six on-board spare sensors were provided for crew replacement during the three Skylab missions.

F. Flowmeters - Two turbine type flowmeters, of different ranges, and three "time of flight" flowmeters, of different ranges, were provided in the AM. The first type used a turbine sensor to transform the flowrate of coolant into a pulse stream whose pulse rate was proportional to the flowrate. The pulse stream was converted into a 0 to 5 VDC signal for telemetry. The second type of flowmeter was used to measure gas flow in the AM circulation and atmosphere revitalization system. The gas flowmeter was composed of two subassemblies, the flow sensor and the flowmeter converter. The sensor was a hollow tube in which a heating element was located near the input end and a thermal pulse sensor was located near the outlet end. The flowmeter converter was composed of a thermal pulse generator, thermal pulse preamplifier, correlator/delay detector and an isolated output amplifier/bilevel detector. The output amplifier/bilevel detector provided a 0 to 5 VDC output for TM and a switch closure to the C&W system when the flowrate decreased below the minimum allowable.

The system operated by measuring the time delay between the creation of a thermal pulse at the heater, and its detection downstream at the sensor. By using correlation, the system was independent of random fluctuations in temperature. The accuracy of the unit depended only upon the precision of the heater-sensor spacing and the measurement of the time of flight of the thermal pulse. A 0 to 5 VDC output was provided for telemetry, and a switch closure was provided to the C&W system if the flowrate fell out of tolerance.

G. Quartz Crystal Microbalance Contamination Monitor (QCM/CM) - Four QCM/CM's were mounted on the ATM Deployment Assembly to measure contamination in the area of the EREP. One QCM pointed towards the CSM (+X), one pointed away from the CSM (-X) and two away from the ATM (+Z). One of the +Z units was passively temperature controlled to approximately 50°F, the remaining units assumed the ambient temperature. Each QCM/CM used two quartz crystals, one shielded and the other exposed to the environment.

Each crystal oscillated at about 10 MHz. As contamination was deposited on the exposed crystal, its mass increased and its resonant frequency decreased in proportion to the mass of the contamination. The frequency of the shielded and exposed crystals were compared, this difference (beat frequency) being proportional to the deposited mass. The beat frequency was converted to a 0 to 5 VDC signal for telemetry. A range-expanding signal conditioner in the signal conditioner packages provided eight expanded 0 to 5 VDC segments over the full-scale range of the instrument for increased resolution of reading. Four conditioners were supplied, one for each QCM/CM. A 0 to 5 VDC signal which was representative of the crystal temperature was also telemetered. The full-scale range of the QCM/CM was approximately 1.2 x 10^{-4} grams of deposited material.

A requirement unique to the QCM/CM was that MDAC-E had to develop a means to calibrate the device. The calibration was accomplished by depositing a known mass of material on the QCM/CM and measuring its response in terms of the relative change in both the Mass Deposition Output (MDO) and the Beat Frequency Output (BFO). The validity of the calibration was dependent upon being able to:

- Deposit material in the form of thin films whose deposition rate and uniformity were known and reproducible.
- Accurately measure the film thickness.
- Select a deposition material with properties similar to those of the outgassing products expected from Skylab.

A technique for calibrating the QCM was developed in the MDC Applied Optics Lab that satisfied all of the above conditions. The deposits were obtained from a device identified as a Vapor Effusion Source (VES) which consisted of a heated copper cavity equipped with a .05 cm diameter effusion nozzle and used DC 704 diffusion pump oil for the deposition material. DC 704 was selected because in addition to satisfying the third condition, it was chemically stable at the temperatures and pressures of interest. The measurement of the film thickness was accomplished by the adaptation of an optical technique referred to as ellipsometry. The mass sensitivity of a QCM was determined as follows:

> A film of the contaminant was deposited on the surface of a gold mirror by using the VES. The thickness of the resulting film was measured with the ellipsometer to determine the deposition rate as a function of source temperature and deposition time to produce a calibration of the source. The source was then used to contaminate the receiver of the QCM. By varying the exposure time, it was possible to deposit a range of known masses on the receiver, thereby providing the data that was needed to compute the mass sensitivity of the QCM/CM.

H. Acoustic Noise Measuring System - An Acoustic Noise Measuring System was installed in the Airlock to provide sound pressure level data during the launch phase of the mission. The system consisted of a high intensity charge microphone mounted in the payload shroud area and connected to a remote electronics package which filtered and amplified the microphone signal. The system was capable of sensing sound pressure levels of 126 to 146 decibels over a frequency of 20 to 1650 hertz within 3 decibels. The corresponding output was 5.0 volts peak-to-peak with 2.5 $\pm$.05 VDC representing no input signal. This measurement was downlinked via the FM/FM telemetry system in the I.U.

I. Vibration Measuring System - Two vibration measuring systems were installed in the Airlock to provide vibration data during the launch phase of the mission. Each system consisted of a piezoelectric accelerometer connected to a remote electronics package which filtered and amplified the accelerometer signal. One accelerometer sensed X-axis vibration at one ATM attach point in the Payload Shroud; the other accelerometer was used to sense X-axis vibration on the structural transition section. Each system was capable of sensing $\pm$5g levels over a frequency range of 3 to 80 hertz. The corresponding output was a 0 to 5V peak-to-peak signal with 2.5 $\pm$.05 VDC corresponding to a zero input. Both measurements were downlinked via the FM/FM telemetry system in the I.U. This device was also used in the OWS to provide launch vibration data.

J. Signal Conditioning System - Two packages containing individual signal conditioner plug-in modules were installed on coldplates, on electronics module #3, external to the AM. These conditioners interfaced between sensor outputs or existing vehicle system electrical signals and the PCM multiplexer/encoder hardware to provide signal compatibility. They were used for telemetry, crew displays and experiments. The packages provided 51 channels each; a total of 83 slots were used in both packages. Two separate similar packages were provided for C&W; they were installed on electronics module #5. There were 26 active channels per package for complete redundancy. The individual signal conditioner modules, used in all packages, were constructed on circuit boards using printed wiring techniques. This permitted component replacement on the individual modules. The circuit boards were mounted to a mother board via connectors which provided ease of replacement and considerable system flexibility. Filtering was done on individual modules as required. There were 28 different type signal conditioner modules utilized in the AM. Each was custom designed for a specific purpose.

2.9.2.2 Regulated Power Subsystem

Five DC-DC converters were provided in the AM portion of the SWS Instrumentation System; there were two different types. The three used for telemetry requirements were of Gemini Program design, modified for increased power output; the two used for display functions were an existing design adapted for AM use. These units converted the AM bus power of 18 to 32 VDC into regulated voltages of $\pm$24 VDC and +5 VDC. This power was divided into three buses A, B and display, each containing the $\pm$24 VDC and +5 VDC. The bus A was nominally supplied by telemetry converter 1 and was used for nonexperiment system operations. Bus B was supplied by telemetry converter 2 and was active when tape recording of experiment data was required. A third telemetry converter was supplied as a wired spare. Converter/bus selection was via DCS command with complementary crew controls. The display functions were powered from bus A except in the telemetry "off" condition when the display converter automatically supplied the required excitation. A redundant display converter was provided; this selection was performed by crew controls. Figure 2.9-2 illustrates the components and loads which formed this subsystem.

A. DC-DC Converter - The three telemetry converters were installed on electronics module number 4; active coldplates were utilized. Each converter provided 40 watts of +24 VDC, 30 watts of -24 VDC and 1.5 watts of +5 VDC. To accomplish this, the converter utilized the unregulated bus voltage to drive an inverter. The inverter output was then transformer coupled and rectified. This voltage was filtered and regulated by a pulse width regulator. A switching regulator was used to improve efficiency. To achieve increased stability in the +5 VDC output, the $\pm$24 VDC regulated outputs were utilized along with a chopper stabilized regulator.

B. Panel Meter DC-DC Converter - The two panel meter converters were installed on electronics module number 5. Each was capable of supplying 8 watts of +24 VDC, 1 watt of -24 VDC and 1 watt of +5 VDC. The unregulated bus voltage supplied to this converter was first filtered and then applied to a preregulator. The output of this regulator was supplied

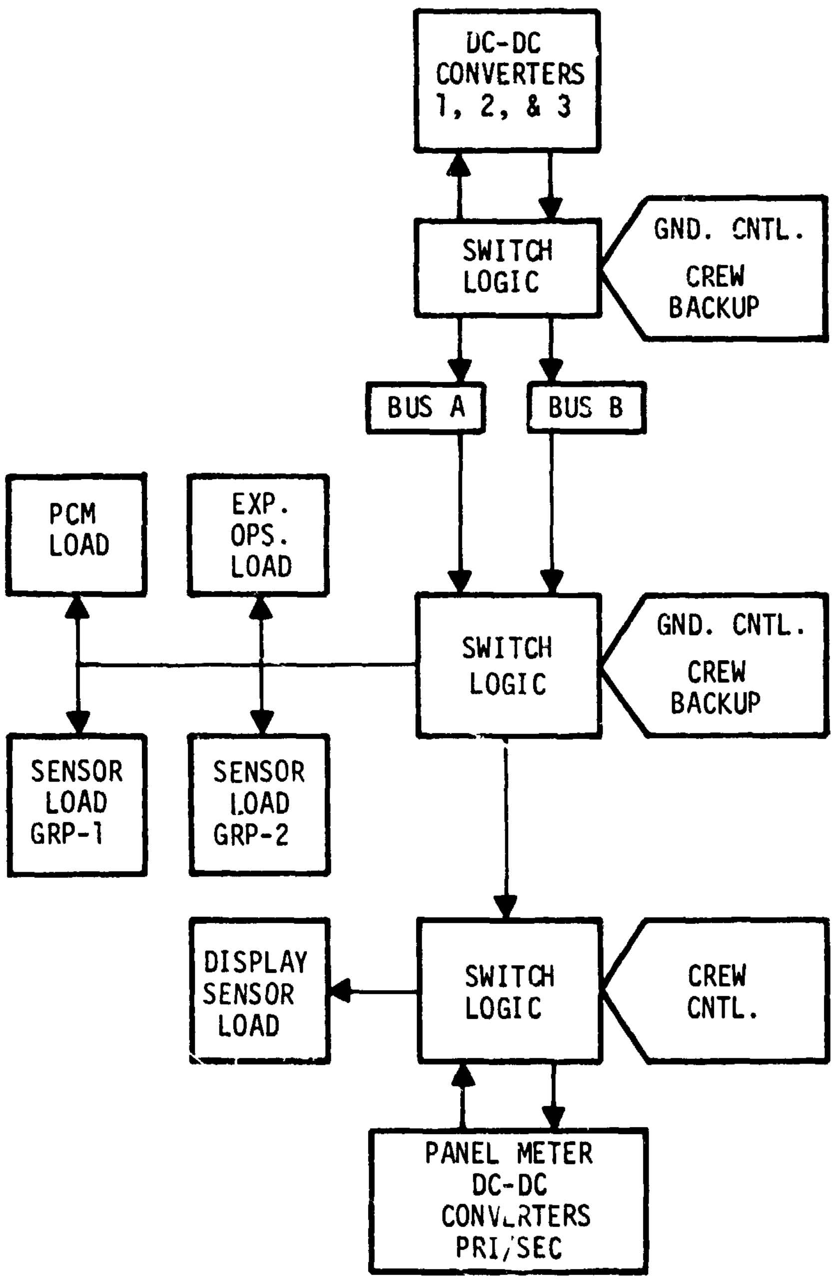


FIGURE 2.9-2 INSTRUMENTATION REGULATED POWER SUBSYSTEM

to a DC to AC inverter whose output was tied to three AC to DC regulated converters which developed the ±24 VDC and +5 VDC outputs.

2.9.2.3 PCM Multiplexer/Encoder System

The PCM System design was constrained by the requirement to utilize existing Gemini hardware designs to the greatest extent possible. The programmers and multiplexers were used with only minor modifications; the interface box (IB) was a new design using the same construction techniques as the programmer and multiplexers. Due to the large number of input channels and the greatly increased complexity, a new design test set was required to facilitate design and acceptance testing. This test set provided more accurate and faster testing. The environmental design requirements were essentially the same as were required on the Gemini Program with the exception of the vibration requirement for multiplexers which were to be located in the OWS. A special test was performed which subjected one low level and one high level multiplexer to a random vibration of 25.1 g's rms in the most critical axis for 12 minutes.

The SWS PCM System consisted of the following major components:

- 2 redundant and switchable programmers
- 1 interface box (redundant electronics)
- 11 high level multiplexers
- 14 low level multiplexers

Design of the PCM allowed interfacing with a maximum of 18 high level multiplexers and 19 low level multiplexers.

The Airlock complement, located on coldplates on electronics module number 3, external to the pressurized area of the Airlock, (shown in Figure 2.9-3) provided an input capability for 1,297 data channels. A summary table of maximum system capability is listed in Figure 2.9-4

The PCM programmer provided a 51.2 Kilo bits per second (KBPS) nonreturn-to-zero (NRZ) real time output for transmission to the STDN, a 51.2 KBPS hardline output for use during prelaunch checkout, and a 5.12 KBPS return-to-zero (RZ) output, identified as Subframe 1, for recording on the tape recorder/reproducer

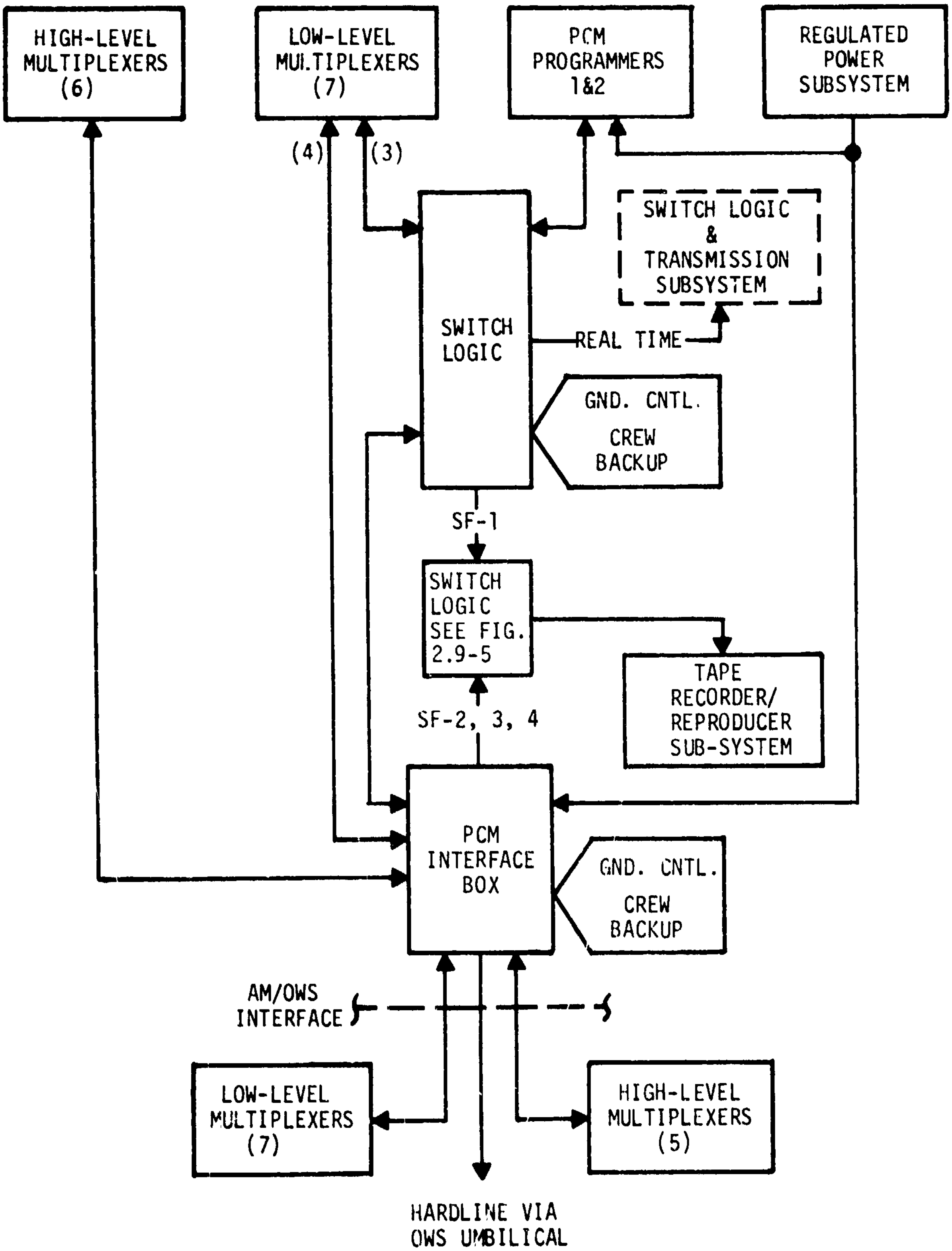


FIGURE 2.9-3 PCM MULTIPLEXER/ENCODER

SAMPLE RATE (SPS)	TYPE SIGNAL	QTY	UNIT
320	HL	5	IB
160	LL	6	PROG
80	HL	8	IB
80	LL	9	PROG
40	HL	1	IB
20	HL	5	IB
10	HL	6	PROG
10	HL	18	IB
10	BL	40	PROG
10	BL	256	HLM
10	BLP	192	HLM
1.25	HL	551	HLM
1.25	HL	32	PROG
1.25	LL	152	LLM
0.416	LL	455	LLM
0.416	SERIAL DIG	24	PROG
TOTAL		1,760	

HL MULTI	USE	QUANTITY HL	BL	BLP
B	AM	32	24	16
C	AM	32	0	16
D	OWS	32	0	16
E	OWS	31	0	16
F	AM	32	24	16
G	SPARE	32	24	8
H	SPARE	32	24	16
J	OWS	32	24	8
K	OWS	24	24	8
L	SPARE	32	0	0
M	SPARE	32	0	0
N	SPARE	32	0	0
P	SPARE	24	8	8
Q	SPARE	24	8	0
R	AM	32	24	16
S	AM	32	24	16
T	OWS	32	24	16
U	AM	32	24	16
TOTALS		551	256	192

LL MULTI	USE	QUANTITY 1.25 SPS	.416 SPS
B	OWS	8	24
C	AM	8	24
D	OWS	8	24
E	AM	8	24
F	AM	8	24
G	AM	8	24
H	OWS	8	24
J	OWS	8	24
K	SPARE	8	24
L	OWS	8	24
M	OWS	8	24
N	SPARE	8	23
P	AM	8	24
Q	OWS	8	24
R	SPARE	8	24
S	AM	8	24
T	AM	8	24
U	SPARE	8	24
V	SPARE	8	24
TOTALS		152	455

TOTAL HL ANALOG (0-5 VDC)	626
TOTAL LL ANALOG (0-20 MVDC)	622
TOTAL BILEVEL (ON-OFF)	296
TOTAL BILEVEL PULSE	192
TOTAL SERIAL DIGITAL	24
TOTAL	1,760

FIGURE 2.9-4 PCM MULTIPLEXER/ENCODER CHANNEL CAPABILITY

system. The Interface Box (IB) provided three additional 5.12 KBPS RZ signals to the tape recorder/reproducer system which were identified as Subframes (SF's) 2 through 4.

A. Programmer - The programmer provided the functions of data multiplexing, analog-to-digital (A-D) conversion, digital-data multiplexing, and the required timing functions for the IB. The programmer contained some input gates but primarily consisted of the circuitry necessary to provide 51.2 KBPS nonreturn-to-zero (NRZ) change PCM pulse trains to the transmitter and to provide 5.12 KBPS return-to-zero (RZ) pulse train signal and clock pulses for Subframe 1 to the tape recorder/reproducer subsystems.

B. Interface Box - The interface box (IB) accepted timing signals from the programmer and provided signals to the remotely located multiplexers. It also provided timing signals necessary for the generation and multiplexing of the data in Subframes 2, 3 and 4. The programmer provided the 51.2 KBPS signal to the interface box where Subframes 2, 3 and 4 were separated and prepared for transfer to the tape recorder/reproducer subsystem. Three internal power supplies were located in the interface box. One was used by the internal circuitry in the IB and the other two provided power to the multiplexers. The interface box was composed of a redundant set of electronics each capable of full systems operation independent of the other.

C. High Level Multiplexer - The high level multiplexer functioned as a high level analog commutator and an ON-OFF digital data multiplexer. The purpose of this unit was to sample 32 high level data channels (0 to 5 VDC), 24 bilevel signals (0 or 28 VDC), and 16 bilevel pulse signals (0 to 28 VDC).

   All high level multiplexer analog data outputs were switched through the interface box to the programmer. Each multiplexer was individually wired to the IB where third tier switching was performed before the data was sent to the programmers. Individual third tier switching was used to prevent a short in one multiplexer line from shorting all other multiplexers and to keep line capacitance at a minimum.

The high level multiplexer used a slaved timing chain driven by signals from the IB to support the required sampling functions.

D. Low Level Multiplexer - The low level multiplexer was a differential-input analog commutator whose purpose was to sequentially sample 32 low level (0 to 20 MVDC) signals. The multiplexer contained a slaved timing chain and digital logic to support the required sampling functions. Operating power and timing slave signals were received from the IB.

All low level multiplexers, except E, F and G, were individually switched through third tier switches located in the IB. Multiplexers E, F and G were gated by, and switched through, the selected programmer.

2.9.2.4 Tape Recorder/Reproducer Subsystem

Three tape recorders capable of simultaneous operation were employed to provide continuous data coverage during periods when the Skylab vehicle was out of STDN contact. This recorder was a Gemini Program design modified for voice record and multiple playback capability. They were installed on coldplates in the forward compartment of the AM. Each recorder received from the PCM programmer or interface box, a 5.12 KBPS RZ data stream comprising one of the four recordable PCM subframes. This data was recorded on track A, while crew voice was recorded on track B; record speed was 1 7/8 IPS. Maximum record time was 3 hours per recorder. In addition to the subframe data, the recorders could also accommodate experiment M509 or T013 data at a bit rate of 5.76 KBPS. The recorder played back the PCM data in a NRZ - space format into one transmitter; the voice was played back simultaneously into another transmitter. The playback occurred at a speed of 22 times the record speed; data and voice was played back in an order reverse to what they were recorded. Playback of 3 hours of data was accomplished in 8 minutes, 24 seconds. Upon removal of the playback command, the recorder switched from the playback mode to the record mode. In the event of faulty data reception, the tape recorders could be rewound at the playback speed for another dump by application of a fast-forward nonrecord command. During this rewind no modulation was present at the transmitter. Figure 2.9-5 provides a flow diagram of the recording process.

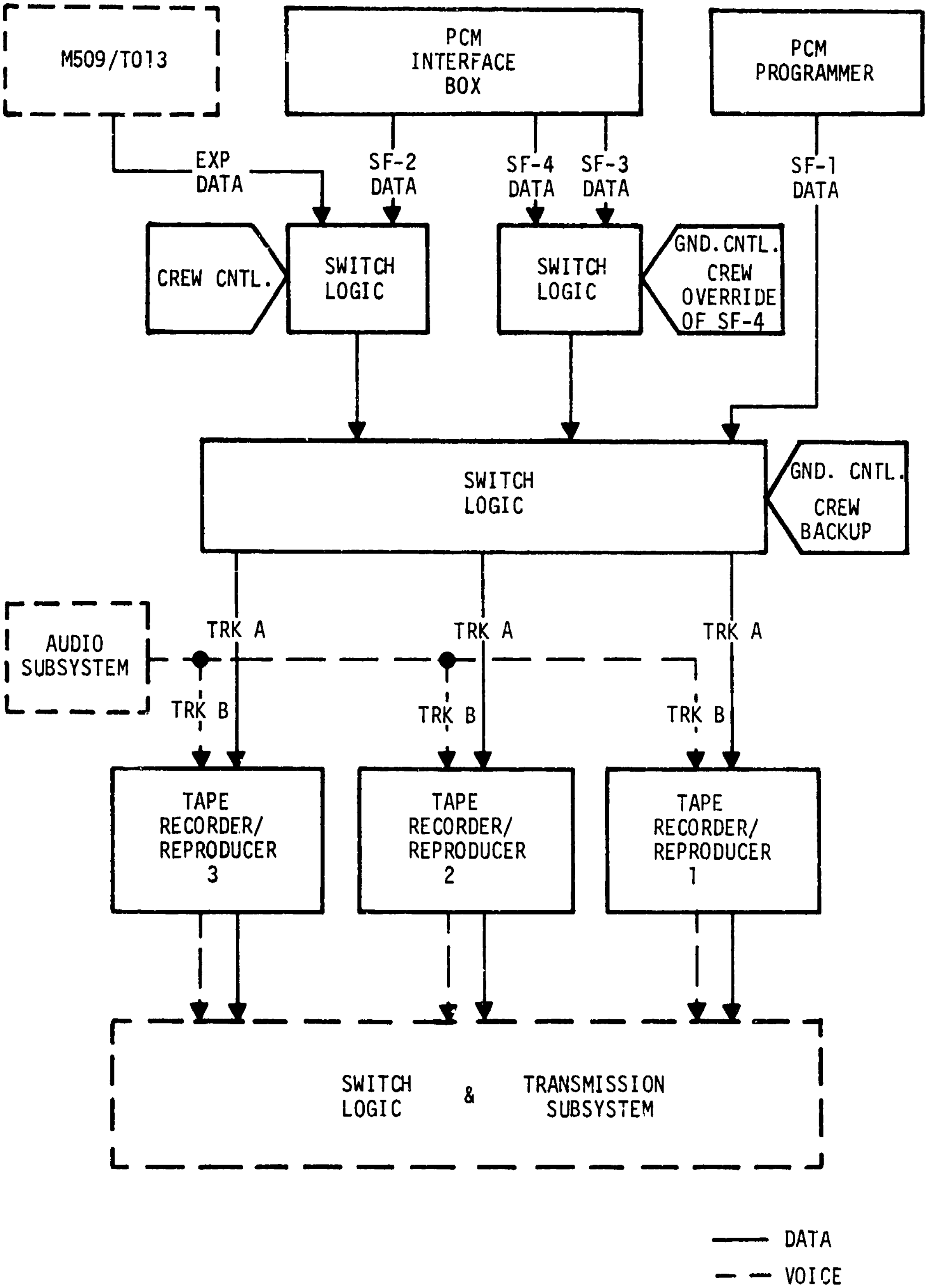


FIGURE 2.9-5 RECORDED DATA SIGNAL FLOW

Recorder management was primarily a ground control function; the crew however, were provided with duplicate controls. The crew exercised control over voice and experiment record functions. Telemetered recorder functions included tape motion and playback mode detect. The crew was supplied with tape motion and tape stopped lights at all recorder control stations.

Four recorders were launched on SL-1 as in-flight replacements. These units were installed in the AM lock compartment for launch and were transferred to the OWS for stowage after activation. Two additional recorders were resupplied during the second manned mission.

2.9.2.5 Latch Relay Monitor

The latch relay monitor circuit identified a particular position of the AM latch relays via a telemetry parameter. This circuit was vital to the mission since SL-1 was launched unmanned and these circuits were accessable only through DCS during countdown.

During testing at MDAC-E and at KSC this latch relay monitor circuit served a dual purpose: to gain confidence in test procedures and to aid in locating an improperly positioned latch relay. Over 500 latch relays were positioned in the ECS, EPS, Lighting, Sequential, Instrumentation, Communication, and C&W Systems so that the latch monitor circuit could be used to establish the integrity of all the latch relays. Once this was established those few systems which contained relays requiring a different lift-off position were cycled to the proper position and, by other telemetry indications, these circuits were established as operational with the associated latch relays being properly configured.

2.9.2.6 Configuration Documentation

The Instrumentation hardware, measurement characteristics and calibration data was documented primarily by three reports:

MDAC Report F639 - Instrumentation System Description

AM IP&CL - Instrumentation Program & Components List

MDAC Report E0502 - Telemetry and Recording Technical Manual

Report F639 consisted of five separate volumes and was mainly used for in-house design, test and mission support activities. Report E0502 provided PCM Multiplexer/Encoder and tape recorder details. Both of these reports were submitted as information items to MSFC. The IP&CL measurements list and a set of computer punched cards containing calibration data were submitted to MSFC for approval. These documents were kept current via periodic updates during the design and test phase of the program. Measurement and calibration changes were handled on an individual basis and included in the appropriate ECP or CCP that defined the change; these were supplied to MSFC. The contents of each report was as follows:

A. F639 Volume 1 - This report presented descriptions of each component of the Instrumentation System. Design information, theory of operation and physical characteristics were documented. The controls and displays were also discussed. In addition, a word description of each telemetry and display measurement defining the function monitored was given.

B. F639 Volume II - The second volume of this report was measurement oriented It consisted of a series of tabulations geared to specific user requirements; these lists were generated by a computer/magnetic tape file, storage and retrieval system. The different lists and their contents were as follows:

   (1) Summary Table - This was a list by parameter sequence number presenting the basic measurement characteristics.

   (2) Setup Table - This tabulation by parameter sequence number listed characteristics as well as sensor and signal conditioner information on each measurement.

   (3) Equipment List - This listed Instrumentation System hardware by part number, parameter sequence number, serial number and location of the part in the vehicle.

   (4) Format Assignment and Decommutation Setup - This was a tabulation by parameter sequence number and provided multiplexer/encoder channel identification correlated to equipment connector pin assignments associated with that channel. It also contained the decommutation code used to retrieve the measurement from the data bit stream utilizing the St. Louis PCM ground station.

   (5) Signal Conditioner Assignments - A tabulation by signal conditioner channel of part used with associated serial number and parameter sequence number. Location within the four signal conditioner packages were also provided.

(6) Parts List - This was composed of two different tabulations. One listed flight hardware by part number with numerical quantity of each kind used on the vehicle. The second list provided similar information for spare and test hardware.

(7) Parameter Type Summary - This was a one-page tabulation providing quantity of parameters by sample rate and type. It included a list of spare multiplexer/encoder channels available.

(8) Telemetry Format Logic Diagram - This provided a definition of the multiplexer/encoder channel sampling sequence in terms of a matrix which showed channel number versus parameter sequence number.

(9) Format Allocation - This list provided both the telemetry equipment vendor channel code and the IP&CL channel code versus the IP&CL parameter identification number.

C. F639 Volume III - An end-to-end schematic of each telemetry and display parameter was presented in this report. Electrical schematics of each signal conditioner, regulated power distribution, controls and other circuitry associated with these parameters were also provided.

D. F639 Volume IV - This volume presented calibration data for each telemetry and display measurement originating in the AM, MDA and PS. Extensive utilization of computer technology was employed in collecting, controlling and processing this data. The calibration information for each sensor and/or signal conditioner was processed through a least squares curve fitting routine to establish the best first, second or third order curve. The data was presented in combinations of the following four basic formats:

(1) Calibration Plot - The input/output data points were presented in an X-Y plot with a line representing the best first, second or third order curve fit meeting the specified accuracy requirements.

(2) PCM Counts Tabulation - Calculated engineering unit values were tabulated versus each PCM count. Zero counts represented an underscale, one count was zero, 254 counts represented full-scale and 255 counts indicated an overscale.

(3) Percent Tabulation - This was a tabulation in 5 percent increments from 0 to 100 percent versus the equivalent in engineering units.

(4) Real Data Tabulation - This contained the raw data taken during the calibration test.

Discrete event calibration data was presented in tabular form only. This list contained the identification of the binary ones and zeros of each bit of the bilevel channels. Voltage levels for step functions on analog channels and the meaning of on/off indications of crew display lights were also identified. The values presented were actual calibrated trigger points when level sensing devices were used in the circuit. A complete calibration data package for the AM, MDA and PS was also supplied in a different format to MSFC. This was derived from the same calibration file as the F639 Volume IV data. The format used was a key punched computer card utilized by MSFC to generate a master calibration tape for all Skylab data users. Update cards were supplied with each change to insure the tape reflected the vehicle configuration.

E. MDC Report F639, Volume V containes the calibration data for the second (U-2) AM/MDA and PS.

F. IP&CL - This document was used to define and control telemetry and display parameters and the associated hardware. Additions and deletions required MSFC approval. All measurements originating in the AM were presented; the ATM measurements multiplexed and encoded by the SWS Instrumentation System were also included. This listing was derived from the same computer/magnetic tape system used to generate the F639 Volume II lists. The tabulations contained all flight and launch instrumentation measurement numbers, names, transmission mode, range, telemetry channel identification, accuracy, sample rate, transducer, signal conditioner and sensor locations where applicable. Measurements which required in-flight recording, on-board display and prelaunch display were identified. This document was updated as required and submitted to MSFC for approval prior to implementation.

G. E0502 - The Telemetering & Recording Technical Manual provided a discussion of the PCM multiplexer/encoder and tape recorder functions. Details were presented on the physical description of this hardware together with the building block modules utilized in their construction. Block diagrams were also provided, these were broken down on a functional basis consisting of 30 separate sections for the PCM hardware and 10 sections for the tape recorder/reproducer. The system timing and logic diagrams, building block module schematics, a PCM format explanation, and an engineering drawing list for this equipment was also included. Sensors, signal conditioners, DC-DC converters, downlink transmitters and telemetry parameters were not discussed in this report.

2.9.2.7 Results Documentation

The SWS Instrumentation System was utilized to provide operations and evaluation information during vehicle testing and from the actual flight. This data was processed and reduced into time history tabulations and plots. Strip charts and other data presentation formats were also used. This section presents a discussion of the treatment of test and mission data at MDAC-E in St. Louis.

A. Vendor Test Data - Component Qualification and acceptance test data was presented in vendor reports. The format included plots as well as tabular data. All qualification test results were reviewed and approved with NASA concurrence. Actual tests were witnessed by Government Inspection and in most cases by MDAC Quality Assurance. Acceptance test records were used as the input to the calibration data file described in the previous section or as baseline data to provide test criteria for subsequent testing at MDAC. This was the only processing of vendor test data performed at MDAC.

B. MDAC Test Data - Pre-Installation Test data was processed into calibration information for use in data reduction. All data from vehicle systems testing was recorded on magnetic tape; processing of this data into plots and tabulations was on a selected basis. During the various systems tests, pertinent data values were noted on data sheets contained in the test procedure. Real time strip charts and visual displays were also available at the test ground station during all testing. Processing of KSC systems test data was similar to that performed at MDAC.

C. Mission Data - There were two main sources of vehicle telemetry data during the Skylab missions. The Skylab Test Unit (STU/STDN) station provided processed data acquired during all St. Louis passes. MSFC supplied raw and processed data for all vehicle orbits.

(1) STU/STDN - This facility included tracking capability which provided a source of real time telemetry data; when data dumps were performed within range, the on-board recorded data was also obtained. All telemetry data that was received was recorded on magnetic tape.

This data was routed to a computer where the first sample of each AM measurement in the 2.4 second data frame was placed in disc storage. Following the pass, this stored data was available for conversion into engineering units and to be presented in tabular form on a system oriented basis. Processing was on a keep-up basis. There were 75 tabulations, each containing a maximum of nineteen measurements. Measurement number, name range, engineering units, GMT and the actual data points were provided on each page. A capability existed to flag data values which fell outside of predefined tolerances. The tolerance values were printed on the tabulations. A summary sheet listing all flags was provided for each pass. Special tabulations and printout of all available data points were provided upon request. Block diagrams of selected ECS/TCS subsystems were constructed for each pass. These presented single data values at the sensor locations on the block diagram; they were used as quick look system status information. The data processed by MSF was also used to construct system oriented graphs for the complete mission. One data value per day per measurement was plotted. These graphs were displayed in the Skylab Communications Center in St. Louis and updated on a daily basis with current data. Strip chart recordings of 32 analog and 100 discrete measurements were made on selected vehicle passes over St. Louis. The capability was available for real time monitoring of critical measurements. Additional strip charts could be generated post pass from the magnetic tape. MSF operational details are presented in Section 7 of this report.

(2) MSFC - Additional mission data was obtained via approved Data Request Forms (DRF). These sheets defined raw and processed data required by MDAC-E for mission support and evaluation. One hundred thirty-seven DRF's were prepared and submitted to MSFC for approval. Many different forms of data were requested. Compressed user tapes were provided by MSFC for all available mission time periods. These magnetic tapes were in data redundancy removed (DRR) form and contained all AM and selected MDA, OWS and ATM measurements. Each user tape contained four dictionary records followed by data records. The dictionary records contained tape identification, MSFC ID numbers,

corresponding measurement numbers and engineering unit definitions. The data records contained data source, GMT, and MSFC ID for each data value. A software program was written to reduce this data into time history plots and tabulations. Processing of the tapes was limited to problem time periods where detailed systems information was required. The processing software consisted of two programs written for the IBM 370-145 and the IBM 360-55 computer systems. The first program processed the tape and provided data tabulations and a magnetic tape. The second program converted the magnetic tape into data plots. The tabulation program operated in three phases: initialization, processing, and presentation. The initialization phase consisted of processing the control cards, base file and portions of the user tape dictionary records. The control cards contained the processing directions specifying the time intervals, measurement tabulations and program options to be exercised. The base file was a magnetic tape in Extended Binary Coded Decimal Interchange Code (EBCDIC) containing card images of information defining tabulation and plot formats. Tabulations were defined by tab set number, column number, measurement number, MSFC ID number, engineering units, band-edit value and measurement subframe. Fifty-two tab sets were defined for analog measurements, one for time and five for discrete events. Plot information was defined for each measurement on a tab set. Information filed was tab number, plot number, grid number, measurement number, name, range, symbol and right or left scale notation. Grids were defined by minimum, maximum and number of divisions. Changes to this base file were accomplished via new card image inputs. All processing was done in one pass through the user tape. Band editing of the data was performed on a selected basis. This feature compared the absolute difference of each data value and the previous value output with a band-edit delta read from the base file. A difference of less than or equal to the delta was not tabulated. A value greater than the delta was tabulated and used for comparison to succeeding data values until the delta was again exceeded. Time was incremented on the tabulations equal to the sample rate of the measurements being tabulated. In the case where the number of the measurement samples was greater than the tabulation rate, the last sample in the time increment was used. There were eleven columns of data values together with time and measurement identifiers. The first data line on each page presented the last printed value from the previous page. A notation was made of each change of real or delayed time and the end of each data segment was identified. The plotting program translated the data from the magnetic tape created during the tabulation program into appropriate language for the plotter. The data was point plotted with data gaps identified. There were four separate grids per page; the number of measurements per grid varied from one to four with a maximum of eleven measurements per page. All data on one page was limited to the measurements on one tab set. A maximum of four plots could be defined for one tab set. The X-axis of each grid was fixed at 24 divisions, time periods of 2, 4, 6, or 8 hours including start time could be used. Each Y-axis was defined by a minimum, maximum and number of divisions: 2, 3, 4, 5, 6, 8, 10, or 12 divisions were available. Both left and right scales were used. Figure 2.9-6 presents a flow diagram depicting the processing steps. Several analog tapes from various STDN sites were also processed for detailed systems data. This reduction activity was performed in the MSF and is described in Section 7 of this report.

2.9.3 Testing

The Instrumentation System was subjected to a comprehensive test program to verify system performance and insure vehicle compatibility. Individual components were tested by both the equipment vendor and MDAC prior to installation in the vehicle. Subsequent testing was conducted on installed hardware at the subsystem and system level at MDAC and KSC. All test requirements were successfully completed prior to Skylab launch. The backup Airlock Module designated U-2, was also subjected to subsystem and system level testing in St. Louis. These tests were the same as those performed on U-1; all tests up to altitude chamber were included. Following this activity, the U-2 article was installed in an altitude chamber at MDAC for use in mission simulations and support.

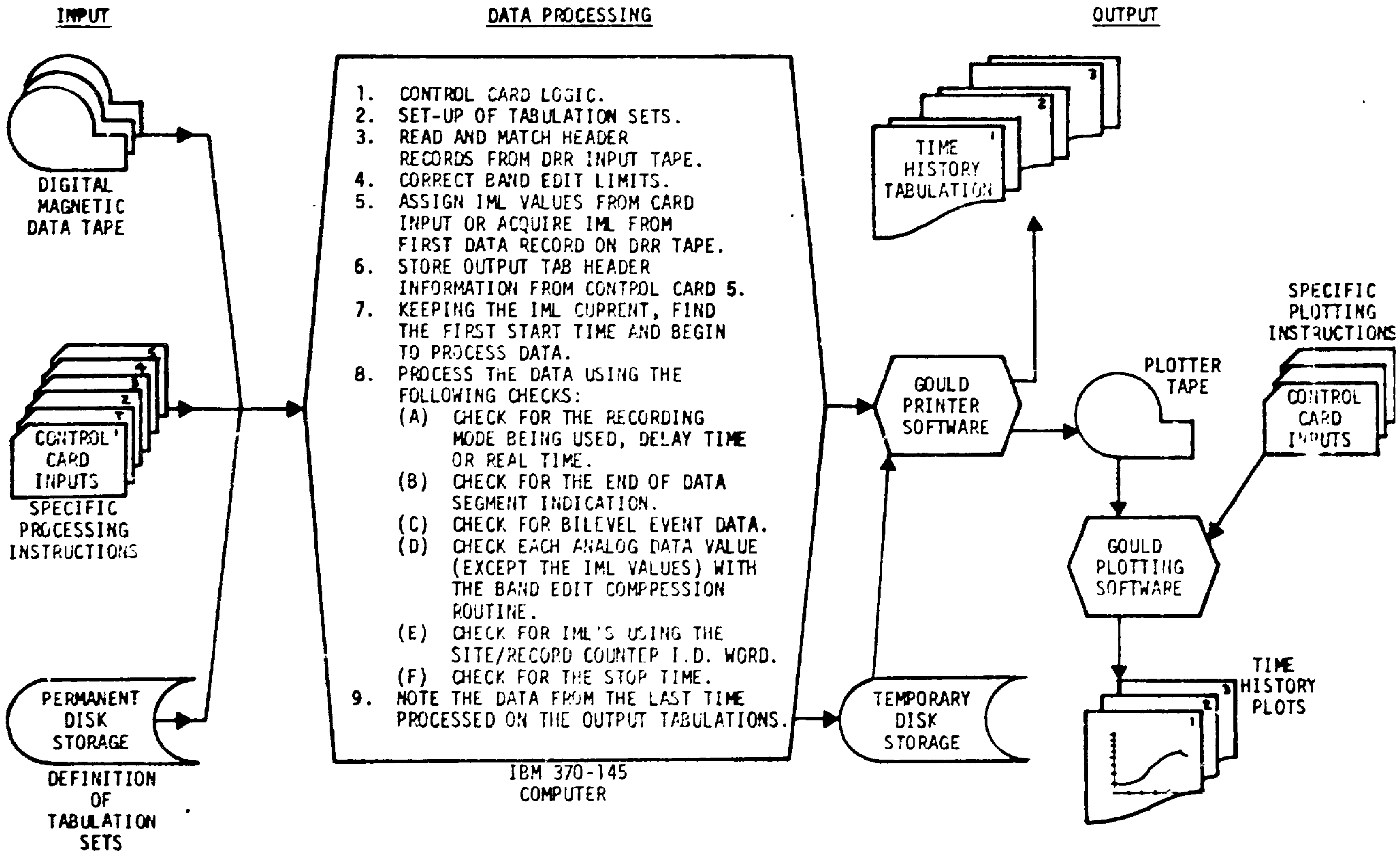


FIGURE 2.9-6 MISSION DATA PROCESSING FLOW (DRR MAGNETIC TAPE)

2.9.3.1 Component and MDAC-E System Tests

Each different hardware item was either subjected to a qualification test or qualified by similarity to a previously tested like item. During these tests the component was exposed to various environmental conditions while the performance characteristics were monitored. The environments included vibration, shock, acceleration, acoustic noise, high and low temperature, pressure, oxygen, atmosphere, humidity, fungus, salt spray, life and RFI. Some of the individual environmental tests were run with the test article in a nonoperating mode, however, performance checks were run prior to and after the exposure period. Problems encountered during qual testing that necessitated equipment design changes resulted in retest at that particular environment and repeat of selected previously run tests that were considered pertinent to the design change verification. All individual components were acceptance tested by the equipment vendor to verify specification compliance prior to hardware delivery; test records were delivered with each part. At MDAC, each delivered item was subjected to a pre-installation acceptance (PIA) test; calibration data was taken during these tests. Uninstalled equipment was re-PIA'd on a periodic basis. Individual component testing resulted in detection and removing suspect parts prior to system assembly; these included tape recorders, PCM telemetry equipment, power converters, and some sensors.

The Instrumentation System was utilized as a source of vehicle systems performance data during final tests at MDAC; GSE displays were also available for a limited number of measurements. This dictated checkout of the Instrumentation System prior to start of formal vehicle testing. Electronics module 3 containing the PCM multiplexer/encoder hardware, electronics module 4 containing DC-DC converters and the tape recorder module were interconnected and operated together. Simulated inputs were provided to the PCM hardware and each channel was individually monitored to verify proper operation. After installation of these modules in the vehicle, there were six major systems tests. These were performed at both ambient and at altitude; some included prime and backup crew participation. The MDA was mated to the AM during the final tests. The DA and FAS were also included in these tests; a simulator was used in place of the workshop. During this testing all system parameters were evaluated versus performance specifications using a validated GSE test complex, with NASA participation. A summary of this testing is shown in Figure 2.9-7.

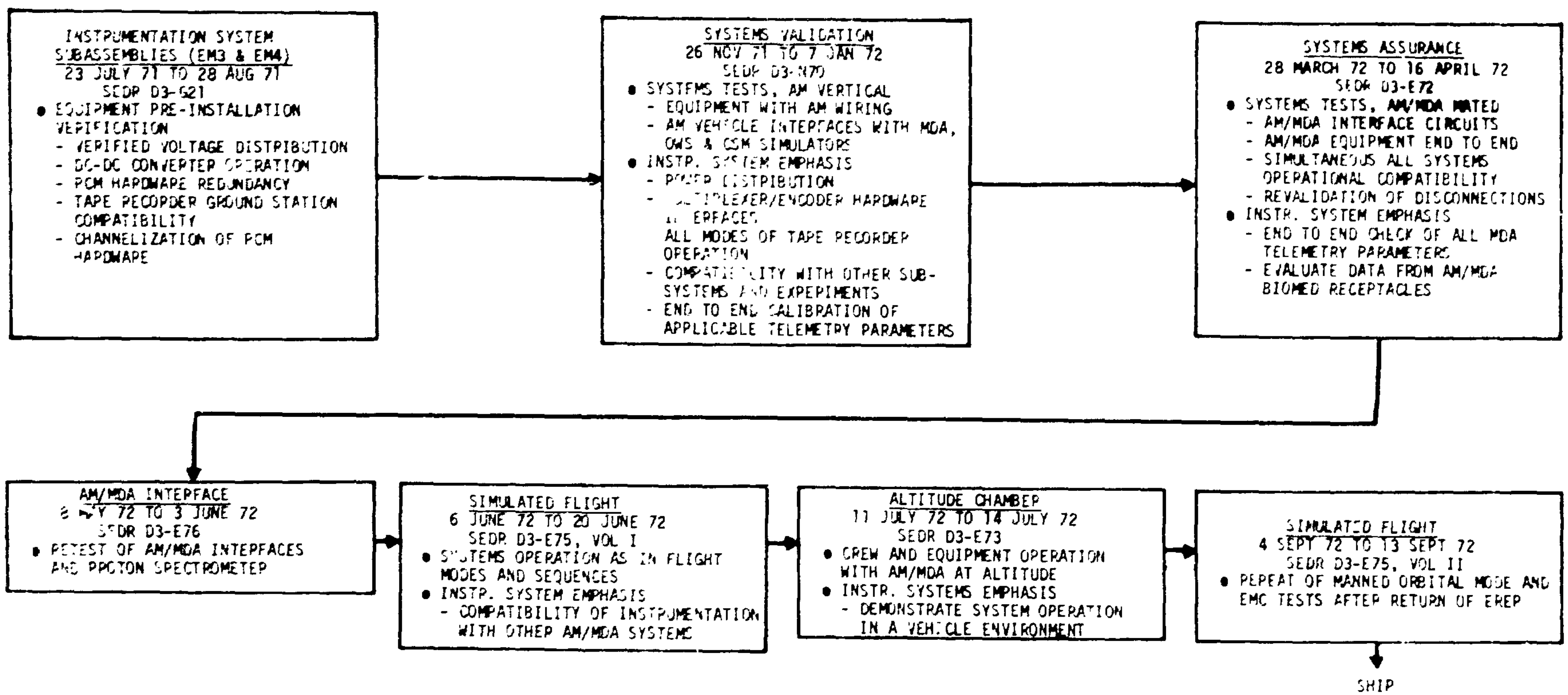


FIGURE 2.9-7 INSTRUMENTATION SYSTEM TEST FLOW – MDAC-E

A. The purpose of Instrumentation System Subassembly, SEDR D3-G21, testing was to verify the integrity of the PCM System and its redundancies prior to installation on the Airlock. The hardware tested included the electronics and relay panels on EM #3 and EM #4, the tape recorder module, tape recorder power and control relay panels. The detailed tests included verification of the power and return wiring on the modules, DC to DC converter operation, redundancy within the PCM system, interface between the ground station and the PCM/tape recorder system, and channelization of multiplexer channels.

B. The Coolant System Leak Test and EPS distribution check, SEDR D3-N46, brought to light two latch relay monitor circuit problems.
Both were minor miswiring problems, corrected by returning the circuits to blueprint configuration.

C. The Instrumentation System test objective in the Systems Validation Test, SEDR D3-N70, included the following:
   - A check of the regulated power distribution through the AM.
   - A total PCM system interface check including redundancy, accuracy, OWS interface verification, RF interface test and tape recorder interface test.
   - Verification of proper tape recorder operation in record, playback, and fast forward modes.
   - Interface compatibility tests with other Airlock systems and experiments.
   - End-to-end calibration check of the majority of the PCM channels.

D. The AM/MDA Interface Test, SEDR D3-E76, uncovered a break in the latch relay monitor circuit chain. Troubleshooting traced the problem to incorrect relay wiring (the open contact was being monitored rather than the closed contact). The circuit was rewired and the retest was satisfactory.

E. The principal instrumentation test objectives in the Systems Assurance Test, SEDR D3-E72, were to perform an end-to-end check of the MDA telemetry parameters, determine any adverse effects of the MDA loads, and perform an evaluation of the data from the AM/MDA bio-med receptacles.

F. During the AM/MDA Interface Test, SEDR D3-E76, several parameters were retested, a reworked interface box, which contained blocking diodes in the course time shift register circuitry, was verified, and a retest of the modified Proton Spectrometer was performed. In the time interval between SEDR's D3-E76 and D3-E75 several minor tests were performed by MPS. A final test of the Experiment M509/Tape Recorder Interface was performed by MPS's 146 and 151.

G. During Simulated Flight Test, SEDR D3-E75, Vol. I, the primary instrumentation objective was to demonstrate that no mutual incompatibility existed between the Instrumentation System and the other AM/MDA systems. A special test was conducted to demonstrate that the PCM split phase converter was capable of driving the long facility lines at KSC. This test, performed in St. Louis, was successful and the split phase converter was shipped to KSC.

H. The instrumentation objectives in the Altitude Chamber Test, SEDR D3-E73, were to demonstrate that equipment mounted on the exterior of the Airlock would operate properly in a vacuum and to verify those instrumentation components in the ECS that could only be checked at reduced pressure.

I. The instrumentation objectives in the abbreviated Simulated Flight Test, SEDR D3-E75, Vol. II, was to verify compatibility with the EREP. Following this last St. Louis test and prior to shipment to KSC, several MPS's were performed on the Airlock. The $PPCO_2$ end plates were reworked to replace the cartridge springs which were subject to permanent set. This rework and cartridge inspection was performed per MPS 214. A pressure transducer and pressure switch that failed during SEDR D3-E75 were replaced and retested by MPS 223.

2.9.3.2 Problems and Solutions

The significant discrepancies resulting from the MDAC-E test phase and their resolution are presented below. A significant discrepancy is considered one which requires component or vehicle modification for resolution.

A. Tape Recorder/Ground Station Sync - During systems validation testing, the telemetry ground station was unable to maintain sync on the delayed time data from subframe 4 and experiment M509. Investigation revealed that the data signal was severely attenuated by high cable capacitance between the tape recorder and data transmitter. The data had a high successive zero bit content due to some unused high sample rate channels in the PCM hardware. The tape recorder output circuitry was modified to be compatible with the actual 3500 picofarad wiring capacitance in the vehicle. Reference voltages were also added to the unused data channels. Retest was successfully performed.

B. Elapsed Time Errors - Random errors were observed on the least significant bit (LSB) of elapsed time data byte contained in subframes 2, 3, and 4 during systems assurance testing. It was determined that the digital data insert pulse from the PCM programmer to the interface box generated an extra pulse on the bilevel signal outputs. This discrepancy was resolved by adding blocking diodes to the 24 bilevel gates in the interface box. Retest with the modified unit resulted in acceptable time data.

C. Inconsistent Gas Flowmeter Data - Erratic flowmeter data was observed on the gas flowmeter outputs during systems assurance testing. Investigation revealed that a combination of gas turbulence and excessive sensitivity to that turbulence was causing the erratic data. The performance of one of the flowmeters was improved by relocating the sensor to an area of lower turbulence and making a slight adjustment in the C&W trip point to avoid false low flow warnings. The testing also identified an incompatibility in the wiring shield grounding configuration which was causing a reduction in the sensitivity of the flowmeters. The shields for all gas flowmeter systems were reterminated. A retest of the modified configuration resulted in acceptable data.

D. Proton Spectrometer Digital Output Errors - During systems assurance testing, it was determined that an impedance mismatch between the proton spectrometer and the PCM interface box caused attenuation of the timing signal to the spectrometer. The spectrometer was returned to the vendor for modification. Verification of this fix occurred during AM/MDA interface retest; again digital errors were noted. The AM side of the interface was changed to expedite solution of this problem. The isolation capacitor between the PCM interface box and the proton spectrometer was replaced by an isolation resistor and resulted in acceptable operation.

E. Inconsistent $PPCO_2$ Detector Readings - Erratic $PPCO_2$ data was first noted during the unmanned altitude chamber runs. All filter cartridges were replaced prior to the manned testing phase. During the manned phase, the crew changed out cartridges in an effort to correct the inconsistent reading, An analysis of these cartridges indicated flow blockage due to caking of calcium sulfate from excessive mositure content. The source of this moisture was traced to excessively high gas flow rate through the cartridge. The inlet connection to the $PPCO_2$ detector was removed from the mole sieve heat exchanger outlet and connected to the inlet of the heat exchanger. The flow rate was reduced and the detector active cartridges were recharged with silica gel lithium hydroxide monohydrate in lieu of sodium hydroxide and calcium sulphate to eliminate moisture clogging.

F. Tape Recorder Erratic Operation - Three related problems were observed on three different tape recorders during the delta simulated flight testing. These resulted in a design modification to the tape recorder. The first was observed via the absence of a record light when S/N 22 recorder was manually commanded on. The cause was determined to be too little radial play in the tape reels resulting in the magnetic tape twisting and coming off an idler pulley. Some metal-to-metal rubbing of the reels also occurred which caused loops in the tape. During this same test period the tape motion telemetry signal from a different recorder was erratic. Inspection of the tape recorder revealed several turns of tape around the capstan drive which permitted the motion signal to be on until the tape slack was taken up. The third problem was noted when the motion monitor continued to indicate tape motion after the recorder was off. Operation of this recorder in the laboratory indicated that when end of tape was reached, the reels slowly oscillated causing the motion monitor to remain on most of the time. This reel cycling was related to the problems noted with the other recorders. These conditions were corrected by reshimming the reel mechanism, modifying the end of tape switch to remove power from the recorder at end of tape, and subjecting the recorders to a confidence test. During this testing, a tape recorder drive mechanism ceased operation. Investigation revealed dry bearings. A record search indicated one of the recorders on the U-1 vehicle, which was now at KSC, was suspected to have dry bearings. This unit was returned for inspection and repair. All three flight units, and the four flight spares were changed-out prior to launch.

2.9.3.3 Launch Site Testing

At the launch complex, the AM/MDA, FAS, ATM, and PS were mated to the OWS and subsequently to the launch vehicle. Integrated systems testing was performed at each step; the AM/MDA was also tested with the CSM. There were 18 major tests in this series that culminated in launch of the Skylab. Figure 2.9-8 presents a summary

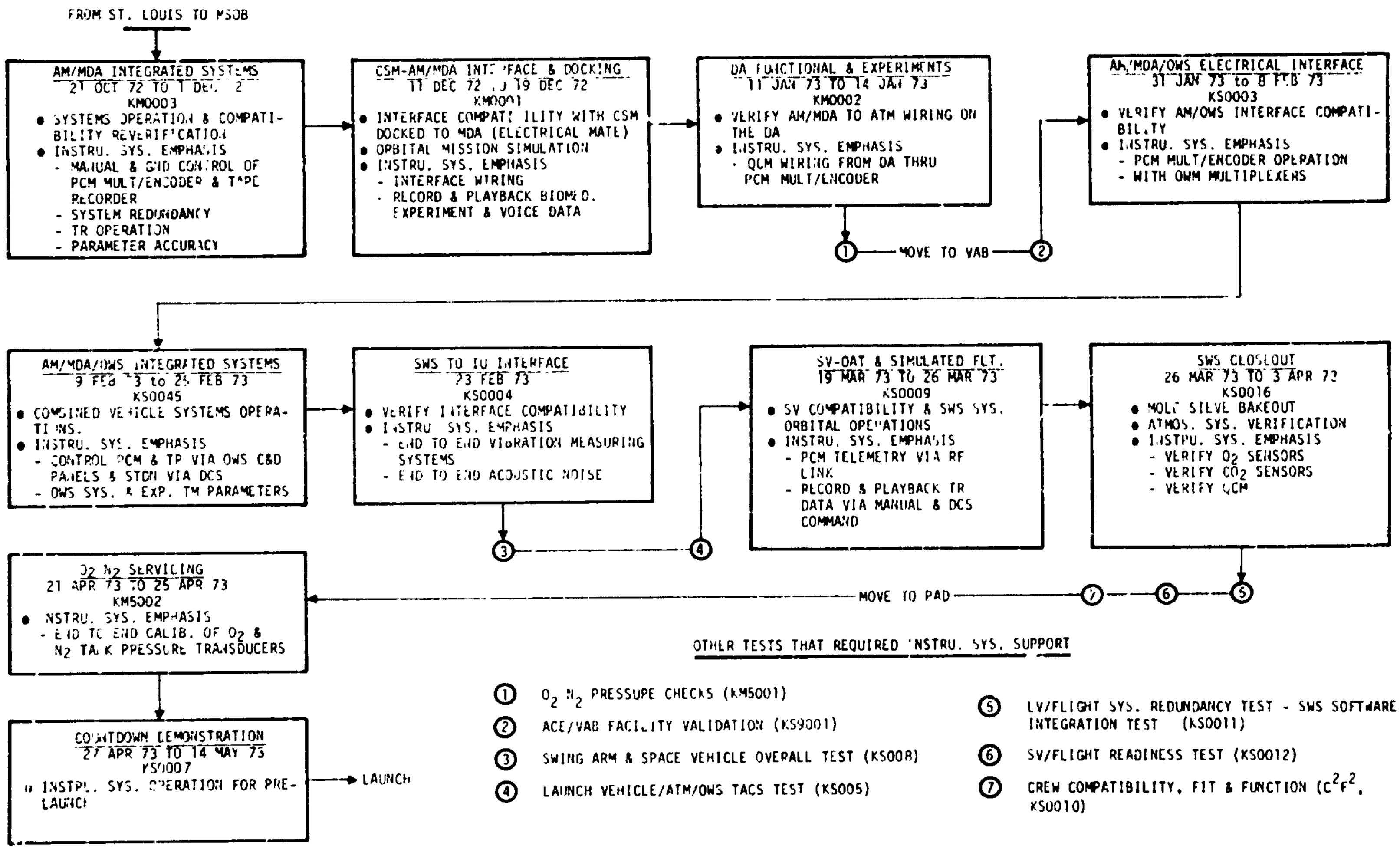


FIGURE 2.9-8 INSTRUMENTATION SYSTEM TEST FLOW – KSC

history of the testing at the launch site. There were two significant discrepancies that occurred during the KSC testing. Other incidents encountered during the KSC test phase were the shortage of O_2 sensors due to mechanical damage from being dropped, and recorder shortages due to the problems noted in paragraph 2.9.3.3(F).

A. Low Level Multiplexer Noise - Noise spikes were observed on the timing lines from the PCM interface box to the PCM multiplexers in the OWS; the noise caused erroneous data. This condition was noted during the AM/MDA/OWS electrical interface testing. Filter capacitors were added to the multiplexer test connectors and the noise spikes were reduced to a level that produced acceptable data.

B. Perturbations on Group 2, +5 VDC Bus - A depression on the +5 VDC bus used for group 2 sensor excitation was observed during the simulated flight and space vehicle overall test. Investigation revealed that this condition resulted from a short circuit within a pressure transducer. To preclude a similar occurrence during flight, all pressure transducers of this design were disconnected from the excitation bus. As a result the following telemetry measurements were not active during the mission.

- ATM Control & Display H_2O Pump #1 Differential Pressure.
- ATM Control & Display H_2O Pump #2 Differential Pressure.
- ATM Control & Display H_2O Pump #3 Differential Pressure.
- AM H_2O System #1 Pump Differential Pressure.
- AM H_2O System #2 Pump Differential Pressure.

A more detailed accout of the vehicle test program and philosophy is presented in Section 5 of this report.

2.9.4 Mission Results

The Saturn Workshop Instrumentation System was activated during the SL-1 launch countdown and continued successful operation during all the remaining mission phases. A total of 6507 hours of operation was accumulated. All functions required of this system were accomplished. The functional success was marred by the failure or suspected failure of 19 instrumentation hardware items. This resulted in discrepant readings on approximately 8% of the telemetry measurements. Less than 14% of these were outright failures, the remainder were operational with minor off-nominal indications. The redundant PCM multiplexer/encoder and DC-DC converter hardware was first activated during the third mission in an attempt to clear a low level channel noise problem. Planned consumable replacement items were utilized as scheduled except for the two tape recorders which failed during the first mission due to ruptured motor drive belts. The STU/STDN and the U-2 Airlock Module were used throughout the flight for special testing and resolution of mission problems. Assuming a 30% STDN coverage, over 50 billion bits of data were sensed and encoded by this system during the three missions. This data was transmitted in both real and delayed time to the STDN by the data transmission subsystem.

Although pre-mission planning called for an expenditure of 4,327 hours of AM tape recorder operation, the nine recorders (seven original and two spares flown up on SL-3) operated for a total of 6925 flight hours. Besides the hours remaining on the three recorders operating at power down, an undefined additional capability remains in the two recorders (S/N 30 and S/N 23) that were replaced while still operating and in the recorder (S/N 28) repaired by the SL-3 crew. Two other recorders (S/N 13 and S/N 22) were repairable using the tape recorder repair kit flown up on SL-4. The average recorder flight time was 769.4 hours and the average life was greater than 1,101.4 hours per recorder. What the actual life might have been cannot be determined; however, it should be noted that six of the nine recorders were operable at power down. The subsequent paragraphs present a discussion of the discrepancies encountered during each of the three mission periods.

2.9.4.1 First Mission

The time period discussed includes the launch and initial storage of the Skylab vehicle, the first crew operations and the storage period following deorbit of the first crew.

All Instrumentation System hardware, except for the tape recorders, Quartz Crystal Contamination Monitors (QCM) and the sensors associated with life support, was powered during launch on DOY 134. After max. "Q", tape recorders 1 and 3 and the QCM's were activated by RF command. The only Instrumentation System discrepancy resulting from the meteoroid shield/solar wing incident during boost, consisted of lower than expected readings from some electrical power system telemetry parameters; usable data, however, was obtained from these measurements throughout the mission.

DOY	DISCREPANCY	CAUSE	MISSION EFFECT	CORRECTIVE ACTION
134	M112, M161, M162, M163 provided lower than expected values.	Meteoroid shield/solar wing problem	Nuisance.	Data was usable thru addition of correction factor
139 140	Unprogrammed automatic switchover of coolant loops (PRI to SEC) (K234).	Unknown. Assumed to be sensor(s) failure.	None. Redundant circuit available.	Redundant circuit, or RF command control used (See Sect. 2.4)
148	Mole Sieve B, $PPCO_2$ inlet - D213/tape recorder interaction.	+24V bus depression caused by recorder mode switching.	Nuisance. Disturbance lasts approx. 2 minutes.	$PPCO_2$ data ignored during tape recorder mode switching.
158	Primary coolant flow-rate measurement (F214) failed.	Unknown. Assumed to be contamination in sensor.	Minor. Alternate data available.	Flow data inferred from temp. and pressure data.
159	Tape recorder, S/N 13, Pos. 1 failed to play-back recorded data.	Broken motor drive belt.	Loss of 3 hours max. of recorded data.	Crew replacement from on-board spares - S/N 22.
173	Tape recorder, S/N 22, Pos. 1 ceased operation.	Broken motor drive belt.	Loss of 3 hours max. of recorded data.	Failure occurred during unmanned period - alternate recorder selected by RF command. Recorder replaced by second crew on DOY 212, (S/N 32).

FIGURE 2.9-9 INSTRUMENTATION SYSTEM SUMMARY - FIRST MISSION

There were four instrumentation hardware items that failed during the first mission. Two tape recorders ceased operation due to broken drive belts; both units were replaced from on-board spares resulting in minimal loss of data. A flowmeter in the primary coolant loop failed on DOY 158. Contamination was suspected to be the cause. The cause of the coolant loop switchovers that occurred early in the first mission was attributed to failure of one or more of the sensors used in the automatic switchover circuit.

During this time period, there were more than 935 tape recorder dumps to the STDN. The premission estimate of tape recorder usage was exceeded by more than 400 hours. This was due to the 10-day delay in launch of the first crew.

Figure 2.9-9 presents a summary history of Instrumentation System discrepancies encountered during the first mission.

2.9.4.2 Second Mission

This time period started with the launch of the second crew on DOY 209 and continued through the storage period following deorbit of this crew.

The system performance during this time period was similar to the first mission, all functional requirements were accomplished. There were two hardware items that experienced partial failure. The fine output from the +X QCM became erratic and went below scale. The associated signal conditioner is believed to have caused this discrepancy. The other failure concerned the intermittent operation of low level multiplexer B in the OWS. No cause was found for this problem. The multiplexer experienced intermittent operation during the remainder of all missions. Other discrepancies included minor off-nominal performance for several other parameters. These are discussed in Figure 2.9-10 in chronological order of occurrence.

There were approximately 1400 tape recorder dumps to the STDN during this mission. One tape recorder was replaced due to excessive bit errors after it had exceeded its specification life.

DOY	DISCREPANCY	CAUSE	MISSION EFFECT	CORRECTIVE ACTION
212	Mole Sieve B, $PPCO_2$ inlet, (D213) provided erratic data after scheduled cartridge replacement.	Assumed to result from unseated O-ring in sensor detector block.	None - Alternate data source from experiment.	$PPCO_2$ O-ring repair kit designed for SL-4 mission.
212	$PPCO_2$ sensor would not lock into place - data is acceptable.	Assumed to be bent index springs in interface connector.	None - Lock position required for launch only.	None - lock position has no effect on data.
215	Low-level Multiplexer B in OWS ceased operation, subsequent operation has been on an intermittent basis	Unknown. Problem cannot be duplicated; temperature is suspected.	Loss of Eng. evaluation data - no mission critical measurements monitored by this multiplexer.	Alternate measurements used for evaluation.
232	+X QCM contamination monitor fine output (M016) became erratic & went below scale.	Unknown. Assumed to be signal conditioner.	None - Coarse output (M015) from QCM was operative.	None - Coarse data used for contamination measurement.
251	AM transfer duct flow-rate (F205) provided gradually decreasing output.	Unknown. Cleaning of associated heat exchangers did not correct problem.	Loss of Eng. evaluation data.	Other measurements used for system status assessment.
256	Tape recorder S/N 28, Pos. 3 had numerous bit errors and loss of sync.	Troubleshooting established tape path was incorrect.	Loss of 3 hours max. of recorded data.	Crew replacement from on-board spares (S/N 23). Crew repaired S/N 28 for spare pending retest.
295	MDA external CM docking port temp. measurement (C0052) exhibited intermittent operation.	Unknown.	None - Not mission critical.	Adjacent temp. measurements used for Eng. evaluation.
310	-X QCM contamination monitor (M018) & +X QCM contamination monitor (M015) provided full-scale readings.	Unknown. Assumed to result from contamination buildup.	Loss of cue for optical surface clean-up.	Past history data used for same purpose.

FIGURE 2.9-10 INSTRUMENTATION SYSTEM SUMMARY - SECOND MISSION

DOY	DISCREPANCY	CAUSE	MISSION EFFECT	CORRECTIVE ACTION
320	Tape recorder S/N 32, Pos. 1 motion monitor (K508) became erratic.	Unknown	None - Does not affect record/ playback process.	None
326	Pri. coolant control valve A outlet flowrate (F212) failed.	Unknown - Assumed to result from contamination in sensor.	None - Not a mission critical measurement.	Temp. and pressure data used to infer flowrates.
349	Excessive noise on first 8 channels of AM low-level multiplexer P	Analytically determined to result from change in turn-on characteristics of second tier switch in multiplexer.	None - No mission critical measurements monitored by this multiplexer.	Visual inspection of strip charts was used to provide usable data.
357 359	Excessive noise on first 8 channels of all AM low-level multiplexers and first 9 channels of programmer (52 measurements total).	Unknown - Suspected to be voltage propagated on the 3MV (15%) reference line connected to the affected equip.	None - Data from all of the multiplexers except "P" recoverable by strip charting.	None
019	Tape Recorder S/N 32 (Pos. 1) Failed to dump data completely.	Unknown - tape recorder had operated for 1450 hours.	Loss of 1.5 hours of recorded data.	Replaced by on-board spare S/N 21.
011 014	OWS High Level Multiplexer "J" exhibited erroneous data output during EREP maneuvers.	Unknown - suspected to be due to high temperature.	None - data can be extrapolated.	None.

FIGURE 2.9-11 INSTRUMENTATION SYSTEM SUMMARY – THIRD MISSION

2.9.4.3 Third Mission

The time period to be discussed started with the launch of the final crew and ended with their deorbit on DOY 39. Discrepancies during this period are addressed in Figure 2.9-11.

There was one major discrepancy associated with this system during the final mission. Fifty-two low level telemetry channels exhibited excessive noise on DOY 357 and continued this way through the completion of the mission. The measurements were contained in all the AM low level multiplexers and in the programmer. The redundant programmer, other half of the interface box and an alternate DC-DC converter were activated for the first time in an attempt to clear this problem. Investigation of this problem revealed that the most probable cause of the problem was a voltage propagated on the 3 millivolt (15%) reference line connected to all the affected multiplexers and the programmers, causing a failure in each of the affected boxes. Usable data was obtained through visual inspection of strip chart recordings of the affected measurements.

The one tape recorder failure during the SL-4 Mission occurred on DOY 19, when S/N 32 failed in the playback mode, after accumulating 1450 hours. S/N 23 tape recorder was replaced on DOY 21 by S/N 14. At the time of replacement, it was still operating satisfactorily but had accumulated 1819 hours.

Other discrepancies included failure of a primary coolant loop flowmeter and some minor off-nominal readings from telemetry measurements.

2.9.5 Conclusions and Recommendations

The Instrumentation System design and the adequacy of the development and test programs associated with this system were reviewed in view of its performance during the Skylab mission. The following conclusions and recommendations resulted from this review.

2.9.5.1 Conclusions

The Instrumentation System was operational during all phases of the Skylab mission and successfully acquired, multiplexed and encoded selected vehicle systems, experiment and biomedical data. Data handling included telemetry downlink, crew displays and PCM hardline for prelaunch utilization.

During the mission, the system sampled and encoded over 1200 input parameters and transmitted in real time approximately 4 x 10^{11} bits of data. An additional 10^{11} bits of data, excluding voice comments, were recorded on the AM tape recorders and transmitted during delayed time data dumps. Over 3650 delayed time data dumps were successfully initiated. Following the resolution of early mission STDN station(s) PCM bit synchronization problems, ground recovery of all data was consistently good.

Although some discrepancies occurred during the mission with certain sensors, low level multiplexers and tape recorders, the system concept and design proved extremely feasible for meeting the unique mission requirements and for handling the large quantity of data involved.

2.9.5.2 Recommendations

The following items were identified during system testing and/or mission support activities and are recommended to further improve the capabilities of the Instrumentation System:

A. Reduce quantity of transmitted data by utilizing data compression techniques, or by providing priority selection capability of data to be recorded/transmitted.

B. Minimize dependence on life limited items, such as tape records for data recovery, to the maximum extent possible during system design.

C. Improve techniques to more accurately measure gas flow. Install gas flow measuring device at a location where gas flow turbulence is minimal.

D. Utilize a differential pressure device to measure liquid flow in lieu of the presently used inline turbine type device. This change would reduce handling and contamination problems encountered with the present sensor.

E. Include the capability to control the temperature of the quartz crystal contamination monitor to improve comparative measurement.

F. Improve the design of the water system differential pressure transducer. The Skylab sensor incorporated a fluorosilicone/dacron diaphragm which did not provide an adequate barrier between the water and the electrical protion of the sensor. A diaphragm consisting of Ethylene Propylene using Terpolymer compound #726 as the elastomer is believed to be compatible with water and stronger than the existing diaphragm.

G. Improve techniques to measure $PPCO_2$. The Skylab CO_2 detector was of 1962 vintage. Smaller and more accurate CO_2 monitors are now available, such as an electrochemical sensor. These new sensors are easier to install, are more accurate because of insensitivity to the atmosphere and have less long term drift, and require relatively little maintenance.

H. Provide discrete measurements via telemetry to allow system configuration and control status information to be available for mission support activities.